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Space Transfer Vehicle Concepts and Requirements NAS8-37856

MARTIN MARIETTA

SPACE TRANSFER VEHICLE CONCEPTS AND REQUIREMENTS
NAS8-37856

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FOREWORD

This report, prepared by Martin Marietta Corporation, is submitted to George C. Marshall Space Flight Center, National Aeronautics and Space Administration (NASA), Marshall Space Flight Center (MSFC), Alabama, in response to the requirements of contract NAS8-37856, Space Transfer Vehicle Concept and Requirements, Data Procurement Document No. 709, DR-4.

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Contract Number NAS8-37856

September 1993

1.0 INTRODUCTION

With the initiative provided by President Bush to expand the exploration and habitation of space, a need arose to define a reliable and low cost system for transporting man and cargo from the earth surface or orbit to the surface of the moon or Mars. The definition of this system was two fold, the need for a low cost, heavy lift Earth-To-Orbit system represents one of the major emphasis and the other is the transportation system itself. Phase I of the STV study analyzed and defined an efficient and reliable system that met the requirements and constraints of both the existing and planned ETO systems and the surface habitation needs, as well arriving at the definition of key technologies needed to accomplish the these missions. The results of the study provide a family of systems that support a wide range of existing and potential space missions. The simplest of the systems support the near earth orbital payload deliveries for both NASA and the DoD, requiring very short mission duration with no recovery of any portion of the system. The more complex systems provided support for the interplanetary manned missions to both the moon and to Mars. These vehicles represented state-of-the art systems that provided safety as well as reusable characteristics that allowed the system to be used in a spaced based mode, the next step in the expansion of manned presence in space.

The space transportation tasks that the STV system was to perform, transport humans with mission and science equipment from Earth to high earth orbits or the surfaces of the moon or Mars, were divided into three phases. (1) Transportation to-and-from low Earth orbit (LEO) being accomplished by the NSTS, ELVs, and new heavy-lift launch vehicles (HLLV) capable of 75 to 150 t cargo delivery; (2) space transfer vehicles providing round-trip transportation between LEO, lunar, and planetary orbits; and (3) excursion vehicles providing transportation between lunar/planetary orbits and their surfaces. Where one mode of transport gave way to another, transportation nodes could be utilized. In low Earth orbit, Space Station Freedom or a co-orbiting platform could serve that need. Elements of the space transfer and excursion vehicles were delivered by the HLLV and crews by the NSTS. Once all the elements were delivered, crews from SSF assemble, checkout, and then launched the vehicle. Following completion of the planned stay at the orbital node, lunar surface, or Mars, the transfer vehicles returned the crew and a limited amount of cargo to LEO where the vehicles were refurbished and serviced for additional missions. Performing the transportation functions in this manner maximized the commonality and synergism between the lunar and Mars space transportation systems and brought the challenge of the exploration initiatives within the reach of orderly technology advancement and development.

Our final report for Phase I addressed the future space transportation needs and requirements based on the current assets, at the time, and their evolution through technology/advanced development using a path and schedule that supported the world leadership role of the United States in a responsible and realistic financial forecast. Always, and foremost, the recommendations placed high values on the safety and success of missions both manned and unmanned through a total quality management philosophy at Martin Marietta.

The second phase of the STV contract involved the use of Technical Directives (TD) to provide short-term support for specialized tasks as required by the COTR. Three of these tasks were performed in parallel with Phase I. These tasks were the Liquid Acquisition Experiment (LACE), Liquid Reorientation Experiment (LIRE), and Expert System for Design, Operation, and Technology Studies (ESDOTS). The results of these TDs were reported in conjunction with the Phase I Final Report

2.0 TECHNICAL DIRECTIVES

2.1 TD06, Advanced Avionics Testbed Connectivity Study

Purpose

Many NASA centers have developed and maintained a variety of R&D laboratories in support of various space programs. By linking the sizable avionics laboratory resources of NASA together in an integrated environment, a powerful new national capability can be directed toward new space initiatives. The SDIO's NTB is an example of an integrated test environment aimed at leveraging existing R&D facilities into a network of federated laboratories. This integrated systems approach provided the SDIO with considerable evaluation, test, and validation capabilities at a reasonable cost. The NTB concept was patterned along the lines of NASA's integrated mission simulation capabilities for the Shuttle Program, but greatly expanded to meet the needs of the SDIO's validation missions. Historically many R&D labs have been built support particular vehicle configurations with limited utility to other configurations. This approach was justifiable when computer systems and interfacing devices were extremely expensive. With the growing cost effectiveness of computer systems related to laboratory operations, it is important for new projects to take advantage of this situation by integrating existing facilities to meet the needs of proposed new programs to ensure the cost effective development and implementation of new technology.

Martin Marietta shall formulate a preliminary concept for an integrated avionics laboratory for future space transportation systems. Trade studies and analysis will be conducted to compare and evaluate existing NASA avionics laboratory capabilities and assess the benefits of using an integrated distributed approach similar to the NTB for combining the capabilities of multiple lab systems. The foundation for concept development will be derived from the following reviews:

(1) a study of the avionics requirements derived by the Civil Space Programs
(2) an examination of existing NASA avionics laboratory facilities which support space transportation systems, and
(3) an examination of existing NASA aeronautics avionics facilities which could be of value to space transportation systems.

The reviews of advanced requirements and existing avionics facilities will be used to identify key sources of avionics testing support (hardware and software) and as sources of data and expertise in various technical areas related to advanced avionics technologies. The results of these investigations will provide definitions of a wide range of avionics test be architecture concepts, at the local level, and at the integrated avionics systems level.

The concept formulation process will include an open, distributed architecture which ultimately, when developed, will allow addressing the following six stages of avionics systems development:

1) The ability to evaluate concepts and technologies employed in the design of transportation systems through the extensive use of software tools.

2) The ability to conduct rapid prototyping (hardware and software) of transportation systems concepts for evaluation

3) The ability to conduct subsystem simulations to explore performance (e.g. dynamics, flight code validation, calibrations, etc.)

4) The ability to conduct en-to-end simulations continuing a mixture of simulated, emulated, and prototype avionics systems for the purpose of validating performance and architectures during the initial phases of program development.

5) The ability to conduct integrated hardware-in-the-loop simulations for the purpose of demonstration, evaluation, validation, and verification.
6) The ability to conduct real-time mission monitoring, analysis, and mission support as required.

The operational concept formulated for this study will define the major and minor node architectures of an integrated avionics test bed which will provide for 1) autonomy of operation for each element in dealing with integration and development issues, within their purview, and 2) an integrated avionics test bed with the capability for interoperability and integration of elements across a wide spectrum of operating ranges. To achieve the interoperability and integration goals of the study, the contractor will define appropriate standards, compatibility, transportability, and other open architecture objectives necessary for an integrated avionics test bed.

Defined Tasks

The contractor shall:
1) Develop a generalized conceptual design that includes the characterization of NASA's existing avionics facilities and laboratories and identification of key resources within the agency which could be of value to an integrated avionics test bed for a space transportation system.
2) Conduct a communication connective analysis of existing NASA systems and identify gaps in capabilities or technology which would not adequately support the concept of an integrated avionics test bed.
3) Develop architectural concepts for an integrated avionics test bed which address transportability of hardware and software components.

Deliverables

The outputs of this study will be two viewgraph presentations, the second of which will include facing page text. Hard copies of the second presentation will be provided as a final report.

2.2 TD07, Lunar Transportation System

Purpose

The contractor shall support the MSFC Lunar Transportation Study Team through the development of key study data. Parametrics, sensitivities, analysis, and trade studies will be conducted to define the vehicle and operational characteristics for an alternative approach to the Option 5 SEI lunar mission architecture. An assessment of technology/advanced development benefits will be conducted using parametric analysis and trade studies to develop options and a plan which can become part of the mission architecture analysis and transportation system definition process. The contractor shall conduct an assessment of mission architectures recommended by the synthesis committee at a level of detail directed by MSFC.

The foundation from which this analysis activity is based but not constrained, includes:
1) Phase I STV Concept and Requirements Study recommendations for LTS configuration design, operation, and technology/advanced development implementation plan.
2) SEI Lunar Outpost Phased Exploration Plan (05 June 1990)

3) SSF/STV Accommodations Study, supplement with recommendation from the 90-day Redesign study.
4) MASE SRD requirements apply except for payload and staytimes.
5) Phased Lunar Approach programmatics and assumptions documented in the January 1991 handout.
6) MSFC-PD will provide the HLLV configuration dimensioned drawings to develop vehicle designs, the HLLV & STS launch costs to perform the Earth recovery mode trade, and the storable engine development costs and programmatics to perform the cryo/storable vs all cryo trade.

Defined Tasks

The contractor will:

1) Develop an alternate LTS concept that uses a rendezvous and docking assembly approach the define the corresponding detailed vehicle design, operations concept, and LCC profile. Parametrics and studies shall be performed to evaluate delivery mass ranges, mission scenarios, propulsion systems, vehicle stage quantities, and technology/advanced development impacts. Develop a lunar transfer vehicle design to perform phase II of the Phase Lunar Approach.
   a) design vehicles for 4 different vehicle configurations:
      i 2 propulsion/avionics (P/A) vehicle (90 day ref. optimized)
      ii Single P/A vehicle
      iii 3 stage vehicle (2 stage lander vehicle)
      iv 3 stage vehicle with storable ascent vehicle
   b) design vehicles for 3 different earth return mission modes (all ground based)
      i earth reenter directly to ground base (consider ground & water recovery)
      ii Aerobrake EOI, STS recovery
      iii All-propulsive EOI, STS recovery
   c) perform sensitivities for the following vehicle parameters:
      i payload size for piloted (0-15t) and cargo expendable (5-50t) modes
      ii lander stay time when base not available
   d) identify design impact if ground based (HLLV crew launch)

2) The contractor shall execute a three phase performance and benefit assessment of Technology/Advanced Development needs for "Option 5" transportation systems.

   Phase one shall assess the technologies within the following categories; they are listed in the order of their priority:
   a. Cryo Systems
   b. Avionics/Software
   c. Engine/Propulsion
   d. Aerobrake
   e. Vehicle In-Space Assembly
   f. Orbit Launch and Checkout
   g. Vehicle Structure
   h. Crew Module
   i. Environmental Control Life Support System
   j. Lunar/Mars Surface Operations
   k. Ground Operations
   l. Vehicle Flight Operations

Evaluate the technologies using the following criteria:

Cost - Life Cycle Cost (LCC) and Nonrecurring
Recurring savings per vehicle
Design, Development, Test, and Evaluation and Research and Technology (R&T)
Benefit - LCC verses R&T Cost
Net Present Value at 5%

Performance - Safety, Reliability, Space Transfer Vehicle (STV), impacts, Launch Vehicle and infrastructure impacts

Schedule - Technology readiness Level 6 by STV PDR, Determine Lead time required to mitigate risk

Other - Reusability, Producibility, Maintainability, Adaptability, Man-rateability, Fault Tolerant Capability, and Space Base Capability

Phase two, perform a more in-depth analysis of a selected group of the technologies from Phase one using the above criteria. The technologies to be studied will be identified by NASA at or near the completion of Phase one.

Phase three, assess the refined technologies with respect to the architectures recommended by the Synthesis group.

3) For the following Lunar Mission Technology Areas:
   a. Aerobrake
   b. Avionics
   c. Cryogenic Engine
   d. Cryogenic Fluid Management
   e. In Space Operations/Assembly
   f. Structures and materials

Perform parametric studies to determine sensitivities to a range of architectures and mission scenarios. This will include:

Development of a "benefit/cost" analysis to the extent feasible given the parametric nature of this task

Utilization of Taguchi methods where applicable

Assessment of qualitative (maintainability, reusability, etc.) parameters as well as quantitative (cost, performance, etc.) to the extent feasible given the parametric nature of this task

4) Support the MSFC Lunar Transportation Study Team in the assessment of synthesis mission architecture recommendations as requested by the COTR.

Deliverables

The contractor shall provide design data including interior layouts an d dimensioned configuration sketches, one top level engineering drawing of the selected configuration for the complete LTS vehicle and each crew module, mass property statements, and sequential statements, a description of selected subsystems, a description of orbital processing (for space based) and regular maintenance tasks, and a listing of the technology, readiness level for selected subsystems. Results will be presented in viewgraph and facing page format at two reviews, the first of which will coincide with MSFC's April 1991 Space Transportation Week and the second to occur son after task completion. Final documentation will consist of hard copies of the final presentation.

2.3 TD08, Integrated Modular Engine Feasibility Study

Objective

The incorporation of integrated modular engines (IME) in vehicles such as upper stages, transfer vehicles, and landers offers attractive benefits which include differential throttling of engines for thrust vector control, modularization of the propulsion components for reliability and maintainability, and improved propulsion system packaging for performance and operational efficiency. The use of differential thrusting allows the deletion of TVC actuators and gimbaled propellant feed ducts. Modularization provides additional flexibility in location and numbers of pumps, thrust chambers and inlet manifolds.

A study shall be performed that defines concepts for space vehicles incorporating the IME, quantifies potential IME benefits, identifies issues that must be addressed, and defines the technical and programmatic actions required to develop the IME.

Defined Tasks

The following tasks shall be performed during this study.

<u>System Definition</u>

The contractor shall develop conceptual designs for a variety of vehicles including upper stages, landers, and transfer vehicles that use the IME concept. The outputs of the task shall include:

An evaluation of the application of the IME concept to a range of space vehicle applications. This evaluation shall include the definition of configuration options, propulsion operating modes (e.g., tank head idle, full thrust, continuous and step deep-throttling), vehicle/propulsion system interfaces, operations impacts, and evolution paths to other vehicles.

An evaluation of different concepts for achieving turbopump, thrust chamber, and feed system redundancy and reliability. The pros and cons of various strategies shall be quantified and evaluated.

A comparison of vehicle performance parameters for the IME concept and a comparable conventional propulsion system. This comparison shall include propulsion system performance (nominal, throttled, off-nominal), power requirements, and vehicle weight and size impacts.

<u>Analysis</u>

The following analyses shall be performed during this study:

The thrust vector control (TVC) requirements imposed on the propulsion system by the vehicles shall be defined. Strategies for achieving these roll, pitch, and yaw TVC requirements using the IME shall be defined and associated propulsion system parameters shall be quantified. A preliminary weight statement shall be prepared to quantify the benefit of eliminating conventional TVC hardware.

The advantages and disadvantages of various engine exhaust expansion strategies (bell nozzles, plug nozzles, etc.) shall be analyzed. Computer analyses shall be performed to determine the effects of expansion surface geometry on performance and flow parameters during the engine burn phase.

Thermal analyses shall be performed to quantify the heat transfer in the expansion surface region that may be used to drive the turbomachinery. This analysis shall address a variety of mission scenarios including full thrust, throttled, differentially throttled, and "module-out".

Technology Development

The contractor shall identify IME technology issues and recommend a program for bringing the IME concept to a level of technical maturity where it becomes a viable option for a space vehicle propulsion system. This program definition shall include technology development objectives, test objectives and requirements, hardware options, resource requirements, facility requirements, and program schedule.

Deliverables

A project plan which defines the contractor's proposed approach shall be submitted to MSFC within two weeks of initiation of the Technical Directive. The contractor shall produce brief, written monthly progress reports, documenting the previous month's activities, plans for the current month, problem areas. An informal estimate of the cost and manpower status will also be provided each month. The contractor shall conduct a mid-term review and a final review at MSFC. A final report documenting the study, including all analyses, trades, assumptions and conclusions shall be submitted.

2.4 TD09, Upper Stage Evolution Study

Purpose

The contractor shall support the MSFC Upper Stages Group through the assessment and development of a strategy for the planning, definition, and implementation of an NLS Upper Stage program. This will be done by looking back at what has been done, what was learned (both good and bad) from what was done, and where we ought to go based on existing and planned launch vehicles and boosters. Commonality of upper stages across all NLS vehicles will be studied and defined where applicable.

Defined Tasks

The contractor will:

(1) Review work already performed under both Space Transfer Vehicle Concept and Requirements Studies and the 90-Day Study. Based on this identify the following:

a. Key Groundrules and Assumptions
   i. Are they still valid?
   ii. Should they be valid?
   iii. Are there any missing?
b. System Drivers
   i. How do they drive the system?
   ii. Should they drive the system?
   iii. Why do they drive the system?

c. Lessons Learned
   i. What should be done?
   ii. What shouldn't be done?
d. Key or Enabling Technologies
   I. What areas need to be developed?

Recommend a strategy for defining the upper stage or family of upper stages for the planned NLS vehicles (20K, 50K, and evolution options) to perform DoD and NASA missions including manned Lunar missions.

(2) Develop for an NLS Upper Stage program that supports the needs and requirements of NASA and the DoD, including system definition and an implementation plan. Based on the current NLS plan of having an upper stage on the 20K and 50K launch vehicles in the 2004 timeframe, the contractor will:
   a. Identify what the upper stages for the NLS vehicles are likely to be.
   b. Identify what the NLS upper stages need to be for NASA's purposes.
   c. Identify if there is a modular approach which gives us a family of upper stages and vehicles (20K, 50K, evolving to support Lunar missions).
   d. Understand and identify what needs to be done for NLS to support NASA's needs.

(3) The contractor shall perform special task studies and analyses, as directed by the COTR, to support NASA and MSFC in the:
   a. Definition of the upper stage(s) for NLS and existing vehicles to perform NASA missions including Lunar missions
   b. Conduct of other Transportation Vehicle related activities.

Deliverables

Results will be presented in viewgraph and facing page format at reviews as required, including MSFC's March 1992 and June 1992 Space Transportation Week. Final documentation will consist of hard copies of the final presentation.

2.5 TD10, Propulsion Avionics Module Study

Purpose

The contractor shall support the MSFC Upper Stages Group through the assessment and development of a strategy for the planning, definition, and implementation of a propulsion avionics (PA) module. This willl be done by defining the PA module requirements based on planned and future mission needs and launch vehicle capabilities to develop a conceptual definition(s) of the PA module(s) for a family of evolvable upper stages.

Defined Tasks

The contractor shall:
(1) Identify the groundrules and assumptions for this study and obtain MSFC agreement with them.

(2) Based on the CNDB91, the National Mission Model, the ETO Options, an SEI Architecture, and any updates to these define a PA module(s) requirements for the following areas:

- Evolution
    - Growth
    - Commonality
    - Duration
    - Missions
- Subsystems
- Technology

(3) Define concepts and conduct analyses and evaluations of a broad range of candidate PA module designs. Concept definition is to include the following:
- Function
- Elements
- Interfaces

(4) For the recommended configurations(s) define the following:
- Operational Model
- Engineering Model

(5) Define the programmatics for the selected PA module configuration(s):
- Program Schedule
- DDT&E
- Funding Profile

(6) Identify the operations involved in the following areas:
- Scenario Commonality
- Flight
- Ground
- Space

Deliverables

The set of groundrules and assumptions and the set of PA module requirements which were agreed to by MSFC and used for this study. For each configuration the following information will be provided:
- Dimensioned drawings of the configuration
- Launch vehicle interfaces
- Mission/Requirements
- Programmatics (Schedule, Cost, DDT&E)
- Operations (Ground, Flight, Space)
- Analysis Results (Databases)

This information is to be included in a final report which will consist of hard copies of the final presentation.

2.6 TD11, Cryogenic Lander Study (FLO)

Purpose

The contractor shall support the MSFC Upper Stages Group through the assessment and development of a one and a half stage lunar lander using cryogenic propellants. This lander will be based on JSC/SEI requirements to the extent possible.

Defined Tasks

The contractor shall:

(1) Use groundrules and assumptions as provided by NASA/MSFC to identify configurations for a stage and a half cryogenic lander.

(2) Generate performance data to allow to downselect to one option.

(3) For the selected configuration generate the following:
- 3 view drawings of the lander configuration
- mass properties of the vehicle
- mission profile
- any performance deltas due to change in engine number (baseline is 4 RL-10 A3s or RL10-A4s)

(4) Additional work as directed by the COTR that is within the timeframe and scope of this task directive.

Deliveralbes

For each final configuration the following information will be provided:
- Dimensioned drawings of the configuration
- Mass properties
- Mission Profiles
- Benefits/drawbacks of 2-5 engines

2.7 TD12, Upper Stage Requirements and Concepts Study

Purpose

The contractor shall support the MSFC Upper Stage Group in the development of an Upper Stage System that is capable of meeting the needs of a changing space transportation environment. This approach will strive toward providing a system that requires:

- Shorter development times by using existing hardware, modular systems/subsystems, and standard interfaces whenever possible.
- Streamlined Operations supporting processing, launching, and operating of multiple Upper Stage/Launch Vehicle configurations.
- Flexibility in mission support and infrastructure integration so that systems can evolve to meet new mission objectives.

To meet these objectives, definition efforts will include:
- Key design and operations requirements based on the capabilities of existing and planned launch vehicles.
- Indepth definition of the conceptual design(s) to include preliminary mass statements, thermal analysis, and stress analysis.
- Integration of functional requirements and conceptual design into an optimized operations concept which reduces mission/payload unique ground processing, on-site vehicle integration, and ground command and control.

The foundation from which this analysis is based, but not constrained, includes:

(1) Upper Stage Evolution Study (TD09) Mission Requirements (Near Earth, Lunar, and Mars)

(2) Upper Stage Evolution Study (TD09) and P/A Module Study (TD10) Groundrules and Assumptions

(3) First Lunar Outpost Feasibility Technical Support and One and a Half Stage Cryo Lunar Lander Study (TD11)

(4) Existing and Planned ELV characteristics as available from NASA and Industry

(5) P/A Module Study (TD10) Requirements and Conceptual Definition

(6) Existing and Planned NASA/Industry System and Subsystem Test Bed Characteristics and Databases

(7) STV, OTV, USRS, and existing Upper Stage performance and design data

Defined Tasks

The contractor shall:

(1) Develop a detailed Study Task Plan that includes key milestones and connectivity to future study activities.

(2) Definition of an Upper Stage System Functional Profile to the system and subsystem level. Ground and flight operation functions for each mission will be defined and analyzed. Payload independence will be determined with a goal to minimize payload specific functions where possible. Profile to include detailed mission event sequencing and timelines.

(3) Based on Upper Stage DRMs and ETO capabilities, conduct requirements analysis to define system and subsystems requirements for:
- Performance
- Operations
- Interfaces & Integration
- Programmatics
- Technology Availability and Development

Parametric analysis will be utilized to enhance design flexibility. Analysis will provide identification of resolution to design and operations drivers.

(4) Provide detailed conceptual definition based on system and subsystem functions and requirements. Definition to include:
- System and Subsystem concept design and layouts
- Payload/Launch Vehicle Interfaces
- Mass Properties

(5) Develop and submit for MSFC authentication a System Requirements Document/Upper Stage System "A" Specification.

(6) Conduct studies and analysis to define an innovative and efficient Upper Stage Operations concept. Approach is to be capable of processing, launching, and operating multiple Upper Stage/Launch Vehicle configurations.

(7) Develop detailed engineering data to Pre-prototype level. Package to include:
- S/K drawings (system & subsystem)
- Hardware acquisition recommendations (shopping list)
- Detailed test/qualification plan
- Specialized Analysis
  - thermal
  - dynamic
  - stress
  - material
  - etc.

(8) Perform special task studies and analyses, as directed by the COTR, to support NASA and MSFC in the:
a) Definition of upper stages for planned and existing launch vehicles to perform NASA missions including Near Earth, Lunar, and Mars missions.
b) Conduct of other Transportation Vehicle related activities.

Deliverables

The contractor shall provide:
- Detailed Study Plans
  - Initial and an update near task completion
  - Identification of additional studies needed and timeframe needed
- Functional Profile/Events Sequence/Timelines
- System Requirements Document
- Recommended System Concept
  - dimensioned configuration drawings/layouts
  - preliminary interface document
  - mass properties
  - Pre-prototype engineering
- Programmatics
  - cost
  - schedule
  - technology
- Operations Concept (Ground, Launch, Flight)

Results will be presented in viewgraphs and facing page format at two reviews, the first occurring in mid to late June and the final review occurring early in October. Final documentation will consist of hard copies of the final presentation.

2.8 TD13, Phase II, Upper Stage Requirements and Concepts Study

Purpose

Previous Space Transportation Vehicle (STV) Contract activities addressed three areas: Space Exploration Initiative (SEI), Upper Stages, and Technology. Tasks defined in this Technical Directive (TD) build on previous efforts. Tasks include allocating NASA requirements to the TLI/Upper Stage subsystem level, conducting studies to determine internal relationships and operations concepts, and further investigation of Vehicle Health Management (VHM).

Defined Tasks

Tasks, in this TD, are defined to meet the needs of three customers. A requirements analysis task supports the First Lunar Outpost (FLO) Systems Engineering Team. Upper Stage tasks provide support to the MSFC HLLV Product Development Team (PDT) as well as the FLO Systems Engineering Team. And , in the area of Technology, tasks focus on VHM to support the intercenter Integrated Vehicle Health Management (IVHM) Team.

<u>Space Exploration Initiative</u>
1) The STV Contractor shall allocate applicable FLO functional and performance requirements to the TLI/Upper Stage conceptual design. Also, the STV Contractor shall document Element Level Interface requirements. The FLO Earth to Space (ETS) Systems Engineering Team will provide the system and subsystem requirements. This activity supports the FLO engineering design reviews.

<u>Upper Stages</u>
1) The STV Contractor shall conduct trade studies regarding TLI/Upper Stage subsystems to identify programmatic and technical issues and options. The STV Task Team shall document the results of these studies and provide recommendations to the NASA FLO ETS and HLLV SEI Vehicle Systems Development Team (SDT) and SEI Engine Product Development Team (PDT).
2) The STV Contractor shall allocate requirements, provided by the NASA FLO ETS and HLLV SEI Vehicle Systems SDT and Engine PDT, to the TLI/Upper Stage Concept. Based on these requirements, and results of the TLI/Upper Stage trade studies, the STV Contractor shall refine and further define the TLI/Upper Stage conceptual design. Conceptual design shall include, but is not limited to:

- System and subsystem concept design and layouts
- Payload and Launch Vehicle Interfaces
- TLI/Upper Stage mass properties

3) The STV Contractor shall study innovative approaches to the Upper stage Operations Concept. The study shall focus on programmatic and technical benefits derived from the P/A Module (TD10) when used in processing, launching, and operating multiple Upper Stage/Launch Vehicle configurations.
4) The STV Contractor shall perform special task studies and analyses, as directed by the COTR, to support NASA and MSFC in the (a) Definition of upper stages for planned and existing launch vehicles to perform NASA missions including Near Earth and Lunar missions and (b) Conduct of other Transportation Vehicle related activities.

<u>Technology</u>
1) Products from TD12 included quad chart descriptions and supporting rationale and prioritization of near term technologies related to Integrated Vehicle Health Management (IVHM).The STV Contractor shall recommend demonstrations that quantify improved cost and reliability, and performance gained through VHM technologies. Recommended demonstrations shall focus on three target vehicles: Titan III, FLO ETS, and the TLI/Upper Stage with early emphasis on Titan III.
2) The NASA FLO ETS and HLLV Engineering Teams will provide functional and performance requirements. The STV Contractor shall analyze these requirements and extend the VHM system conceptual definition to the subsystem level.

Deliverables

The contractor shall document the results of the tasks in a bound set of 8 1/2" x 11" charts with facing page text. Deliverables shall include, but are not limited to:

- Detailed Study Plans
- Functional Profile or Events Sequence or Timelines at the TLI subsystem level
- Requirements Traceability Documentation for allocated system and subsystem requirements
- Dimensioned configuration and layout drawings
- Mass properties list
- Programmatics - costs, schedules, technologies, issues, recommendations
- Processing and Operations Concepts for Ground, Launch, and Flight mission phases
- Interface Requirements Documents
- IVHM analysis results, recommendations, and rationale.

2.9 TD14, FLO TLI Study

Purpose

Space Transportation elements defined by Space Transfer Vehicle (STV) contract studies include upper stages, transfer vehicles, and landers. These elements can accomplish design reference missions (DRMs) ranging from Low Earth Orbit (LEO) to interplanetary exploration. Common subsystems have been emphasized in these studies. A prime example of a common subsystem is the Propulsion/Avionics (P/A) Module, defined under Technical Directive #10,. The P/A Module has been applied to upper stages for Titan IV, National Launch System (NLS) 2, NLS 3, and a Trans Lunar Injection (TLI) stage. This Technical Directive defines a task for an architectural analysis to provide the "big picture" of how these conceptual elements meet the needs of today and tomorrow.

Defined Tasks

Task 1: Architecture Assessment:
The contractor shall conduct a space transfer vehicle architectural analysis of mission and system requirements to layout a roadmap that will enabler NASA to plan future space transportation systems. The architecture shall identify time periods, evolution capability, requirements, cost, etc. It shall focus on near term missions and explain the evolution path necessary to accomplish far term missions. The contractor shall assist Marshall Space Flight Center (MSFC) in the integration of the upper stage architecture into the overall space transportation architecture. Ground rules and assumptions will be determined in a Technical Interchange Meeting (TIM) between the contractor and MSFC representatives.

Task 2: First Lunar Outpost
The contractor shall perform tasks necessary to complete the First Lunar Outpost (FLO) effort. This effort will focus on the system requirements, interfaces, functional flow, operations, programmatics, and subsystem definitions of the TLI stage.

Task 3: Special Studies
The contractor shall perform special task studies and analyses, as directed by the COTR, to support NASA and MSFC. Studies will focus on upper stages for planned and existing vehicles to perform NASA missions and other transportation vehicle related activities.

Deliverables

The contractor shall document the results of the tasks in a bound set of 8 1/2" x 11" charts with facing page text. Deliverables shall include, but are not limited to:

- TLI Data Package for FLO, as defined by TD #12 and TD #13
- Upper Stage concepts and system requirements derived from architectural analysis

- Programmatics - costs, schedules, technologies, issues, recommendations
- Processing and Operations Concepts for Ground, Launch, and Flight mission phases
- Roadmaps depicting upper stage systems, technologies, and development infrastructure

2.10 TD15, Fluid Acquisition and Resupply Experiment (FARE) Data Analysis and Consultation

Purpose

The Fluid Acquisition and Resupply Experiment (FARE) flew aboard STS-53. Two acrylic tanks, a flowmeter display, accelerometers, and video equipment comprised the experiment. A blue fluid, simulating propellant passed from a supply tank to a receiving tank. Experimental data includes videotapes and 35 mm photographs. This Technical Directive (TD) defines tasks for data analysis and consultation to the MSFC FARE team.

Defined Tasks

Task 1: Data Analysis
The contractor shall analyze FARE videotapes, accelerometer graphs, crew annotations, and still photographs. Analysis shall include a broad review of the entire data set and detailed evaluations as determined by Telecon with the MSFC FARE team. Evaluations shall include correlation between test results and analytical predictions and computational fluid dynamic analysis. During the period of performance, the contractor shall maintain communications with the MSFC FARE team for consultation and discussion of data analysis.

Task 2: Process Improvement
The contractor shall provide commentary regarding the FARE video tapes, and other data, identify problems encountered during analysis, lessons learned, and define applications of experiment results for flight systems.

Deliverables

The contractor shall prepare a brief Analysis Plan that defines the approach to accomplishing Task 1 and 2. The NASA FARE team will have ten (10) working days to revise the plan. The contractor shall document information derived from Task 1 and Task 2 in a final report. The final report shall contain texts with supporting figures and tables.

2.11 TD16, Upper Stage Requirements and Architecture Study

Purpose

Three products from Technical Directive 14 provide the framework for accomplishing analysis tasks related to upper stage systems, technologies and infrastructures. These products include architectures, an upper stage market analysis, and upper stage technical requirements document (TRD). This technical directive will refine these products by determining quantitative requirements associated with architectural elements and establishing relationships between the products.

Defined Tasks

Task 1: Architecture Assessment
The contractor shall refine the architectures developed under TD 14. Architectural elements include upper stage configurations, technologies and infrastructures. The contractor shall assess

these architectures to determine options that lead to cost effective upper stages and provide an evolution path to exploration class vehicles. The contractor shall assist Marshall Space Flight Center (MSFC) in the integration of the upper stage architecture into the overall space transportation architecture. Ground rules and assumptions will be determined in a Technical Interchange Meeting (TIM) between the contractor and MSFC representatives.

Task 2: Upper Stage Market Analysis
The contractor shall analysis the upper stage market to determine the need for upper stage capabilities and programmatic requirements. This analysis includes an assessment of existing upper stages and the economic environment. Results of this assessment will enable the Space Transportation Exploration Office to define upper stage programs. In addition to defining program requirements, the contractor shall define approaches for gaining advocacy of resulting upper stage programs.

Task 3: Requirements Analysis
The contractor shall determine mission and system level requirements for an Upper Stage Program. System requirements must support program requirements determined in Task 2 and provides parameters and constraints that the architectural elements in Task 1 are defined against. The contractor shall perform requirements analysis which provide an understanding of the requirements impact with respect to performance, schedule, cost , technologies, as applicable. The requirements analysis will also serve to provide rationale for values in the TRD.

Task 4: Special Studies
The contractor shall perform special task studies and analyses, as directed by the COTR, to support NASA and MSFC. Studies will focus on upper stages for planned and existing vehicles to perform NASA missions and other transportation vehicle related activities.

Deliverables

The contractor shall document the results of the tasks in a bound set of 8 1/2" x 11" charts with facing page text. Deliverables shall include, but are not limited to:

- Upper Stage Architectures - Packages include a graphic roadmap identifying configurations, technologies and infrastructures with supporting material for each element of the architecture. These architectures provide the structure for the other products of this TD.

- Upper Stage Market Analysis - This deliverable includes assessments of existing and proposed upper stages in terms of capabilities, costs, schedules, technologies, issues, etc. This analysis must provide a basis for recommended programs that fulfill specific needs determined by the market analysis. This analysis shall provide traceability to the architectures and technical requirements.

- Technical Requirements Document (TRD) - Top level requirements document accompanied by results of the supporting requirements analyses and sensitivities performed during the TD.

2.12 TD17, Spacecraft Technology Center Transfer

Purpose

The introduction of the Space Transfer Vehicle (STV) contract states: "This new study will attempt to utilize the emerging launch vehicle definition and the latest mission scenarios to define a flexible, high performance, cost effective, evolutionary upper stage program for NASA and the United States and provide information necessary to proceed with system definition and planning." Previous technical directives (TD) defined program and performance requirements for upper stage systems. To proceed with system definition and planning, MSFC needs the requirements in an electronic format and a the necessary tools to analyze, process, and configure the requirements. This Technical Directive (TD) defines work that results in an upper stage requirements analysis and management system.

Defined Tasks

Task 1: Upper Stage Requirements Database Implementation
The contractor shall port the essential upper stage system program and performance requirements into a Systems Engineering Data Base (SEDB). The Upper Stage SEDB shall provide the capability to analyze the impact to relationships when specific requirements are changed. The contractor shall supervise the installation of the database on a MSFC host computer and provide training to MSFC personnel on the use of the database.

Task 2: Upper Stage Requirements Analysis and Management System
The contractor shall develop a plan for the procurement, delivery, installation of a requirements analysis and management system. Plans shall also describe "hands on" system training of MSFC personnel.

Task 3: Special Tasks
The contractor shall perform special task studies and analyses, as directed by the COTR, to support NASA and MSFC. Studies will focus on upper stages for planned and existing vehicles to perform NASA missions and other transportation vehicle related activities.

Deliverables

Deliverables shall include, but are not limited to:

• Upper Stage Requirements Database - A Systems Engineering DataBase containing Upper Stage System requirements developed under the Upper Stage Architecture Study.

• Upper Stage Requirements Analysis and Management System - The contractor shall deliver the following system components in an electronic format compatible with the platform that will host the requirements analysis and management system. The contractor shall develop a procurement plan that establishes procurement and delivery milestones and describes the support necessary for system installation and training.

1. System Engineering Data Base (SEDB) Management System
2. Oracle for Sun SPARC capable of supporting TBD users
3. An option to upgrade Oracle for an additional TBD users
4. RDD100/SD - one (1) copy (Sun IPX workstation)
5. RDD100/RE - one (1) copy (Sun IPX workstation)
6. RDD 100/DVF - one(1) copy (Sun IPX workstation)
7. 4th Dimension for the MacIntosh - TBD copies

• Installation and Training - A detailed plan explaining procedures for installing, testing, and training MSFC personnel on the use of the SEDB and associated software. The contractor shall perform installation, testing and training functions in accordance with the plan.

3.0 CONCLUSIONS AND SUMMARY

Cost analysis of existing launch systems has demonstrated a need for a new upper stage that will increase America's competitiveness in the global launch services market. To provide a growth path to future exploration class STV's, we must develop near-term low-cost upper stages featuring modularity, portability, scalability, and evolvability.

NASA should establish a concurrent engineering development environment that leverages existing resources within government and industry. The STV study has developed concepts for this concurrent engineering development environment. Such an environment requires executive level support and financial commitment from all participants. With the proper tools and increased communication, future upper stage projects can decrease development costs. The Clinton administration's NII Initiative can provide the communication backbone necessary to implement the network.

We can reduce avionics life cycle costs and systems operation costs through IVHM technologies. IVHM development and demonstration programs should capture resulting data and requirements in a data base accessible through the concurrent engineering development environment. Also, the development environment should provide design tools that assist designers to incorporate IVHM technologies in upper stage designs.

A team comprised of industry and Government should develop an IME/PA module. A module combining the benefits of the IME and P/A would provide a scalable platform for future upper stage systems. Through scalability, an IME/PA module can offer optimized engine thrust for each mission. In the immediate future, NASA could initiate a ground demonstration program that results in three P/A module test articles corresponding to the sizes of upper stages described in this paper. These test articles could function as engine test stand fixtures for a variety of engine sizes and multiple engine configurations.

These recommendations define a program that: (1) leverages ongoing activities to establish a new development environment, (2) develop technologies that benefit the entire life cycle of a system, and (3) result in a scalable hardware platform that provides a growth path to future upper stages.

TECHNICAL DIRECTIVES APPENDIX

Technical Directive 06

Advanced Avionics Testbed Connectivity Study

Advanced Avionics Test Bed - Agenda

• Introduction	D. Scruggs
• Networking for Avionics Laboratories	S. Driskell
• Analyze Existing Avionics Laboratories and Determine Capabilities	S. Driskell
• Integrated Avionics Testbed Concept	D. Scruggs
• Observations Recommendations and Conclusions	D. Scruggs

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TD006 - Advanced Avionics Test Bed Connectivity Study

Title: Advanced Avionics Test Bed Connectivity Study

Customer: NASA/JSC (EG)- Don Brown (713) 483-8241,
NASA/MSFC (PD) Cynthia Frost (205) 544-0268

Purpose: Develop a Conceptual Avionics Laboratory Communications Connectivity Concept which includes the Characterization of Existing NASA Avionics Laboratories and Conduct a Communications Connectivity Analysis Identifying Gaps in Capabilities or Technologies. The Result of the Study is the Development of Architectural Concepts for an Integrated Avionics Test Bed.

Contract Specifics: Task Description (TD006) Under MSFC Space Transfer Vehicle (NAS8-37856), Funded by Marshall Space Flight Center
Period of Performance: 6 Months

MMAG Personnel:
Principal Engineer - Steve Driskell (303) 971-7074
Architecture Study - Rob Mason (303) 971-6489

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TD006 - Advanced Avionics Test Bed Activities

Characterize Conceptual Design of NASA's Existing Avionics Facilities and Laboratories

- Identify Key Assets for Analysis Through A Survey of the Related Labs, Discussions with Center and Lab Personnel, and Use of Published Connectivity Documentation

- Identify Gaps in Capability or Technology which would not Support Future Integrated Avionics Test Beds Connectivity for Future Programs such as Space Exploration Initiative / Space Transportation System

- Develop Connectivity Concept for Avionic Laboratory Architectures and Protocol Compatibility Focusing on Hardware and Software Transportability

- Publish a Midterm and Final Presentation That Summarizes the Study Activities, Results, And Conclusions

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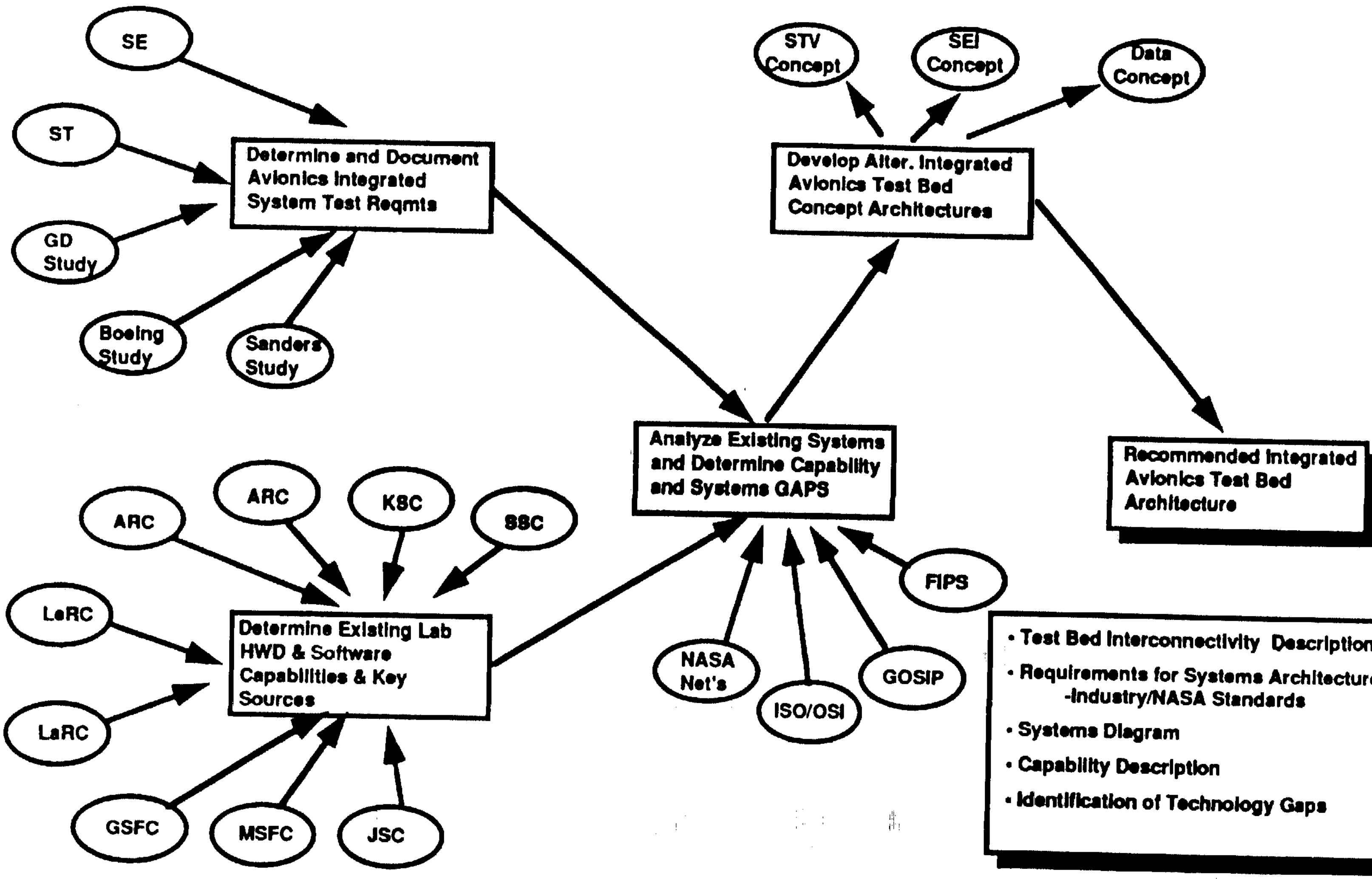
Task Flow - TD006



- Test Bed Interconnectivity Description
- Requirements for Systems Architecture
  -Industry/NASA Standards
- Systems Diagram
- Capability Description
- Identification of Technology Gaps

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Networking for Avionics Laboratories

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DS112491-1B

Networking for Avionics Laboratories

- The Following Questions needed to be Addressed by This Study!

  - What is a Computerized Network?
  - What is an Avionics Laboratory Network?
  - Why Build an Avionics Laboratory Network?
  - What Standards and Security Requirements should be Used?
  - What Kinds of Networks are Available to NASA Organizations and How are NASA Unique Networks Organized ?

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What is a Network?

- Only Computerized Networks were Considered as Candidates for This Study.

  - Definition

    Two or More Computers Geographically Distributed, Usually Capable of Parallel Processing, Multipoint Access, with a Centralized Access Facility.

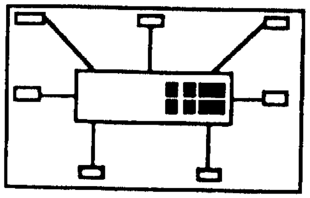
    - LAN

      Local Area Network is a Limited Area Coverage System in a Closed Geographic Area.

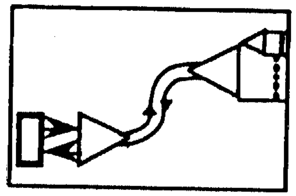
    - WAN

      Wide Area Network has a Large Area Coverage Using Wide Band Communications Systems (Laser, T1 and Satellites)

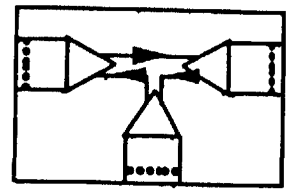
Typical Net/ Internet Topologies



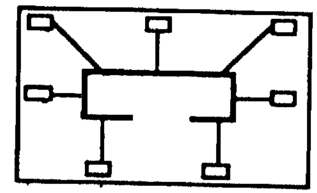
Star



Tree



Ring



Bus or Broadcast

Internet Connection Equipment

Increasing Capability, Intelligence, Cost

- Repeater - Extends Transmission Distance by Amplifying the Signal
- Gateway - Isolates the Local Area Network from Backbone Message Traffic not Directed to the LAN
- Router - Acts as a Gateway and also Provides Network Management / Packet Direction Services
- Bridge - Translates between a Local Area Network Protocol and a Backbone or Wide Area Network Using Different Protocols
- Brouter - Provides Router Services as Well as Bridge Services

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What is an Avionics Laboratory Network?

Networks with Open Distributed Architectures are Capable of Transporting System Analysis and Processing at the Following Levels:

- Evaluate Concept, Technology and Design With Software Tools
- Rapid Prototyping (Hardware/Software)
- Subsystem Simulation — Dynamic Performance, Flight Code Validation Calibration
- End - To - End Avionics Simulations - Including Architectures,
- Integrated Hardware Simulations for Demo, Eval, Validation and Verification
- Real Time Mission Monitoring, Analysis & Mission Support

Each Level should be Design to Support the NASA Methodology for Program Research and Development

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Why Build an Avionics Laboratory Network?

<u>To Access A Wide Range of Information Resources and Services!</u>

- Increase Productivity by Increasing Information Availability
- Shared Resources
  - Printers
  - Computers
  - Large Data Storage Devices
- Ease of Access to Applications Software
  - Common Access to Latest Program Versions (Through a Shared Storage)
  - Extremely Large Custom Programs Can Reside in a Shared Central File
  - Locate, Retrieve and Link Anywhere
  - Interactive with User
- Access to Common Files or Information
  - Common Data Base Access
  - Single Point Access to Information
  - Open System with Shared Reports and Projects
- Achieve Distributed Simulation and Technical Information Utilization
  - Integration of and Interfacing with a Variety of Systems

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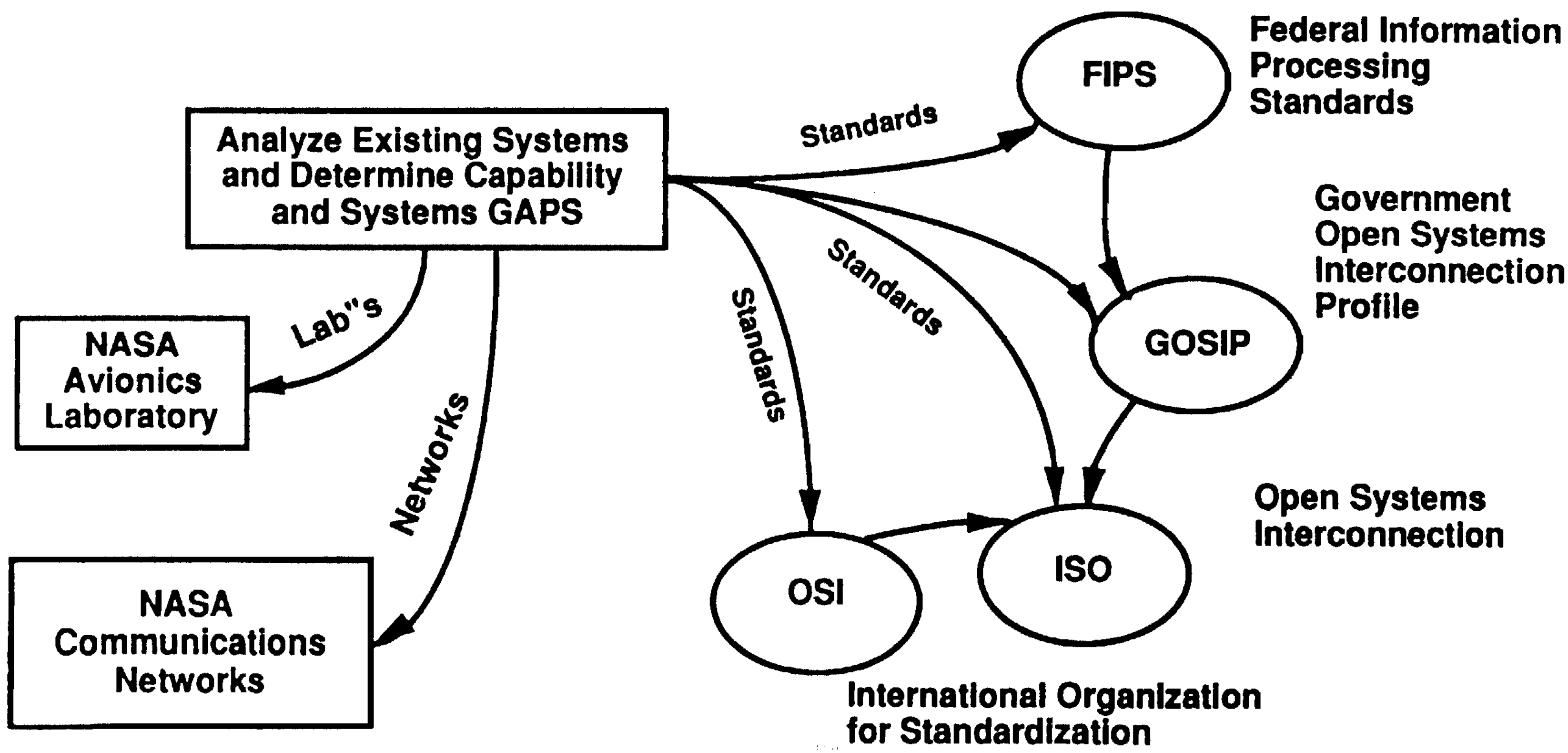
What Standards and Security Requirements should be Used?

- World Wide Connective Standards have Matured during the 1980's.
- Government Standards Center on:
  - Federal Information Processing Standards
  - Government Open Systems Interconnection Profile
- International Standards Center on:
  - International Standards Organization
  - Open Systems Interconnection

Analysis of Existing Systems and Capabilities

- Three Key Inputs to the Connectivity Study
  - Federal Information Processing Standards
  - The Capability of Existing Facilities
  - Existing and Planned NASA Communications Networks



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FIPS

- Federal Information Processing Standards Publication
  - National Bureau of Standards
  - Government Open Systems Interconnection Profile (GOSIP)
- Category: Hardware and Software Standards
- Subcategory: Computer Network Protocols
- Seven Protocol Layers Have Been Identified for Open Systems Interconnection
- International Standards Organization Has Developed These Guidelines in Conjunction With The Institute of Electrical And Electronics Engineers Inc. (IEEE) and the Consultative Committee for International Telegraph and Telephone (CCITT). Representatives of this group supported the National Institute of Standards Study for the Creation of These Requirements. An Annual Update Meeting is chaired by the NBS.

GOSIP

The Following Protocols are Currently Scheduled for Version 2 of GOSIP:

1. Virtual Terminal (TELNET and Transparent Profiles)
2. ES-IS Protocol
3. Connection Oriented Network Service
4. ODA/ODIF

Protocols Scheduled for Version 3 of GOSIP

1. Directory Services
2. Interim Network Management
3. ISDN
4. Virtual Terminal (Page, Scroll and Forms Profiles)
5. Connectionless Transport Protocol
6. MHS Extensions Based on 1988 CCITT Recommendations
7. FTAM Extensions
8. FDDI

The Purpose is to Assist Federal Agencies in Planning for Acquisition and Implementations of OSI Protocols.

International Standardization Organization

<u>IAW The GOSIP Plan -</u>

<u>Requirements</u>: Selected Protocols are GOSIP Mandatory

Intermediate Layer Protocol - Connectionless Network Layer - Transport Class 4 and Session Layer 5

Public Data Network Messaging -Transport Class and Connection Oriented Network Service (CONS), Message Handling Systems (MHS)

All Applications (Except Messaging) - Presentation Layer 6 and Associated Control Service Elements for File Transfer and Management (FTAM,)

Both May Be Specified

<u>Purpose</u>: Minimization of Nonstandard and Proprietary Systems and Applications With the Intention of Creating a Universal Interconnectivity for Government and Industry.

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Open Systems Interconnection

- OSI Network Management is Described in Detail within an NBS Report, Network Management Functional Requirements. Descriptions for an Overall Management Architectural Framework Model Include:
- Faults • Accounting • Configuration • Security • Performance Management Services.
- For an Interim OSI Network Management Specification, the GOSIP Priorities Are Configuration, Fault, and Performance Management for Layers 1-4 fo the GOSIP Protocol Suite for End Systems and Intermediate Systems. A Requirement Also Exists to manage Other Network Elements. A Summary of OSI Subjects includes the Following:
  - Seven Layer Definition
  - Protocol Layer Requirements
  - Future Protocol
  - System Security
  - Purpose of ISO Standards

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The Seven Layers ISO / GOSIP

Seven Layer - Open System Interconnectivity

1. Physical Layer - Data Link Entity Data Transmission Connection, Regulates Network Access, Encodes & Decodes

2. Data Link Layer - Adjacent or Broadcast System Communication, Performs Formatting, Error Checking, Addressing, etc. Ensures Data Transmission Accuracy.

3. Network Layer - Message Routing & Relaying, Flow Control, Load Leveling Network Services Independent of Network or Transport Protocol.

4. Transport Layer - Reliable Transparent Data Transfer in Cooperating Sessions, Provides Session Performance, Detects & Corrects Errors, Regulates End-To-End Flow.

5. Session Layer - Manages & Synchronizes Application Data Exchange Using Transport Connections

6. Presentation Layer - Specifies Syntax of Transferred Data, Including Application Structure & Data Structure Operations During Session Connection

7. Application Layer - Provides Protocols & Services For a Particular User Design/Application Process.

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Network Security Requirements

The Primary Security Services That Will Be Offered in Open System Interconnection Networks Are Authentication, Access Control, Confidentiality, Integrity and Nonrepudiation. These Are Defined in Detail in IS 7498/2 and Are Summarized with the Examples Given Below:

- Data Confidentiality Protects Data Against Unauthorized Disclosure. (Protecting the Details of an Attempted Corporate Takeover Is an Example of the Need for Confidentiality.)
- Data Integrity Protects Against Unauthorized Modification, Insertion and Deletion. (Electronic Funds Transfer between Banks Requires Protection Against Modification of the Information.)
- Authentication Verifies the Identity of Communicating Peer Entities and the Source of Data. (Owners of Bank Accounts Require Assurance That Money Will Only Be Withdrawn by the Owner.)
- Access Control Allows Only Authorized Communication and System Access. (Only Financial Officers Are Authorized Access to a Company's) Financial Plans.

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OSI Security Implementation

- Non-repudiation with Proof of Origin Provides to the Recipient Proof of the Origin of Data and Protects Against Any Attempt by the Originator to Falsely Deny Sending the Data or its Contents.
- Nonrepudiation with Proof of Origin can Be Used in a Court of Law as Proof to a Judge That a Person Signed a Contract.
- Requirements Have Been Identified for Government Applications for All Five of These Services, Especially the First Four. Authentication, Confidentiality and Integrity Are Implemented in Layers 2, 3, 4, and 7 of the OSI Architecture While Access Control and Nonrepudiation are Services Offered Only at Layer 7.
- Applications That Require Security at Layer 7, Such as Electronic Message Transfer and File Transfer, Can Be Provided All Security Services. Providing Security at One of the Layers 2, 3, and 4 Is Generally Required but Not at All Layers. Which Layer to Pick Depends on the Benefits and the Costs Encountered.

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NASA Communications II Security

Access Control for CDOS / NASA Communications II

<u>Identification</u>

1. Source address checked to see if packet should be allowed to use Nascom Network

<u>Authentication</u>

2. Encrypted password in the packet (or a digital signature) is checked to verify source user.

<u>Authorization</u>

3. Destination address is checked to verify that this source is allowed to transmit to desired destination. Destination's security requirements are checked. Integrity e.g., encipher date with appropriate key confidentiality, determine if source routing is necessary.

<u>Availability</u>

4. Audit this data transmission/Monitor for security events.
5. Packet is transmitted to Destination.
6. Destination uses its (public/secret) key to decode data.
7. Destination consults its packet's source address to see if this packet came from a valid source. Packet's encrypted password (or digital signature) used to verify source. [Incoming security services may also be necessary.]
8. Destination audits data arrival/monitor for security events.

NASA Communications Networks

- NASA has Extremely Good Communications Connective Networks.
- Many of these Networks have Dedicated Purposes

ARPA	Advanced Research Projects Agency
BITNET	"Because It's Time" Network for Electronic Mail (RSCS Protocol)
CSNET	Computer Science Network (TCP/IP Protocol)
GSFCMAIL	Goddard Space Flight Center Electronic Mail Service (X.400 Protocol)
HEPNET	High Energy Physics Network (DECnet Protocol)
INTERNET	Interoperable Set of Hundreds of TCP/IP Regional and National Networks
NASAMAIL	NASA Electronic Mail Service (X.400 Protocol)
NREN	National Research and Education Network
NSFNET	National Science Foundation Network (TCP/IP Protocol)
NSI	NASA Science Internet
NSN	NASA Science Network (TCP/IP)Protocol
OMNET	Commercial Electronic Mail and Related Services (X.25 Protocol)
PSCN	Program Support CommunicatioOns Network
SPAN	Space Physics Analysis Network (DECCnet Protocol)
TELEMAIL.	Commercial Electronic Mail Service (X.25 Protocol)

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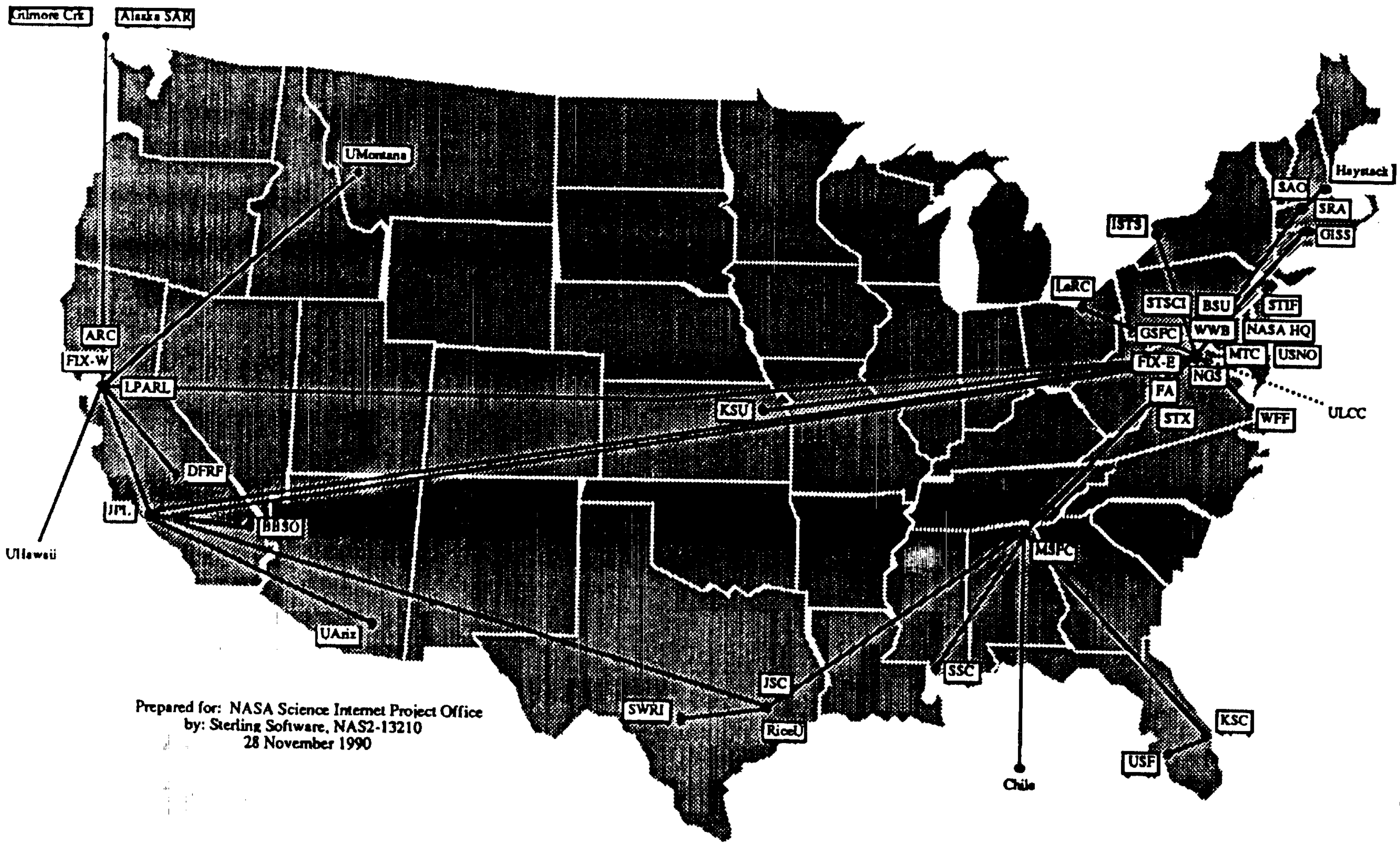
Example: NASA Science Network



Prepared for: NASA Science Internet Project Office
by: Sterling Software, NAS2-13210
28 November 1990

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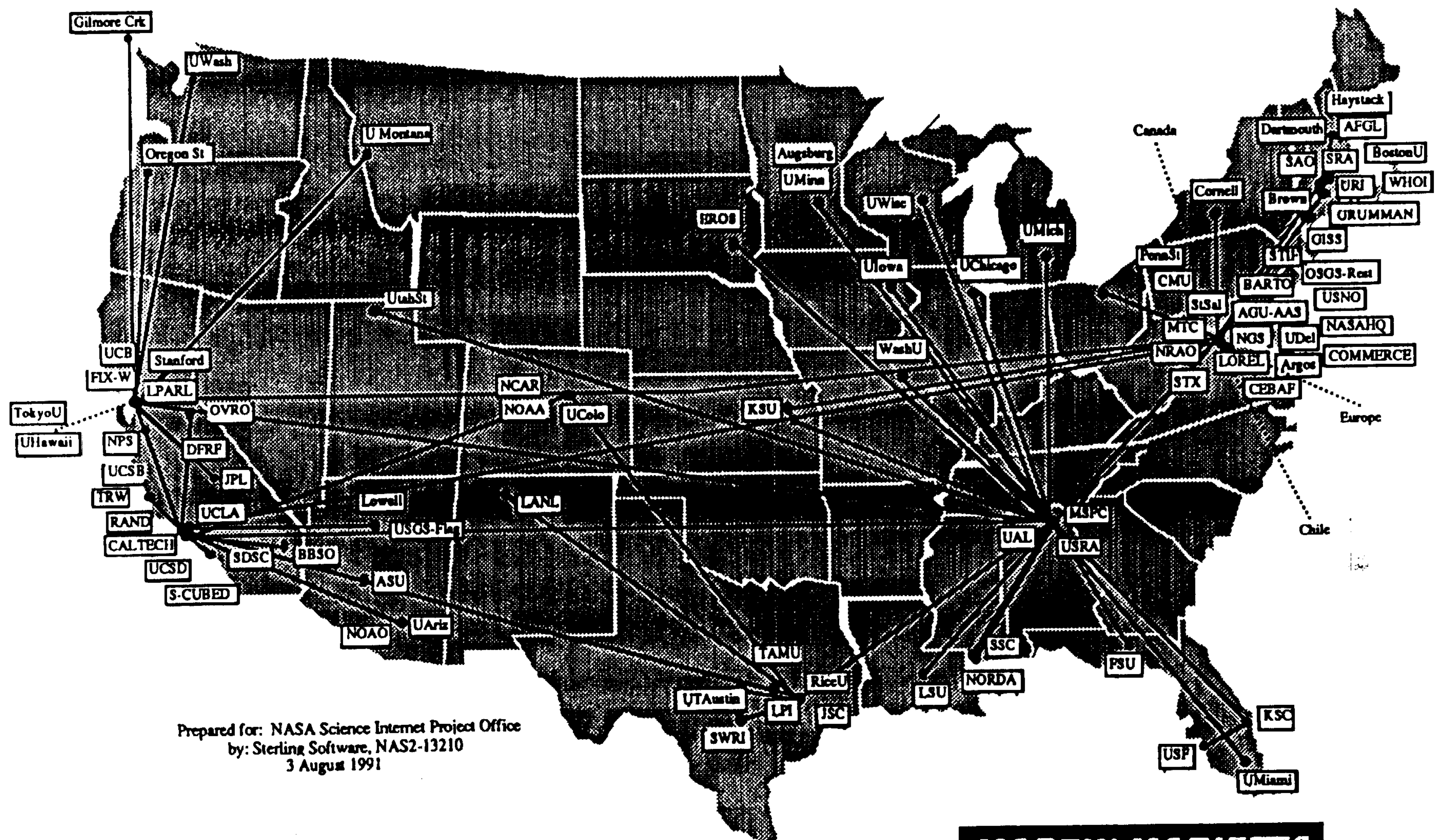
Example: NASA Science Internet



Prepared for: NASA Science Internet Project Office
by: Sterling Software, NAS2-13210
3 August 1991

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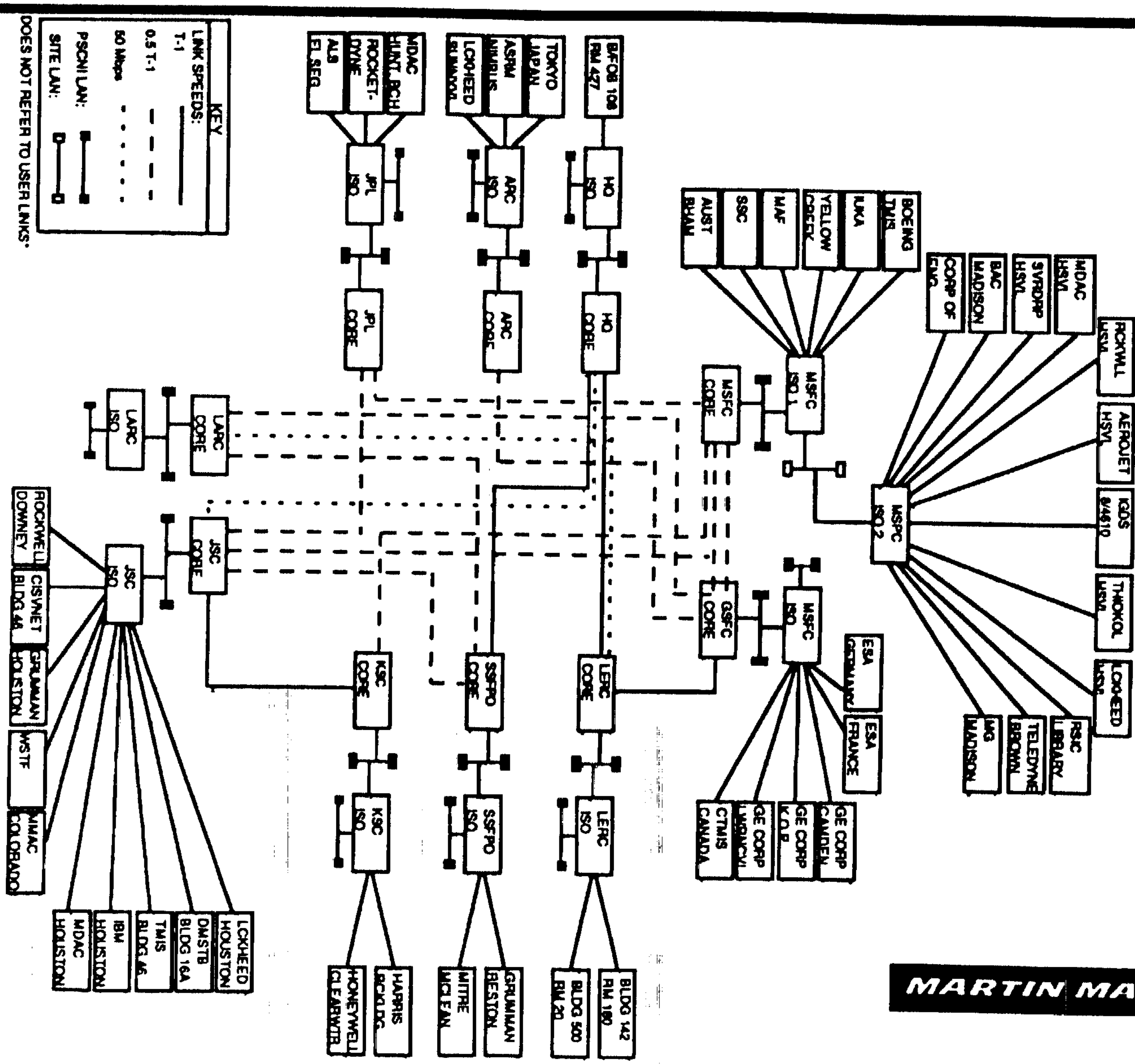
PSCNI Architecture



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PSCNI Architecture Capabilities

PSCNI is the Network Migration Path to Open System Interconnection (OSI). Capabilities Include:

- Advanced Research Project Agency (ARPA) Internet Protocols For Network Operation and Management
- Transmission Control Protocol/Internet Protocol (TCP/IP)
- Digital Equipment Corporation Network (DECnet)
- Xerox Network Systems (XNS)
- Appletalk
- International Standards Organization—Open Systems Interconnection (ISO-OSI)
- Novell IPX
- Ungermann - Bass XNS
- Evolving Family of Protocols to Incorporate New Systems

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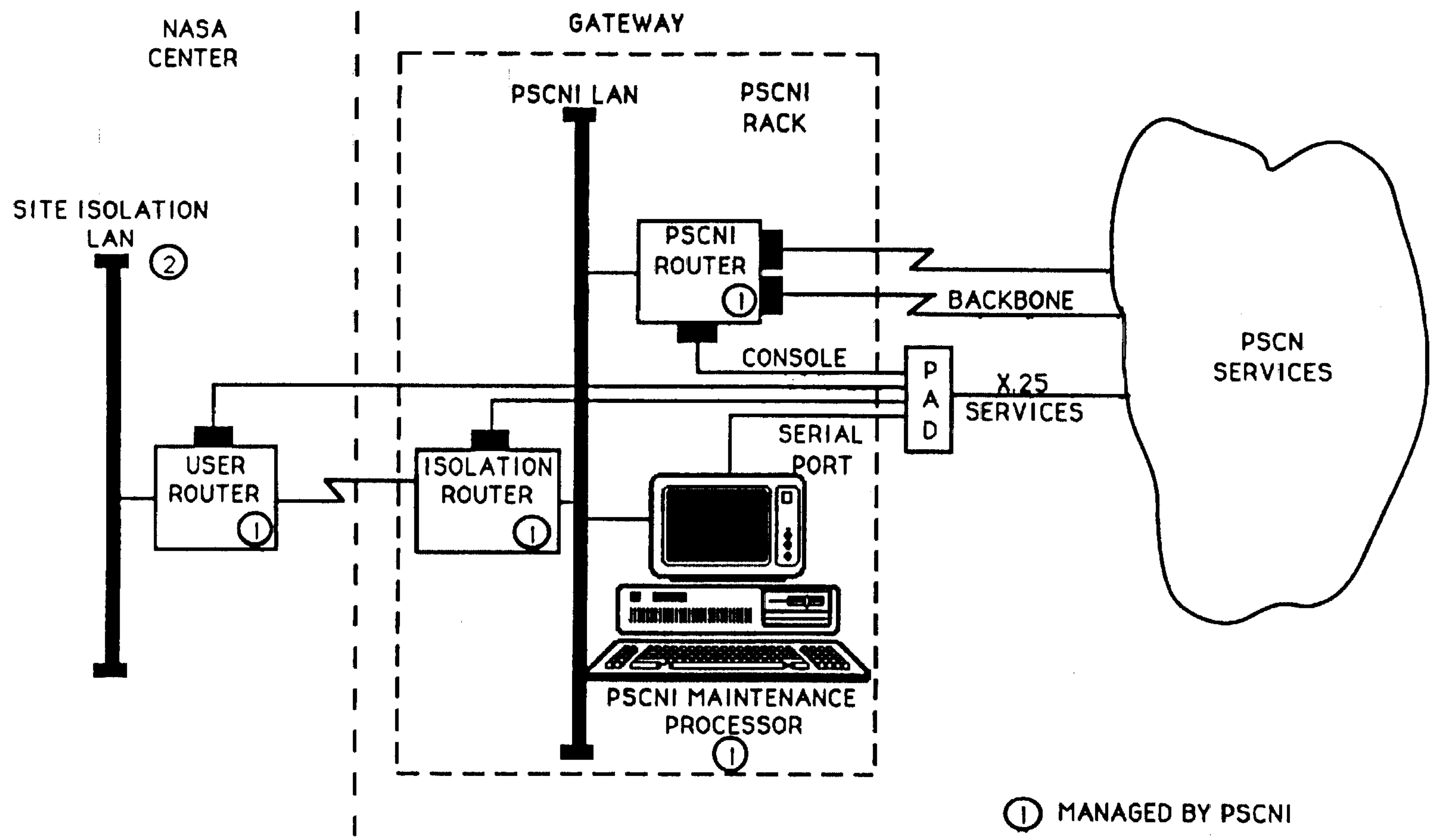
NASA Center Gateway



(1) MANAGED BY PSCNI

(2) USER OR SITE RESPONSIBILITY

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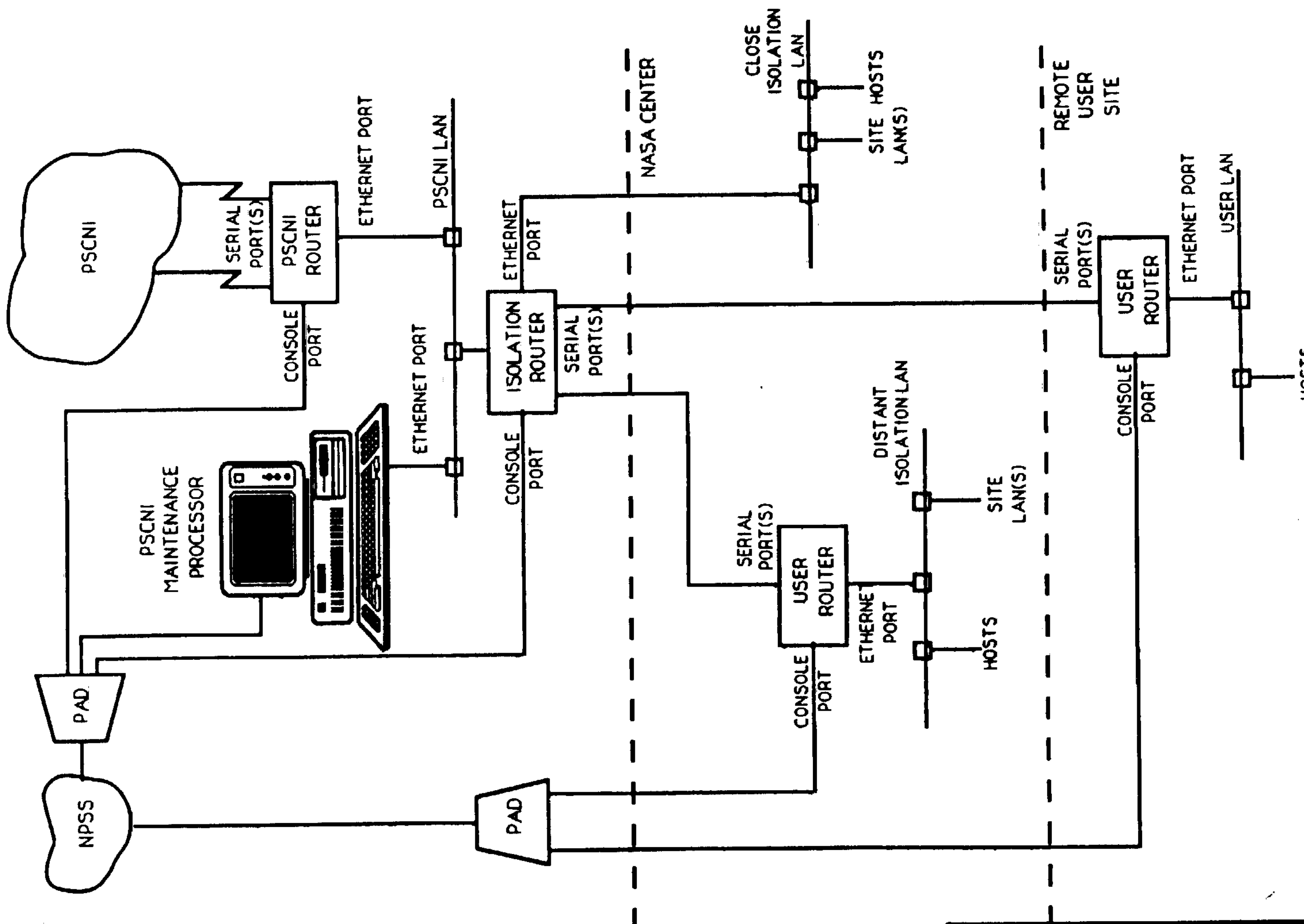
PSCNI Site Configuration (Non-Gateway)



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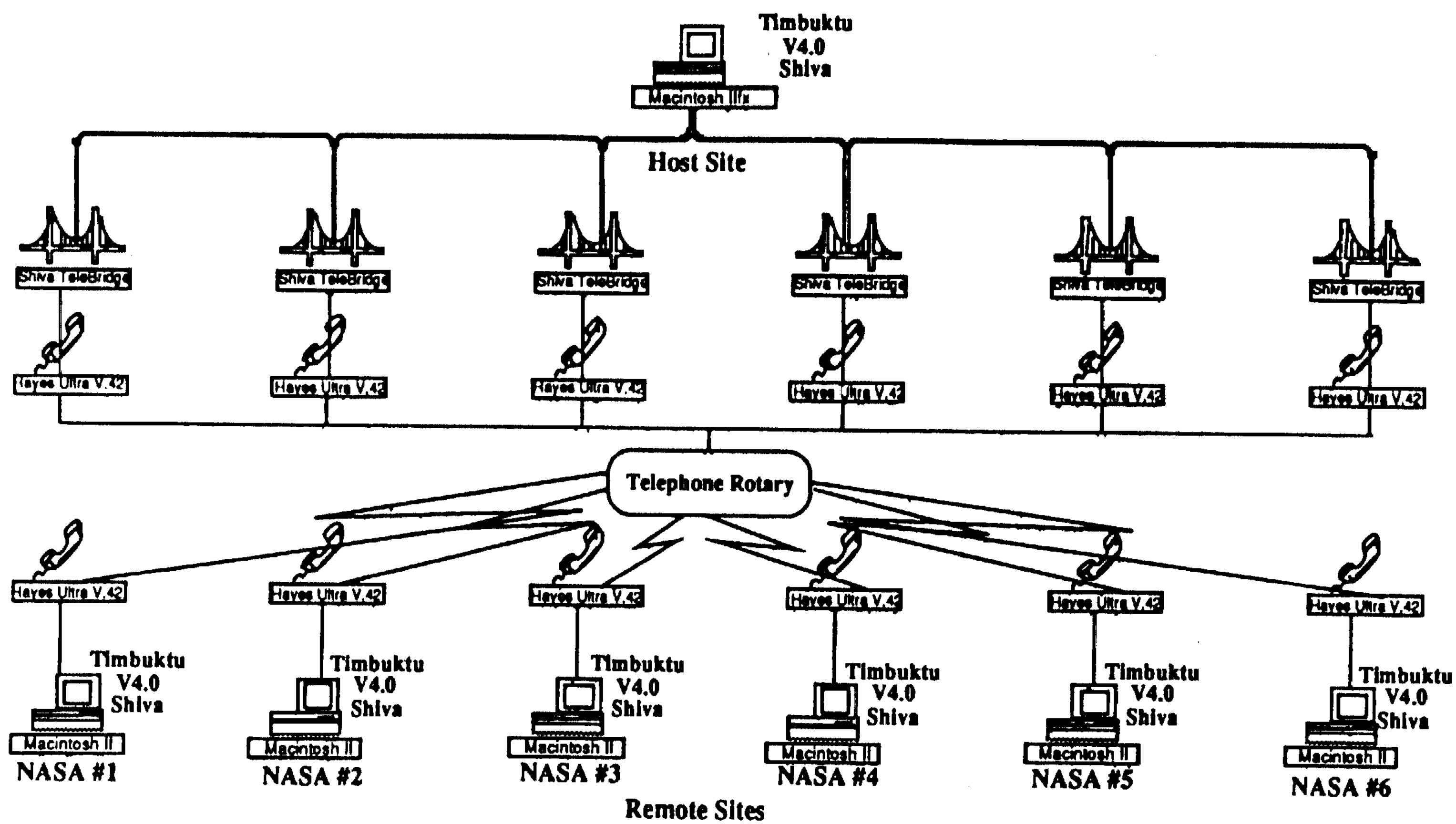
Example: Dedicated Network (SATWG QFD Network)



* Multiple Macs at a remote site would require a Telebridge at the remote site

SUMMARY

7 Mac II Personal Computers
6 Shiva TeleBridges
12 Hayes Ultra V.42 Modems
7 Timbuktu V4.0 Software

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Avionics Laboratory Systems and Determine Capabilities

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Determine Existing Laboratory HWD and SWF

- The Study Approach was:
  1) Contract All SATWG Members with a Survey Letter
  2) Follow Up with Personal Contacts and Phone Contacts
- Assimilate and Correlate Results
  1) Identify Each NASA Center Avionics Laboratory Capabilities
  2) Correlate Results with a Avionics Laboratory Concept

Laboratory Survey Contact Letter 10 May 91

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Jack Gallher,"NASA-KSC, Mail Code DL-DSD-23",Kennedy Space Center,FL,32899
Bill Wood,"NASA-KSC, Mail Code DL-PES",Kennedy Space Center,FL,32899
Ron Eatman,"NASA-KSC, Mail Code DF-FEP-22",Kennedy Space Center,FL,32899

J. F. Creedon,"NASA-LaRC, Mail Code MS ",Hampton,VA,23665
H. Milton Holt,"NASA-LaRC, Mail Code MS 469",Hampton,VA,23665
Wayne H. Bryant,"NASA-LaRC, Mail Code MS 478",Hampton,VA,23665
Floyd S. Shipman,"NASA-LaRC, Mail Code MS 478",Hampton,VA,23665
Harry F. Benz,"NASA-LaRC, Mail Code MS 473",Hampton,VA,23665

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Sherman Jobe,"NASA-MSFC, Mail Code EB",Marshall Space Flight Center,AL,35812

George Zupp,"NASA-JSC, Mail Code ET",Houston, TX,77058
Bill Teasdale,"NASA-JSC, Mail Code EE",Houston, TX,77058
Bob Hendrix,"NASA-JSC, Mail Code EP54",Houston, TX,77058
Aldo Bordano,"NASA-JSC, Mail Code EG",Houston, TX,77058
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Johnson (713)
Don Brown 483-8241
Dave Pruett 483-5269
Tom Jeffcoat
Greg Hite
Larry McWhorter 483-8306
Bob Hendrix 483-8283
George Zupp 483-6604
Kenneth Cox 483-8224

Kennedy (407)
Ron Eatman 867-2712
Jack Warwick 867-4976
Jack Galliher 867-3224
Bob Luken 867-7069

Langley (804)
Floyd Shipman 864-1706
Wayne Bryant 864-1692
H. Milton Holt 864-1596
J.F. Creeden 864-6033
Harry Benz 864-1496

Lewis (216)
Henry Brandhorst 433-6149

Marshall (205)
John Dumolin
Darlene McLaughlin
David Aichele 544-3722
Cynthia Frost 544-0628

Stennis (601)
Glade Woods 688-2777
Don Chenevert 688-3126
Gerald Meeks

Artificial Intelligence
Research (904)
Jackson Driskell 399-0321

Boeing
Bill Ramey 713-483-7544
J.P. McMillen 216- 297-0440

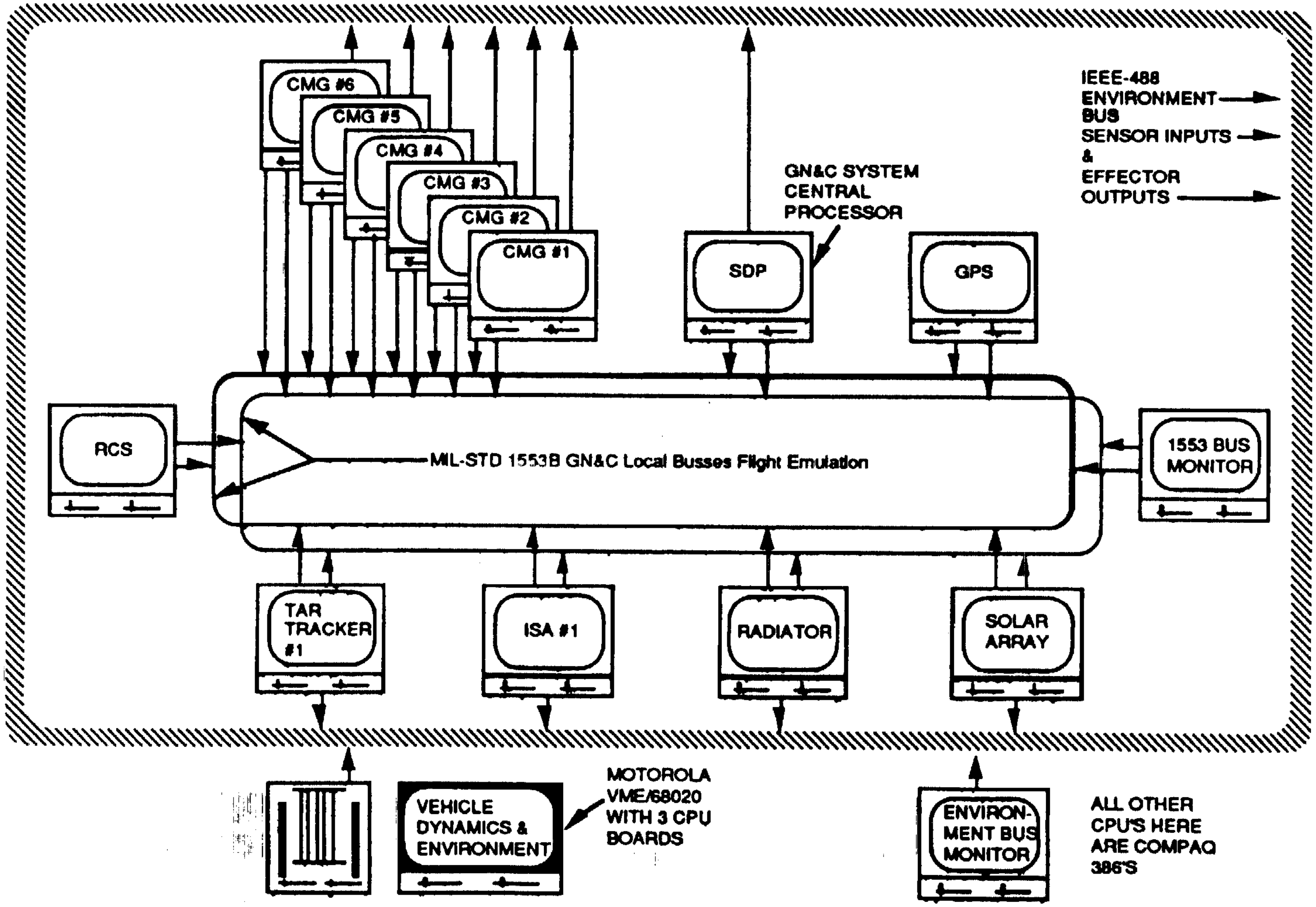
Martin Marietta (303)
Ron Grisell 977-1764
Nancy Ruzicka 971-7992
Ron Eicher 977-5053
David Scruggs 971-4804
Steve Driskell 971-7074
Rob Mason 977-6948
Steve Sorensen 971-6747
Ron Bena 977-5423
Glenda McFarlin 977-3208
Joe Keeley 977-3208
Stephen Johnson 977-1449
Ted Ackerlund 977-1085

NASA Avionics Technology Lab's

The Capabilities of NASA Avionics Laboratories are Closely Attuned to the Technology and Programmatic Charters of the Each Center.

- NASA Centers with the Generic Avionics Laboratory Capabilities
  - MSFC (Launch Vehicle, Docking, Dynamics, Software)
  - JSC (Spacecraft, Rendezvous, Communications, Software)
  - ARC (Controls & Displays, Processors, Software)
  - LaRC (GN&C, Information Processing Technology, system Validation Methodology)
  - GSFC (Unmanned Spacecraft, Sensors, Software)
- NASA Centers with Specialized Avionics Laboratory Capabilities
  - LeRC (Communications and Instrumentation)
  - SSC (Engine Test and Instrumentation)
  - KSC (Instrumentation, Software)

Example: JSC GN&C Test BED



- Micro Vax
- VME
- PCs

From: Frank M. Elam (EG4)

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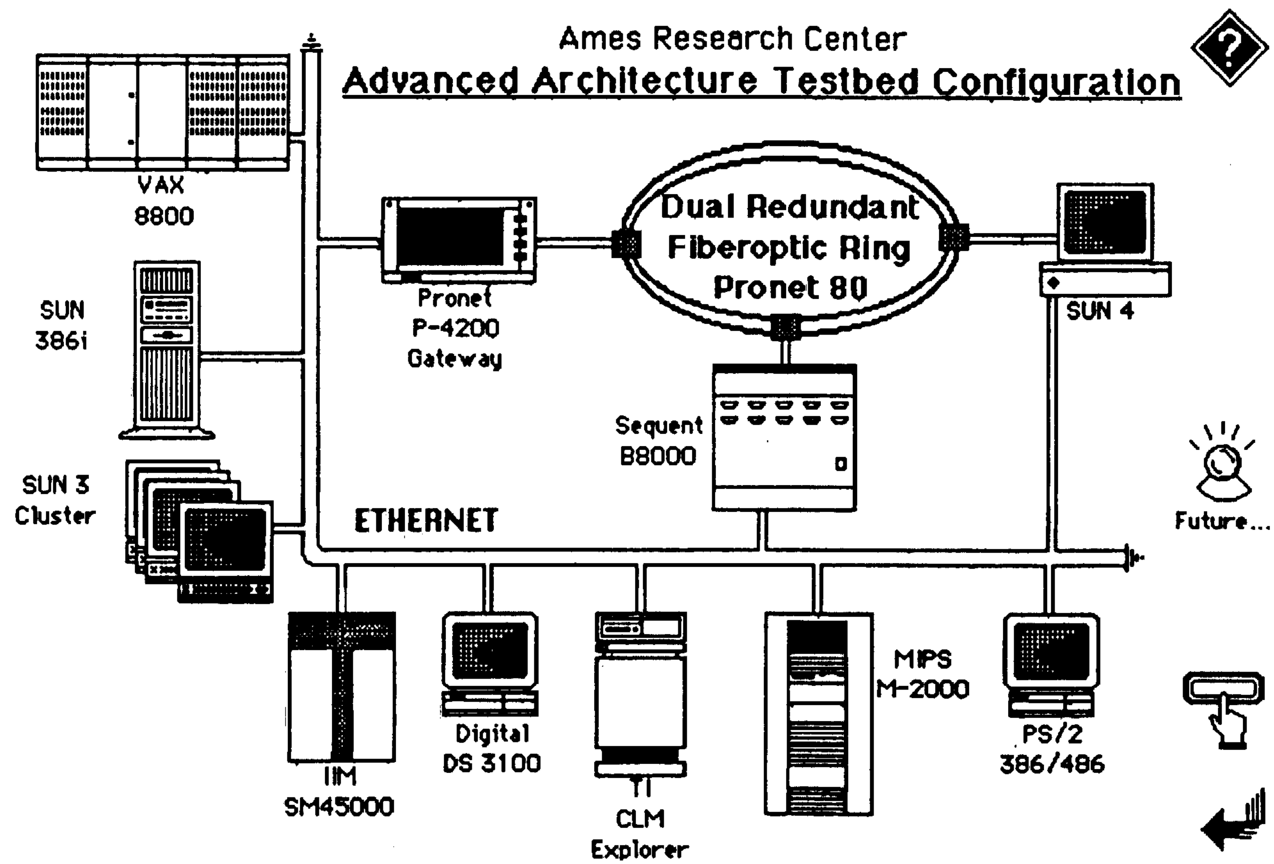
DS112491-6C

Example: ARC DMS Testbed



Summarize Laboratory Survey Results

- NASA Avionics Technologies Laboratories
  - Low to Medium Hardware Content
    - PCs, Macintosh, Vax, Symbolics and Sun's
    - Low to Medium Processing Capabilities
  - Very Good Software Content
    - Languages
      - Fortran, Pascal, ADA, Lisp, C and Others
    - Applications
      - CAD, CA, ADA, ELI, Graphics, and Others

- Connectivity Assessment

  ARC - High
  GSFC - High
  JSC - Medium
  KSC - Low
  LaRC - High
  LeRC - Low
  MSFC - High
  SSC - Low

Integrated Avionics Testbed Concept

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Distributed Avionics Testbed Concept Development

- The Connectivity Study Considered Two Future Programs for Concept Evaluation.

  1) Space Transfer Vehicle

  2) the 90 SEI Lunar Systems Architecture.

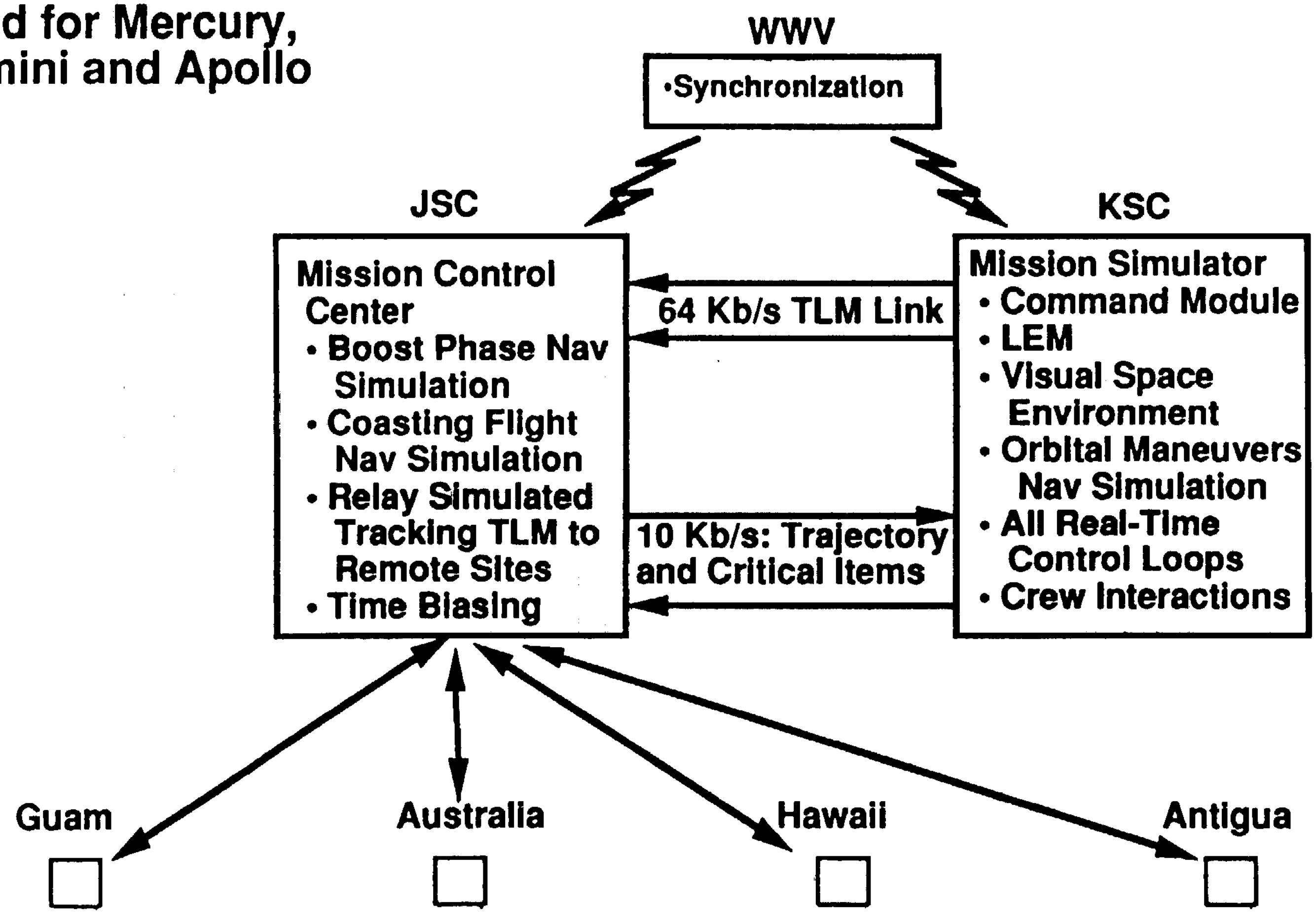
- The Analysis was Limited to Currently Available Hardware and Software which could Support Distributed Avionics Laboratory Needs.

- Previous NASA Programs which Utilized Distributed Simulations for Crew and Flight Controller Training were Analyzed for Applicability to a Distributed Avionics Test Bed Concept.

- The Top Level Constraints, Issues and Requirements are Outlined in Summary Form.

Example: Distributed Simulation System

- Simulation Architecture Used for Mercury, Gemini and Apollo



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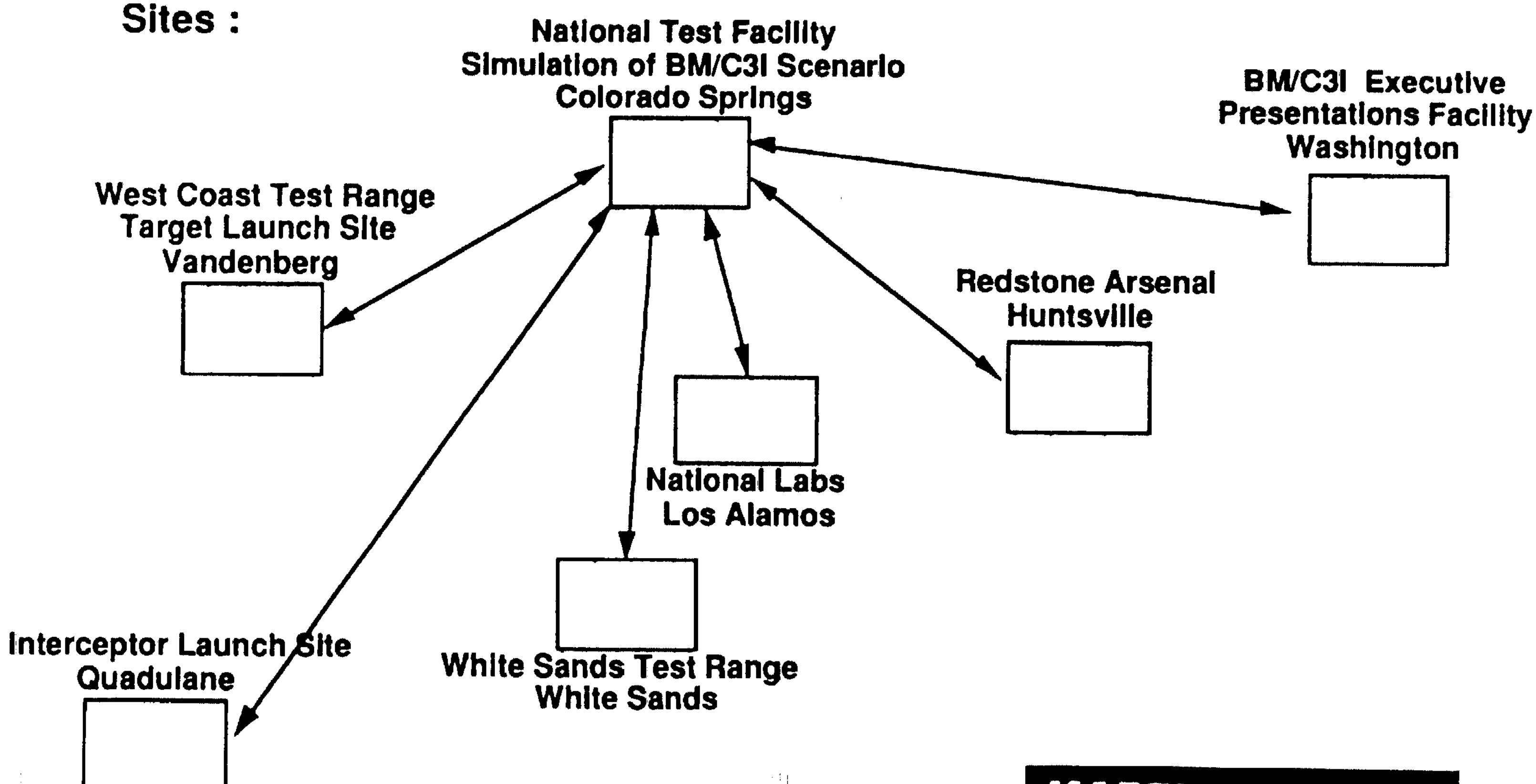
Example: National Testbed Connectivity Diagram

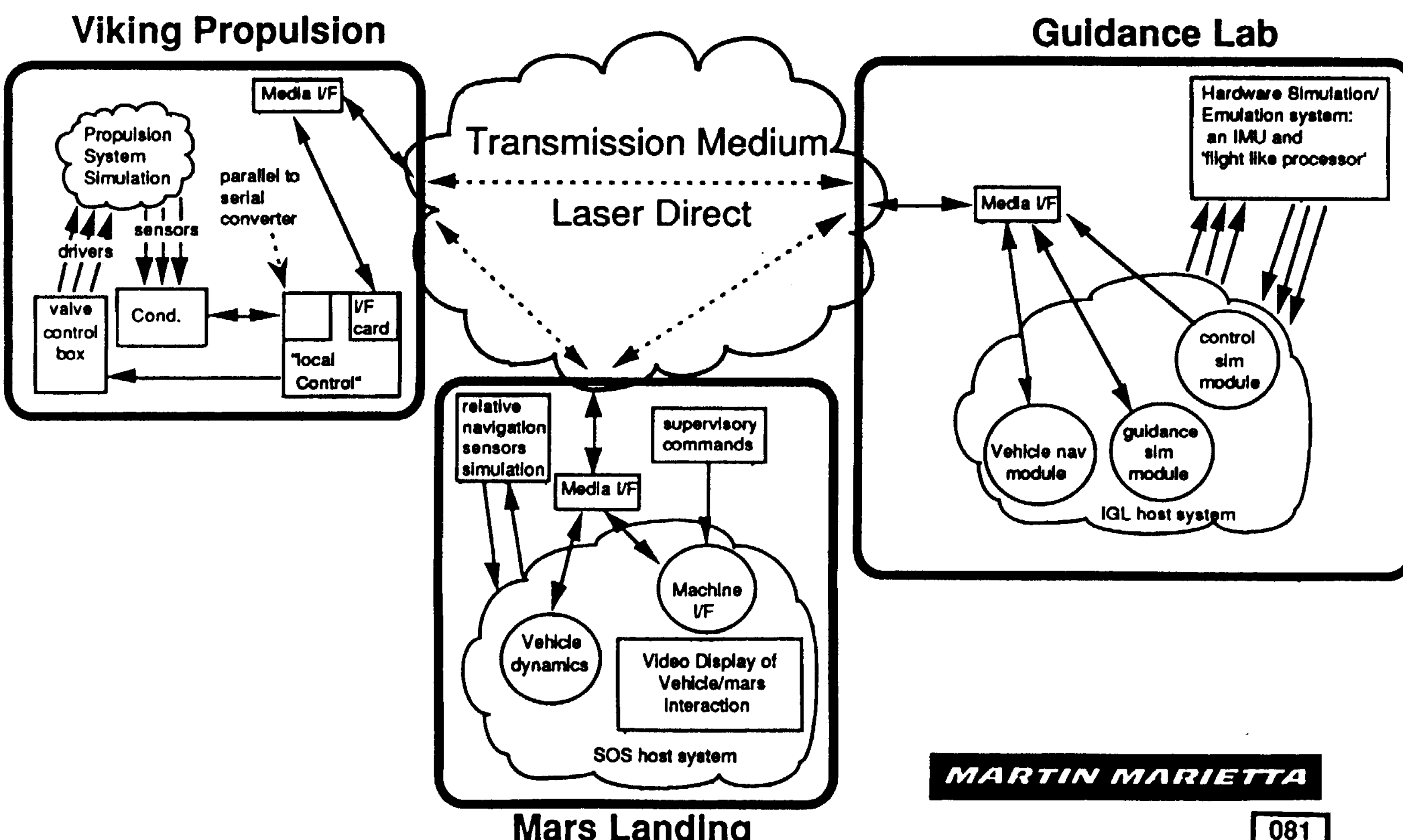
- Initial Architecture for SDIOs Systems Verification and Validation.
- Key Elements were Real Time Integration of Distributed Elements.

Sites :

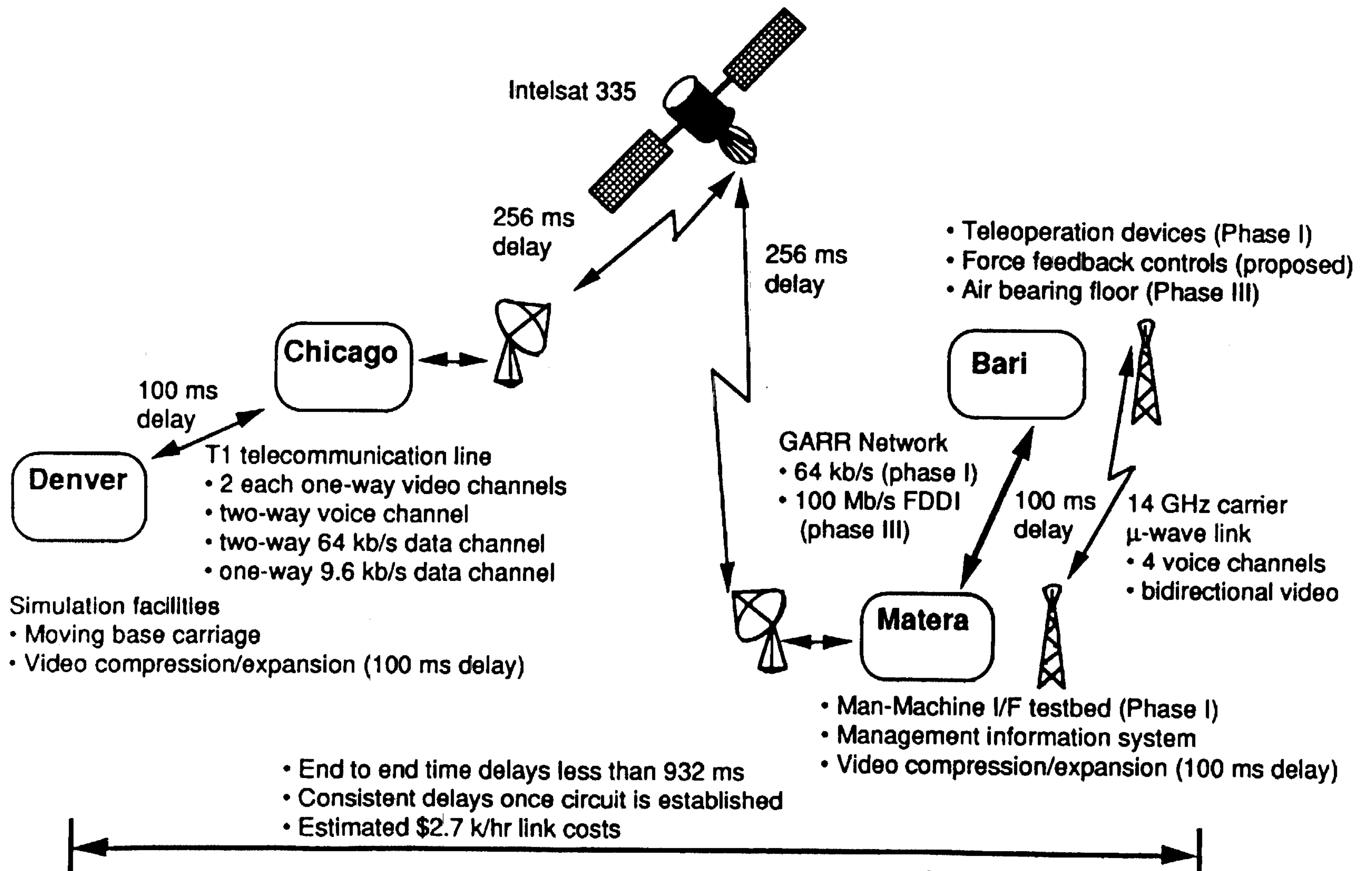


Example: Viking Development and Validation Laboratory

- Localized Distributed Simulation



Example: Planned International Telesimulation Testbed



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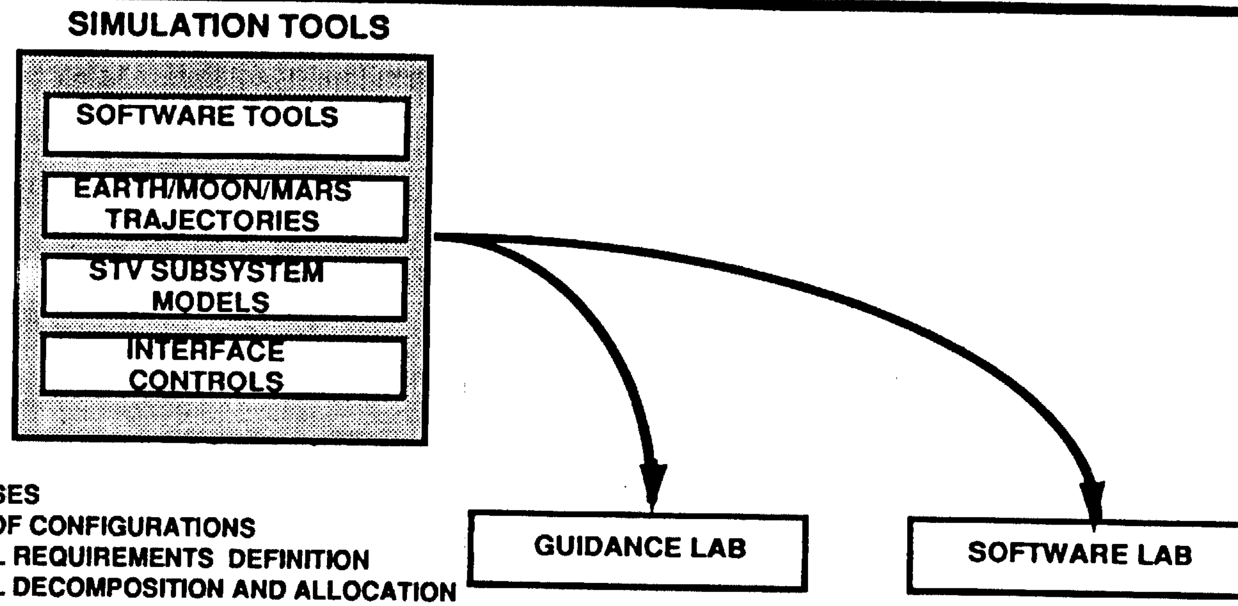
Integrated Avionics Testbed

The Concept for an Integrated Avionics Laboratory should Address the Stages of R&D Development for a New NASA Flight System.

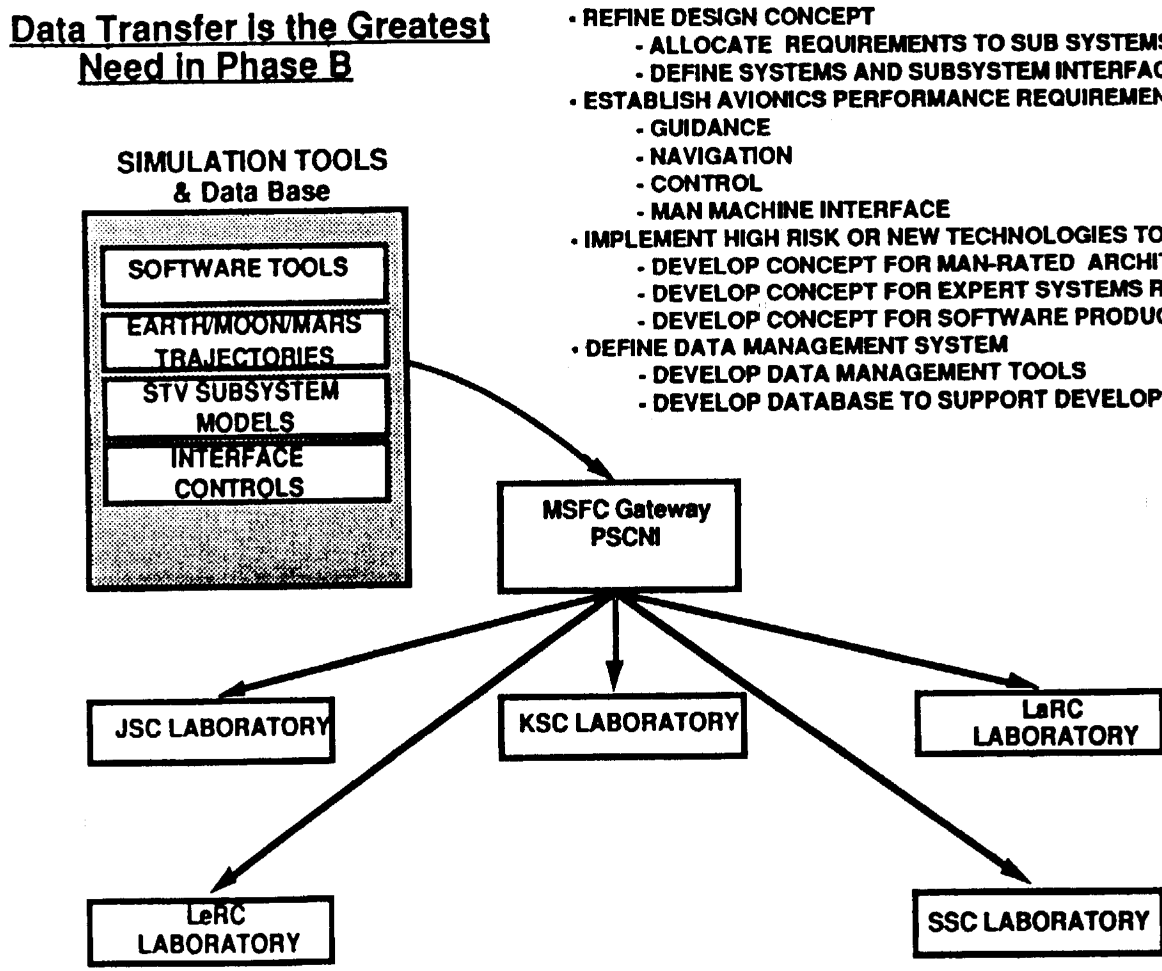
1. The ability to Evaluate Concepts and Technologies Employed in the Design of Space Systems Through the Extensive use of Software Tools.
2. The Ability to Conduct Rapid Prototyping (Hardware and Software) of Concepts for Evaluation.
3. The Ability to Conduct Sub-system Simulations to Evaluate Component Performance.
4. The Ability to Conduct End-to-End Simulations Containing a Mixture of Simulated, Emulated and Prototype Avionics Systems.
5. The Ability to Conduct Integrated Hardware-in-the-Loop Simulations for the Purpose of Validation and Verification.
6. The Ability to Conduct Real-Time Mission Monitoring, Analysis and Mission Support.

PHASE A: Concept Development (STV Type Program)



- SYSTEMS ANALYSES
  - ANALYSES OF CONFIGURATIONS
  - FUNCTIONAL REQUIREMENTS DEFINITION
  - FUNCTIONAL DECOMPOSITION AND ALLOCATION
  - AVIONICS SUBSYSTEM REQUIREMENTS DEFINITION
- MODELING AND SIMULATION
  - DEFINE METHODOLOGY AND TOOLS TO ENHANCE DESIGN PHASE
  - DEVELOP MODELS FOR: VEHICLE CONCEPTS, PRODUCTION OPERATIONS, FLIGHT OPERATIONS, AND SYSTEMS COSTING.
  - DEVELOP SIMULATIONS TO SUPPORT CONCEPT DEVELOPMENT
- CONCEPT DEVELOPMENT
  - CONDUCT AVIONIC SYSTEMS TRADE STUDIES
  - DEFINE FUNCTIONAL AVIONICS ARCHITECTURE
  - DEFINE CONCEPTUAL AVIONICS DESIGN INCLUDING HARDWARE, SOFTWARE, OPERATIONS, AND SUBSYSTEMS.

PHASE B: Design Definition (STV Type Program)



- REFINE DESIGN CONCEPT
  - ALLOCATE REQUIREMENTS TO SUB SYSTEMS
  - DEFINE SYSTEMS AND SUBSYSTEM INTERFACES
- ESTABLISH AVIONICS PERFORMANCE REQUIREMENTS
  - GUIDANCE
  - NAVIGATION
  - CONTROL
  - MAN MACHINE INTERFACE
- IMPLEMENT HIGH RISK OR NEW TECHNOLOGIES TO SUPPORT DESIGN CONCEPT
  - DEVELOP CONCEPT FOR MAN-RATED ARCHITECTURES
  - DEVELOP CONCEPT FOR EXPERT SYSTEMS RDT&E ARCHITECTURE
  - DEVELOP CONCEPT FOR SOFTWARE PRODUCTION AND MAINTENANCE
- DEFINE DATA MANAGEMENT SYSTEM
  - DEVELOP DATA MANAGEMENT TOOLS
  - DEVELOP DATABASE TO SUPPORT DEVELOPMENT

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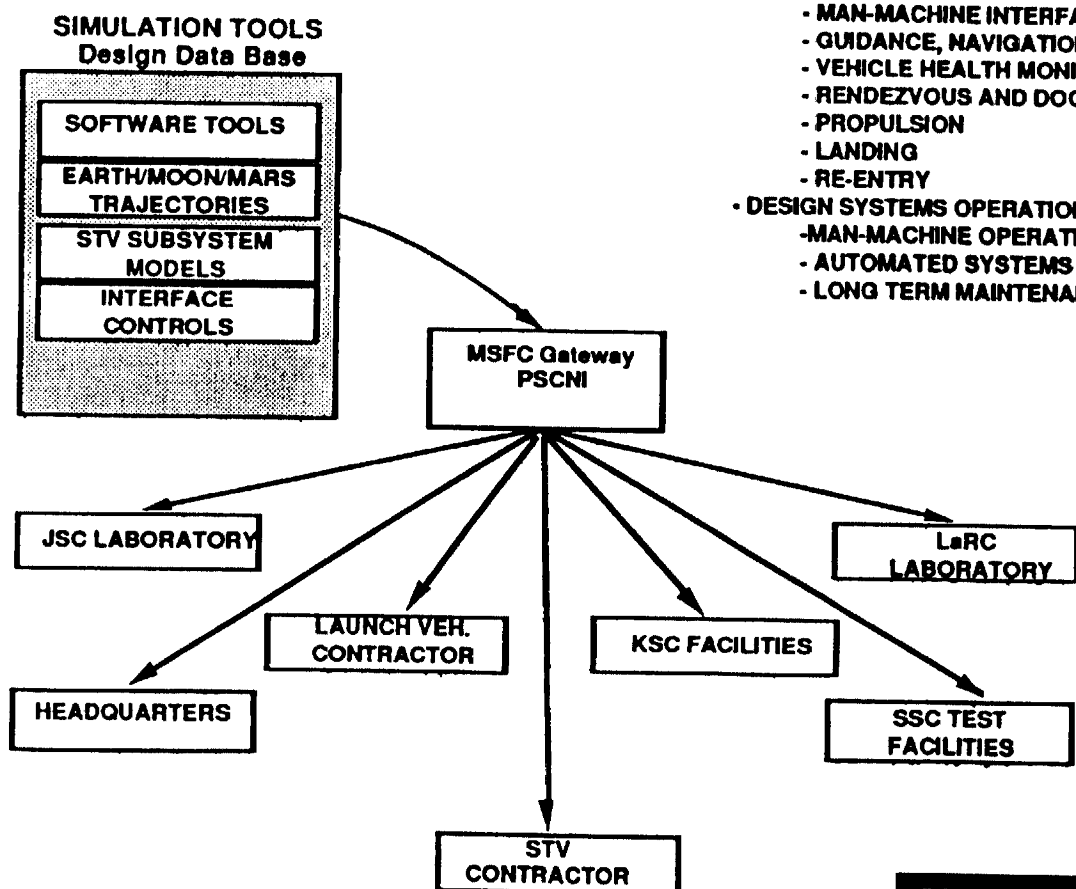
PHASE C: Systems Design (STV Type Program)

Design Phase Requires Extensive Interfaces to Multiple Facilities and Laboratories



- DEFINE VEHICLE SYSTEM
  - ALLOCATE SYSTEMS REQUIREMENTS TO COMPONENTS
  - ALLOCATE OPERATIONS REQUIREMENTS TO COMPONENTS
- DESIGN COMPONENTS
  - MAN-MACHINE INTERFACE
  - GUIDANCE, NAVIGATION, AND CONTROL
  - VEHICLE HEALTH MONITORING
  - RENDEZVOUS AND DOCKING
  - PROPULSION
  - LANDING
  - RE-ENTRY
- DESIGN SYSTEMS OPERATIONS
  - MAN-MACHINE OPERATIONS
  - AUTOMATED SYSTEMS OPERATIONS
  - LONG TERM MAINTENANCE, TEST, AND CHECKOUT

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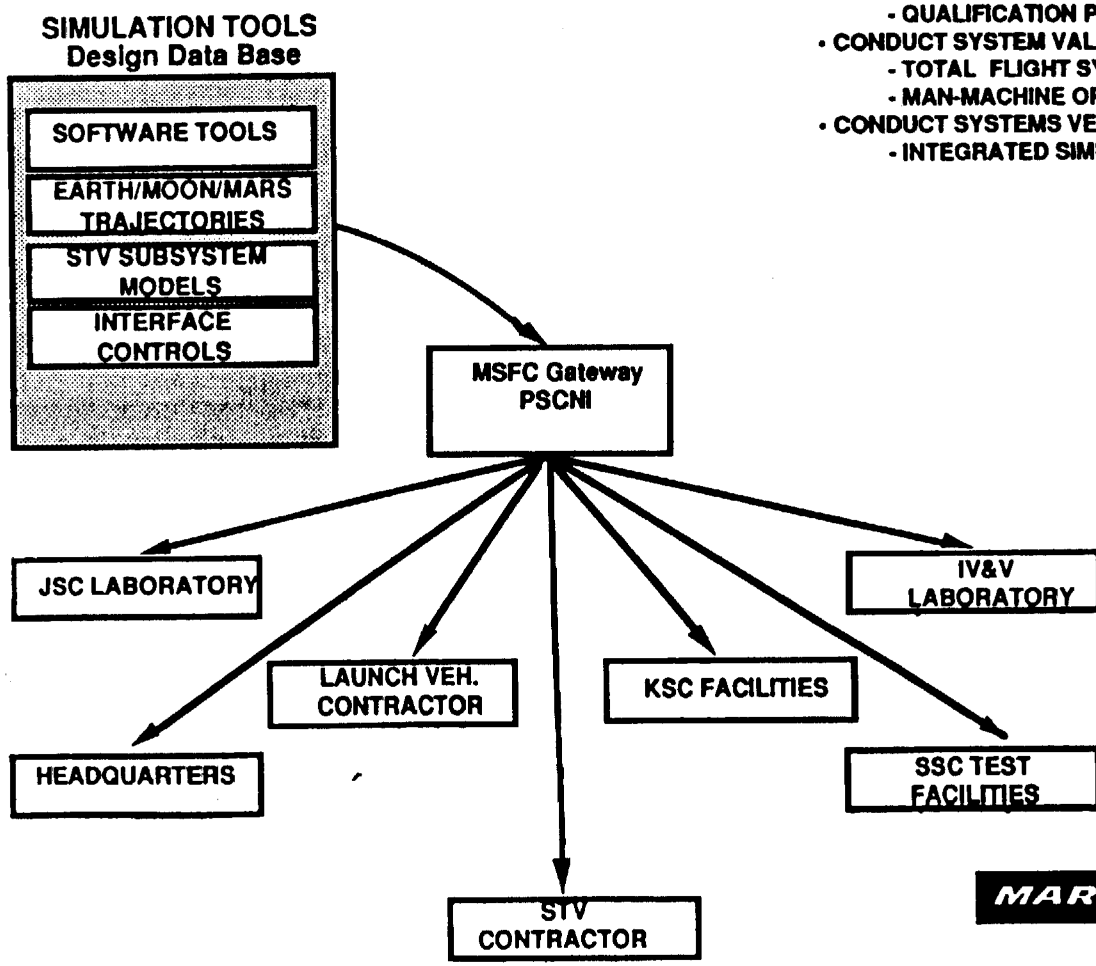
PHASE D: Validation and Verification (STV Type Program)

Similar to Phase C with the Addition of IV&V Laboratories

- DEFINE VERIFICATION AND VALIDATION METHODOLOGY
  - HARDWARE AND SOFTWARE TOOLS
  - VERIFICATION AND VALIDATION PLAN
- CONDUCT SUBSYSTEM QUALIFICATION
  - HARDWARE AND SOFTWARE TOOLS
  - QUALIFICATION PLAN
- CONDUCT SYSTEM VALIDATION
  - TOTAL FLIGHT SYSTEM
  - MAN-MACHINE OPERATIONS
- CONDUCT SYSTEMS VERIFICATION
  - INTEGRATED SIMULATIONS



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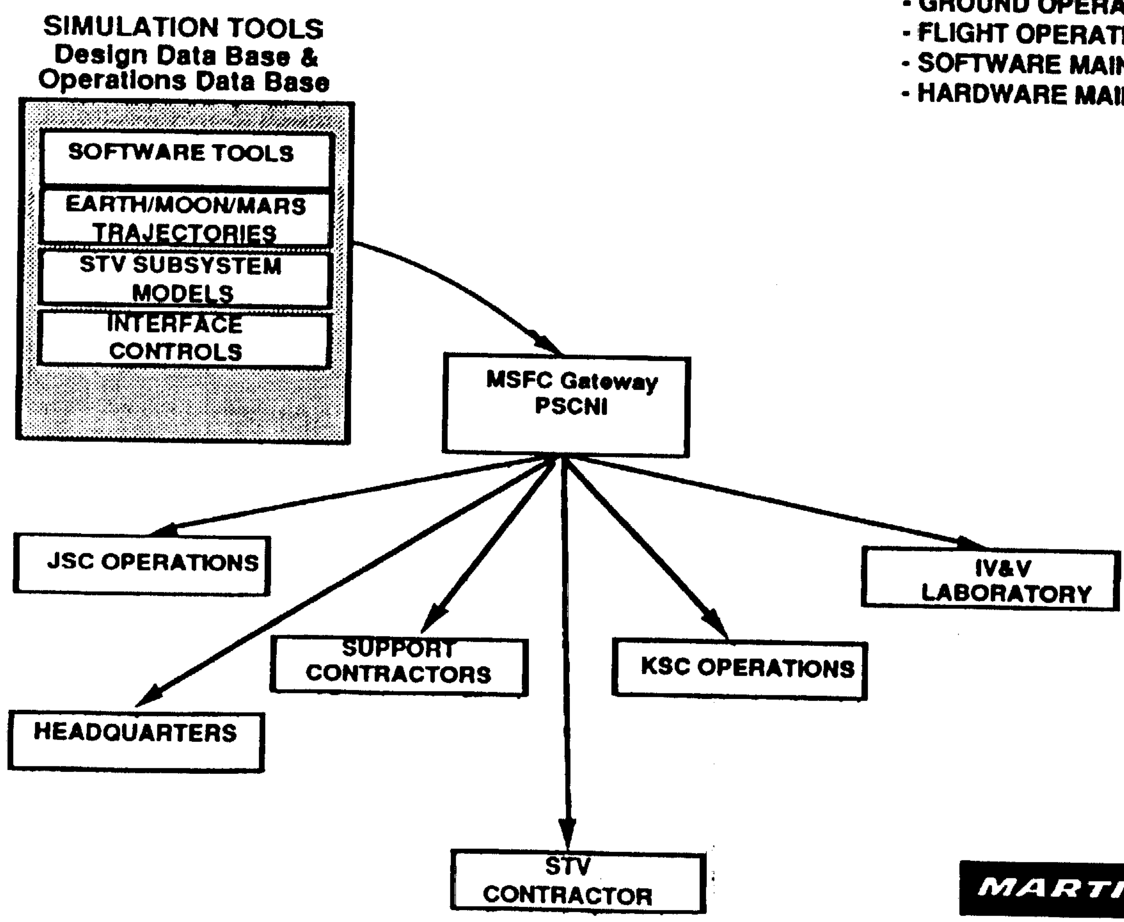
PHASE E/F: Production and Mission Operations (STV Type Program)

Operations are Stressed in This Phase - Laboratories are Still Required to Support Day-to-day Operations

- OPERATIONS AND PRODUCTION PHASE
  - TRAINING
  - GROUND OPERATIONS SUPPORT
  - FLIGHT OPERATIONS SUPPORT
  - SOFTWARE MAINTENANCE
  - HARDWARE MAINTENANCE



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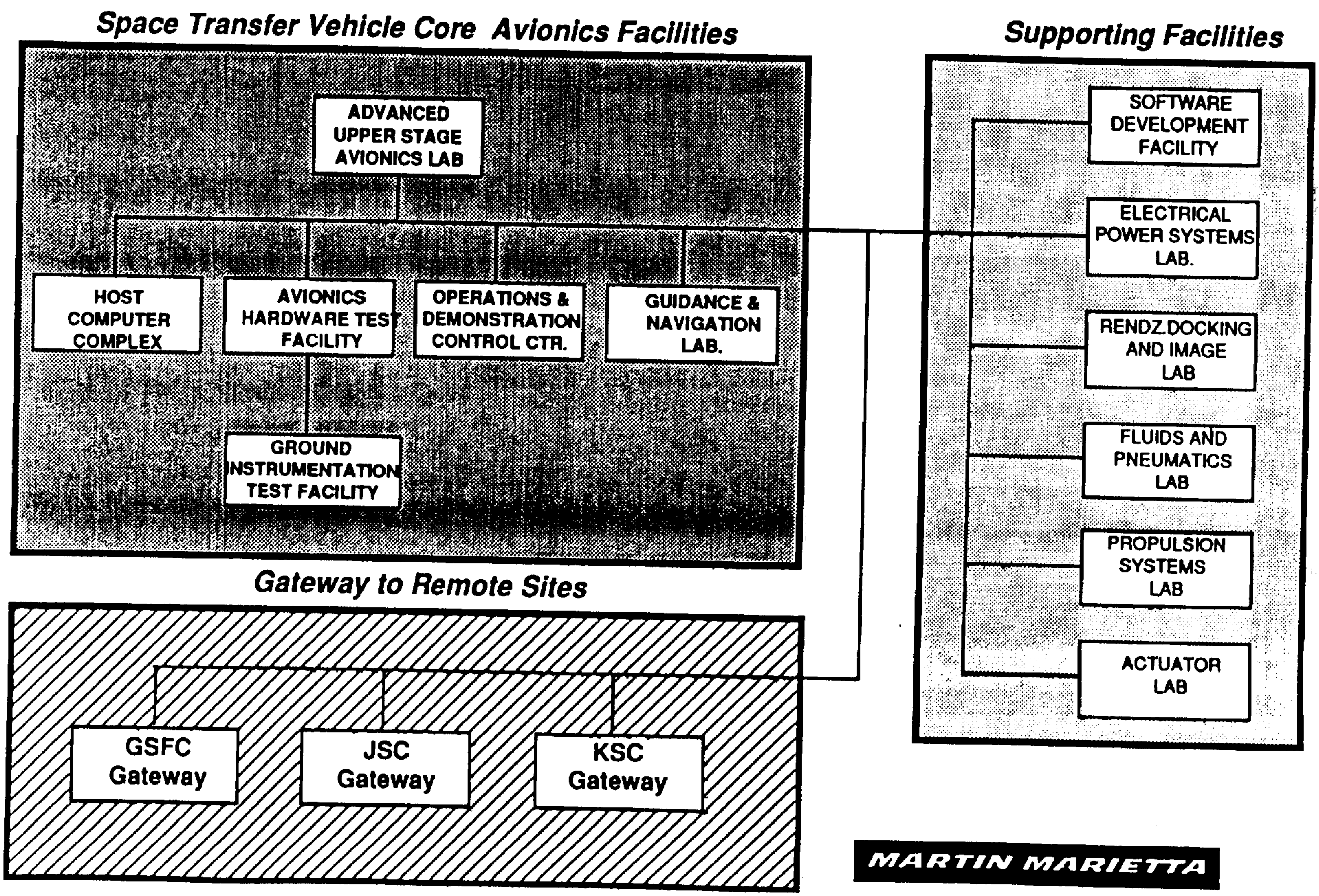
STV Advanced Avionics Test Bed Concept



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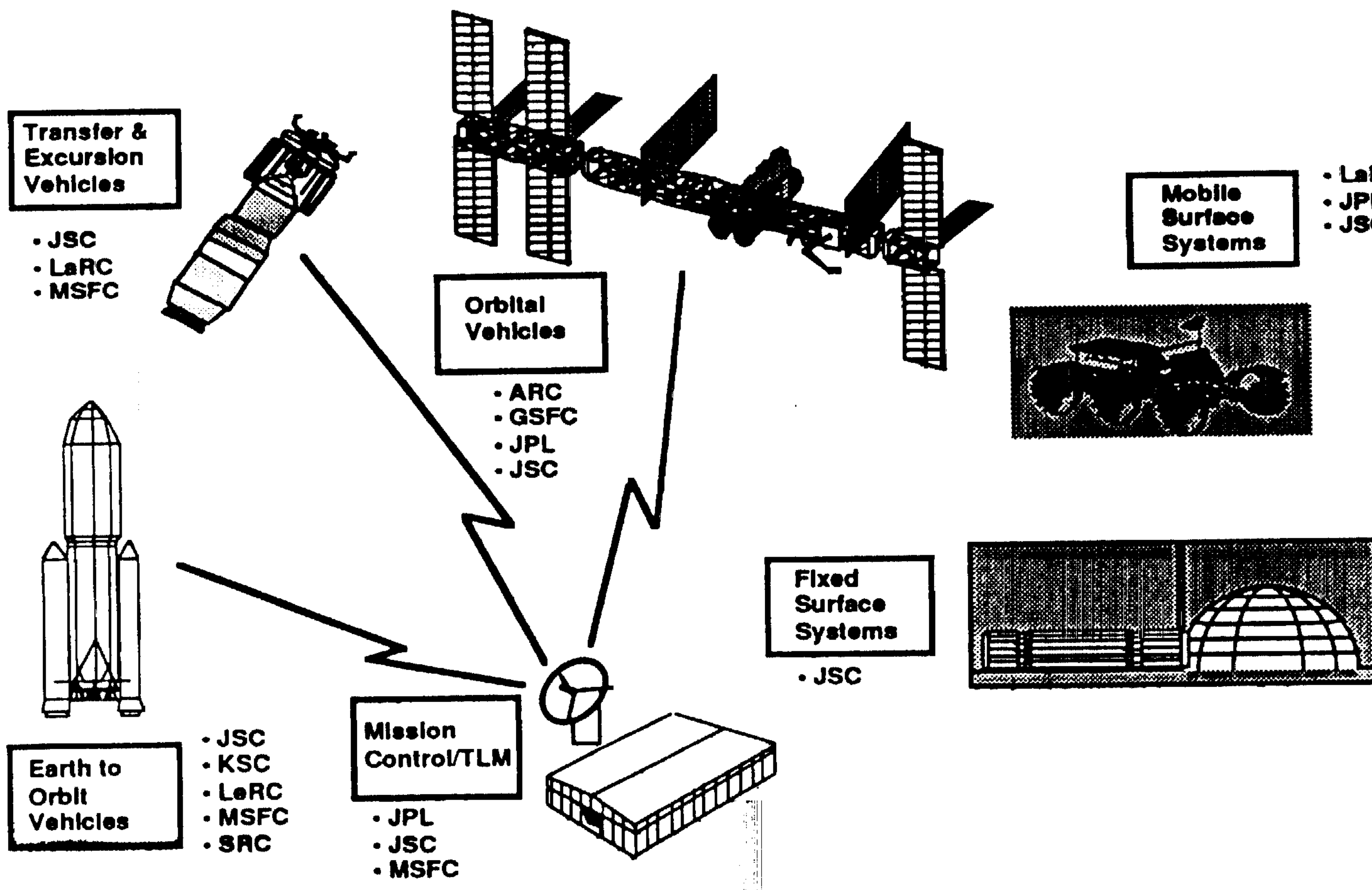
Space Exploration Initiative Concept



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Connectivity Arch. for SEI Distributed Avionics Lab.

SPACE INFRASTRUCTURE

EXAMPLE NASA SIMULATION COVERAGE

FUNCTIONS: / CENTERS:	EARTH TO ORBIT								
	ARC	GSFC	JPL	JSC	KSC	LaRC	LeRC	MSFC	SRC
GN&C				X		X		X	
ATTITUDE CONTROL				X		X		X	
TELEMETRY				X	X		X	X	
INSTRUMENTATION (SENSORS)				X	X		X	X	X
VEHICLE HEALTH MONITORING				X	X		X	X	X
PROPULSION CONTROL				X	X		X	X	X
RANGE SAFETY / DESTRUCT				X	X			X	
MISC PYRO CONTROL				X	X			X	
HUMAN I/F (CONTROL / DISPLAY)				X	X			X	
EFFECTOR CONTROL					X		X		X
EMERGENCY SYSTEMS				X	X			X	
GENERAL DATA PROCESSING				X	X			X	
MASS DATA STORAGE				X	X			X	
EXPERIMENT CONTROL / MONITOR				X		X		X	
NON-TLM DATA COMMUNICATION		X		X	X		X	X	
AUDIO / VIDEO COMMUNICATION		X		X	X		X	X	
MEDICAL / DIAGNOSTIC SYSTEMS				X					
POWER SOURCE / CONVERSION							X	X	
POWER MANAGEMENT							X	X	
REACTOR CONTROL							X	X	

FUNCTIONS: / CENTERS:	TRANSFER/ EXCURSION								
	ARC	GSFC	JPL	JSC	KSC	LaRC	LeRC	MSFC	SRC
GN&C				X		X		X	
ATTITUDE CONTROL				X		X		X	
TELEMETRY				X			X	X	
INSTRUMENTATION (SENSORS)				X			X	X	X
VEHICLE HEALTH MONITORING				X			X	X	X
PROPULSION CONTROL				X			X	X	X
RANGE SAFETY / DESTRUCT									
MISC PYRO CONTROL				X				X	
HUMAN I/F (CONTROL / DISPLAY)				X				X	
EFFECTOR CONTROL							X		X
EMERGENCY SYSTEMS				X				X	
GENERAL DATA PROCESSING		X		X				X	
MASS DATA STORAGE				X				X	
EXPERIMENT CONTROL / MONITOR				X		X		X	
NON-TLM DATA COMMUNICATION		X		X			X	X	
AUDIO / VIDEO COMMUNICATION		X		X			X	X	
MEDICAL / DIAGNOSTIC SYSTEMS				X					
POWER SOURCE / CONVERSION				X			X	X	
POWER MANAGEMENT				X			X	X	
REACTOR CONTROL							X	X	

FUNCTIONS: / CENTERS:	ORBITAL SSF								
	ARC	GSFC	JPL	JSC	KSC	LaRC	LeRC	MSFC	SRC
GN&C				X		X			
ATTITUDE CONTROL				X		X			
TELEMETRY		X		X		X	X		
INSTRUMENTATION (SENSORS)		X		X		X	X	X	
VEHICLE HEALTH MONITORING				X		X	X	X	
PROPULSION CONTROL				X		X			
RANGE SAFETY / DESTRUCT									
MISC PYRO CONTROL				X				X	
HUMAN I/F (CONTROL / DISPLAY)				X		X		X	
EFFECTOR CONTROL				X		X	X		X
EMERGENCY SYSTEMS				X			X		
GENERAL DATA PROCESSING		X		X			X	X	
MASS DATA STORAGE		X		X		X	X		
EXPERIMENT CONTROL / MONITOR	X			X		X	X		
NON-TLM DATA COMMUNICATION		X		X			X	X	
AUDIO / VIDEO COMMUNICATION		X		X			X		
MEDICAL / DIAGNOSTIC SYSTEMS	X			X					
POWER SOURCE / CONVERSION							X	X	
POWER MANAGEMENT				X			X		
REACTOR CONTROL				X			X		

FUNCTIONS: / CENTERS:	MOBILE SURFACE								
	ARC	GSFC	JPL	JSC	KSC	LaRC	LeRC	MSFC	SRC
GN&C	X		X	X		X			
ATTITUDE CONTROL	X		X	X		X			
TELEMETRY		X	X	X			X		
INSTRUMENTATION (SENSORS)	X	X	X	X		X	X		
VEHICLE HEALTH MONITORING	X		X	X			X	X	
PROPULSION CONTROL			X	X			X		
RANGE SAFETY / DESTRUCT									
MISC PYRO CONTROL			X	X		X			
HUMAN I/F (CONTROL / DISPLAY)				X			X		
EFFECTOR CONTROL			X	X			X		
EMERGENCY SYSTEMS	X			X		X			
GENERAL DATA PROCESSING	X	X	X	X			X		
MASS DATA STORAGE		X	X	X					
EXPERIMENT CONTROL / MONITOR	X		X	X		X			
NON-TLM DATA COMMUNICATION		X		X			X		
AUDIO / VIDEO COMMUNICATION		X		X			X	X	
MEDICAL / DIAGNOSTIC SYSTEMS	X			X					
POWER SOURCE / CONVERSION							X	X	
POWER MANAGEMENT				X			X		
REACTOR CONTROL				X					

FUNCTIONS: / CENTERS:	FIXED SURFACE								
	ARC	GSFC	JPL	JSC	KSC	LaRC	LeRC	MSFC	SRC
GN&C			X	X		X			
ATTITUDE CONTROL	X		X	X		X			
TELEMETRY	X		X	X			X		
INSTRUMENTATION (SENSORS)	X	X	X	X			X		
VEHICLE HEALTH MONITORING	X		X	X			X	X	
PROPULSION CONTROL			X	X			X		
RANGE SAFETY / DESTRUCT									
MISC PYRO CONTROL	X		X	X		X			
HUMAN I/F (CONTROL / DISPLAY)	X			X		X			
EFFECTOR CONTROL	X		X	X			X		
EMERGENCY SYSTEMS	X			X		X			
GENERAL DATA PROCESSING	X	X	X	X			X		
MASS DATA STORAGE		X	X	X					
EXPERIMENT CONTROL / MONITOR	X		X	X		X			
NON-TLM DATA COMMUNICATION		X		X			X		
AUDIO / VIDEO COMMUNICATION		X		X			X	X	
MEDICAL / DIAGNOSTIC SYSTEMS	X			X					
POWER SOURCE / CONVERSION							X	X	
POWER MANAGEMENT				X			X		
REACTOR CONTROL				X			X		

LAN: Real Time Systems Simulation

Key Attributes:

- Fully Synchronous Operation - Eliminates Time Skews and Aliasing, Simplifies Analysis
- Modular Architecture - Allows Partitioning of Functions, Easy Expansion, Support of Multiple Development Efforts
- Dedicated Real-time Buses - Provide Strict Timing Determinism
- Intelligent Interfaces - Allow Standalone or Fully Integrated Operations
- Standard Interfaces - Allow Rapid Prototyping and Integration of Wide Variety of Off-the-Shelf Components (Ethernet, 1553B, IEEE-488, VME, VAX/VMS, Ada, X-Windows, UNIX)
- Intelligent Data Logging - Provides Data Compression, High-Capacity Data Storage, Real-time Graphic Data Display, Real-time Signal Processing and Analysis

WAN: Real Time Simulation

- Key Issues to be Addressed
  - Laboratory Hardware Installation Variations
  - Laboratory Computer Interface Variations
  - Data Modeling & Transmission Time Domains
  - Application In/Output Formats Deviation From OSI
  - Multiple Operating System Control & Overhead Management Design
  - Software Change Activities (in Progress)
  - System Software Applications & Definitions
  - Transport Layer Connection-Oriented or Connectionless Protocols
  - System Hardware Installation for Space Exploration Initiative

Observations, Recommendations and Conclusions

Observations: Existing Network Real-time Deficiencies

- The Local Area / Institutional Area Networks at some Centers Provide Geographically Restricted Net Access. These would Need Modification for the Development of a Distributed Integrated Simulation.

- Network and Computer Standards which Support Real-time Distributed Work are Still being Evolved by the Industry.
  - The GOSIP / OSI Standards do not Support or Recognize the Need for Time-deterministic Communications.
  - Real-time Operating Systems are not Used Universally Throughout Existing Networks.
  - Probabilistic Synchronization Techniques (Which Could be Used with Current Nets) are not Mature, and Would Present Verification and Validation Concerns if Used To Implement Complicated Flight Systems.

- Internal Standards and Disciplines must Evolve for Cooperative Distributed Simulations
  - Simulation Architecture Standards
  - Data Interface Standards
  - Process Synchronization Standards, etc.

Observations: Future Real-time Network Capabilities

- Some Existing Router and Bridge Equipment has the Potential to Support Time-Deterministic Networks.
- Networks Based on the FDDI Fiber Optic Standard are being Implemented in Parallel with Existing Networks at Some NASA Centers.
  - FDDI Data Rates (100 Mb/s) Represent Approximately a 10 Times Improvement over Existing Nets
  - The FDDI Standards Provide Modes which Allow Time-deterministic Message Passing
- Specific Real-time Network Protocols such as XTP are Now Available
- Widely Used Computer Operating Systems, such as UNIX (System V) [AT&T], AIX [IBM], and HPUX [H-P], Allow Real-time Interprocess Communications
- The NASA Science Internet (NSI) Project Office Is Currently Involved in the Ops and Engineering for the Connection of the Five Research Centers via the National Research and Education Network (NREN), Which Is to Support Real-time Network Requirements.
- The Integrated Services Digital Network (ISDN) Is In the Process of Installation Nation Wide, and Offers Inexpensive, On-demand, Easily Accessible, Moderate-Bandwidth Data Communications with Sufficient Time-determinism To Operate less Communication-intensive Simulations

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General Study Conclusions

- NASA is in the Process of Updating Internal Communications Systems to Conform with the GOSIP / Open Systems Interconnection Standards. This Activity is Preparing NASA for Compliance with Computer Industry Standards Slated for 1995 and beyond. NASA is Conducting This Activity as as an Active Partner in Conjunction with ISO Standards Community.

- The All Existing NASA Avionics Facilities and Laboratories have some Degree of Interconnectivity Hardware and Software Systems. Additional Systems Integration Studies and WAN/LAN Center Coordination is Required for the Implementation of Distributed Simulations such as the Space Exploration Initiative Concept Presented in This Study.

- NASA is On Track for the Evolution of Communications Tools and Protocols for an Integrated Avionics Simulation for the Next Large Space Program.

- Architectural Concepts that utilize "Off-the-Shelf" Components and Multi - System Compatible Protocols will speed the Evolution and Development of The next Generation Space Vehicle, While Satisfying an Intercenter Capability for Integrated Systems Analyses.

Recommendations

- NASA should Establish A Working Group to:
  - Organize and Integrate Avionics Technology Related Information Systems (Technology Sharing)
  - Develop Requirements to be Implemented by Existing Communications Organizations (Reduce Cost of Implementation)
- All NASA Organizations Involved with Integrate Avionics Facilities for New Initiatives should Coordinate Communication Requirements with Existing Networks (Coordination)
- NASA should Generate Information Systems Integration Studies to Address Future Requirements for New Initiatives (SEI, NLS, EOS) as They Relate to Avionics Laboratories and Data Systems.
- The Concept of an Integrated Avionics Test Bed for New Programs such as the Space Exploration Initiative is Feasible, but the Requirements and Justification for such Integration Activities Must be Generated.
- Connectivity Concepts are Integral to the Evolution and Development of The next Generation Space Systems. Existing Work Expended on Connectivity of Ground Systems is Directly Applicable to Flight Avionics Systems for the Future.

Seven Layers, Network Definitions, Acronyms, Abbreviations & References

International Organization for Standardization

Seven Layer - Open Systems Interconnection

1 - Physical Layer - Physical Connection for Transmission of Data between Data Link Entities. Physical Layer Entities Perform Electrical Encoding and Decoding of Data for Transmission over a Medium and Regulate Access to the Physical Network.

2 - Data Link Layer - Provides Communication between Adjacent or Broadcast Systems. The Data Link Layer Performs Formatting, Error Checking, Addressing, and Other Functions Necessary to Ensure Accuracy Data Transmission between Adjacent Systems. The Operation of the Data Link Layer Is Independent of the Particular Network Access Method Used in the Physical Layer.

International Organization for Standardization

Seven Layer - Open Systems Interconnection

3 - Network Layer - Provides Message Routing and Relaying between End Systems on the Same or Interconnected Networks, Independent of the Transport Protocol Use. Hop-by-Hop Network Service Enhancements, Flow Control, and load Leveling. Network Layer Services Are Independent of Interconnecting Network Separation Distance.

4 - Transport Layer - Provides RELIABLE, Transparent Data Transfer between Cooperating Sessions. The Transport Layer Provides the Performance Required by Each Session Entity. Optimization Is Reduced by Concurrent Session System Demands and Network Capacity.

Transparent Protocols Regulate Flow, Detect and Correct Errors, and Multiple Data, on an End-To-End Basis.

International Organization for Standardization

Seven Layer - Open Systems Interconnection (OSI)

5 - Session Layer - Allows Cooperating Applications to Organize and Synchronize Conversation and to Manage Data Exchange. Session Connections Transfer Data Using Transport Connections. During Session Connection, Application Session Services Regulate Dialog by Ensuring Orderly Message Exchange.

6 - Presentations Layer - Syntax of Transferred Data, Specifies or Negotiates the Way Information Is Exchanged between Application Entities Including Application Data Transfer, Application Data Structure, and Data Structure Operations.

7 - Application Layer - Allows for Protocols and Services Required by Particular User Designed Application Processes (Communication between Applications Is Done at Lower Levels.

Network Definitions

Open	The term used to describe no access restrictions such as the NSN.
PSCNI ADMIN	The PSCNI Administrator works in conjunction with the PSC Service Representative (same as PSCNI Site Coordinator) to coordinate with the user to request service.
PSCNI Site Coordinator	The PSC Service Representative functions as the PSCNI Site Coordinator.
CSMA/CD	Carrier sense multiple access with collision detection
NSAP	Network Service Access Points
ES-IS	End System Intermediate System Protocols

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Network Definitions (Cont.)

Terminal Emulation	Software Allowing a Computer to Behave Like a Different type of Terminal (Could Be a Dumb Terminal)
Node	Any Network Device That Has a Network Address
File	Any Information That Can Be Saved to Disk, Printed Out or Transmitted
Volume	General Term for Storage Device; Source or Destination for Information (Disk or Folder)
Host	Multiuser Computer Processor That Serves a Number of Dumb Terminals
Server	A Network Device, Usually Has Software and Delivers Service to Network Users
Client	Computer Receiving Services from a Host or Server

Network Definitions (Cont)

User	Anyone Accessing a Computer Node for Receiving or Sending Information (You)
Packets	Unit of Information Formatted for Transmission Across a Network
Traffic	Transmission Back and Forth Across a Network
Collisions	A Loss of Packets of Information from Simultaneous Transmission
Published	Any File Available to Other Users over the Network File (Report, Newsletter)
Mounted	When a Volume is Recognized by the Computer
Multiuser	Allows Multiple Users to Change Information Simultaneously
Access Privileges	Levels of Passwords, Protection or Access to a File or Volume

Network Definitions (Cont)

Locked	File Can Be Read but Not Modified (Can't Write To)
Read Only	Same as Locked
One Writer	File Can Be Modified by One User While Being Read by Another User
Many Writers	Multiuser, Read and Modified by More Than One User at a Time
Dedicated Server	One Computer Specified to Provide Network Services Typically a Dedicated Computer (No User)
Distributed Server	Computer Can Be a Server and a Client at the Same Time
Background	Operations Running That Are Transparent to User (Print Spooler) While You Do Something Else
Modem	External Device That Prepares Computer Data to a Form for Transmission Over Phone Lines

Network Definitions (Cont.)

M-UNIS	Martin Marietta Engineering Master Operating System
Antonymous Devices	Computer Systems that have no dependency on external systems for operation
Real-Time Processing	A local and/or distributed computing system capable of completing all operations necessary to complete responses in a time domain directly related to the operational system requirements
Computer Network	Two or more computers geographically distributed, usually capable of parallel processing, multipoint access, and simpler central facility requirements.
ARPA	Largest distributed processing system
Packet Switching	Addressed packet data transfer the channel is occupied only during packet transmission

Acronyms and Abbreviations

ADFRF	Ames Dryden Flight Research Facility
ADMIN	Administrator
AMPSLAB	Autonomously Managed Power System Laboratory
ARC	Ames Research Center
ARPA	Advanced Research Project Agency
CPU	Central Processing Unit
DDCMP	Digital Data Communications Message Protocol
DECnet	Digital Equipment Corporation Network
DTR	Data Terminal Ready
ESA	European Space Agency
FTP	File Transfer Protocol
FTS	Federal Telecommunications System
GISS	Goddard Institute for Space Studies
GSFC	Goddard Space Flight Center
NASA HQ	NASA Headquarters
IAN	Institutional Area Network
ICMP	Internet Control Message Protocol
IEEE	Institute of Electrical & Electronic Engineering
IGRP	Interior Gateway Routing Protocol
IPX	Internet Packet Exchange
ISO	Information Systems Office
ISO-OSI	International Standards Organization-Open Systems Interconnection
IUS	Inertial Upper Stage

Acronyms and Abbreviations

JPL	Jet Propulsion Laboratory
JSC	Johnson Space Center
Kbps	Kilobits per second
KSC	Kennedy Space Center
LAN	Local Area Network
LaRC	Langley Research Center
LeRC	Lewis Research Center
MAF	Michoud Assembly Facility
MSFC	Marshall Space Flight Center
NCC	Network Control Center
NCSA	National Center for Supercomputing Applications
NFS	Network File System
NMCS	Network Management Control System
NMIP	Network Management Interface Processor
NPSS	NASA Packet Switched System
NSI	NASA Science Internet
OSI	Open Systems Interconnection
MODEM	Modulator/Demodulator
PC	Personal Computer
PSC	Program Support Communications
PSCN	Program Support Communications Network

Acronyms and Abbreviations

PSCNI	Program Support Communications Network Internet
SCC	Slidell Computer Complex
SNA	System Network Architecture
SNMP	Simple Network Architecture
SPAN	Space Physics Analysis Network
SPO	Shuttle Project Office
SSC	Stennis Space Center
SSE	Software Support Environment
SSM/PMAD	Space Station Module Power Management and Distribution
SSME-HSL	Space Shuttle Main Engine-Hardware Simulation Laboratory
TCP/IP	Transmission Control Protocol/Internet Protocol
TNMIP	Turbo Network Management Interface Processor
WAN	Wide Area Network
XNS	Xerox Network Systems

References

NASA Feb 91 Information Systems	Program Support Communications Network
NASA Jul 91 5-206-9 DC13-LAN-CM	Marshall Space Flight Center Communications Systems Directory Vol: II Local Area Networks
NASA-TM-103510 Dec 90	Research and Technology 1990, Annual Report of the Marshall Space Flight Center
NASA 27 Aug 91 Lockheed Sanders Inc.	NASA Open System Architecture Study
Martin Marietta 13 May 86 James J. LaBelle	NTB Communications Subsystem Alternative Operations Concepts

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Martin Marietta Apr 86	Real Time Distributed Systems Laboratory
Martin Marietta Jul 89	Total Quality Management
Martin Marietta Nov 90 Ted Phillips	Software Requirements Specification for Martin Marietta Unified Information System (M-UNIS) of the Engineering Propulsion Laboratory (EPL)
Martin Marietta Sep 90 Rainer Koenig	System Requirements for the Martin Marietta Astronautics Group Unified Information System (U-UNIS)
Martin Marietta 24 Sep 91 Jim McKinnis	D-34S, Space Transfer Vehicle Advanced Technology

a-2

References

Orbital Sciences Corp. Feb 91	Building 46 Johnson Space Center Information Network
ANSI-T1-107 88	Telecommunications Digital Hierarchy Formats Specifications
ANSI/IEEE Std 488.1 Std 488.2	Digital Interface for Programmable Inst. Standard Codes, Formats, & Common Cmds
Std 488.3 Std 488.4	
MSFC - STD-417A	Vehicle Configuration Systems, Data Requirements, for MSFC Data
CJ Suppl, R.T. Suppl 1983 Howard James & Co Indianapolis, Indiana	Computer Dictionary
FIPS PUB 146 24 AUG 88	Government Open Systems Interconnection Profile (GOSIP), Hardware & Software Standards, Network Protocols

References

FIPS-PUB-62 30 Dec 90	I/O Channel Interface Channel Level Power Control Interface, Specifications for Magnetic Tape Subsystems, Operational Specifications for Rotating Mass Storage Systems
FIPS-PUB-60-2 22 Feb 84	I/O Channel Interface
FIPS-PUB-158 29 May 90	User Interface Component of the Applications Portability Profile
FIPS-PUB-107 31 Oct 84	Local Area Network: Baseband Carrier Sense Multiple Access with Collision Detection Access Method and Physical Layer Specifications and Link Layer Protocol
FIPS-PUB-41 30 May 75	Computer Security Guidelines for Implementing the Privacy Act of 1974

References

MIL-STD-1788A 15 May 85	Avionics Interface Design Standard
MIL-STD-1776 12 Aug 83	Transmission Control Protocol
MIL-STD-1782 10 May 84	Telenet Protocol
MIL-STD-1781 10 May 84	Simple Mail Transfer Protocol
MIL-STD-1780 10 May 84	File Transfer Protocol
MIL-STD-1777 12 Aug 83	Internet Protocol
MIL-D-28003	Digital Representation for Communication for Illustration Data
MIL-HDBK-420 20 MAR 87	Site Survey Handbook for Communications Facilities

Technical Directive 07,

Lunar Transportation System

Lunar Transportation Analysis - Agenda

Task 1 - Ground Based LEO Rendezvous and Docking Study

Study Overview	J. Hodge
Mission Analysis	S. Earley
Concept Selection and Definition	L. Rauen/R. Spencer
Operations and Programmatics	J. Cathcart
Summary	J. Hodge

Task 2 - Technology/Advanced Development

Overview "Design Of Experiments" (DOE)	J. McKinnis E. Kiefel

Study Overview

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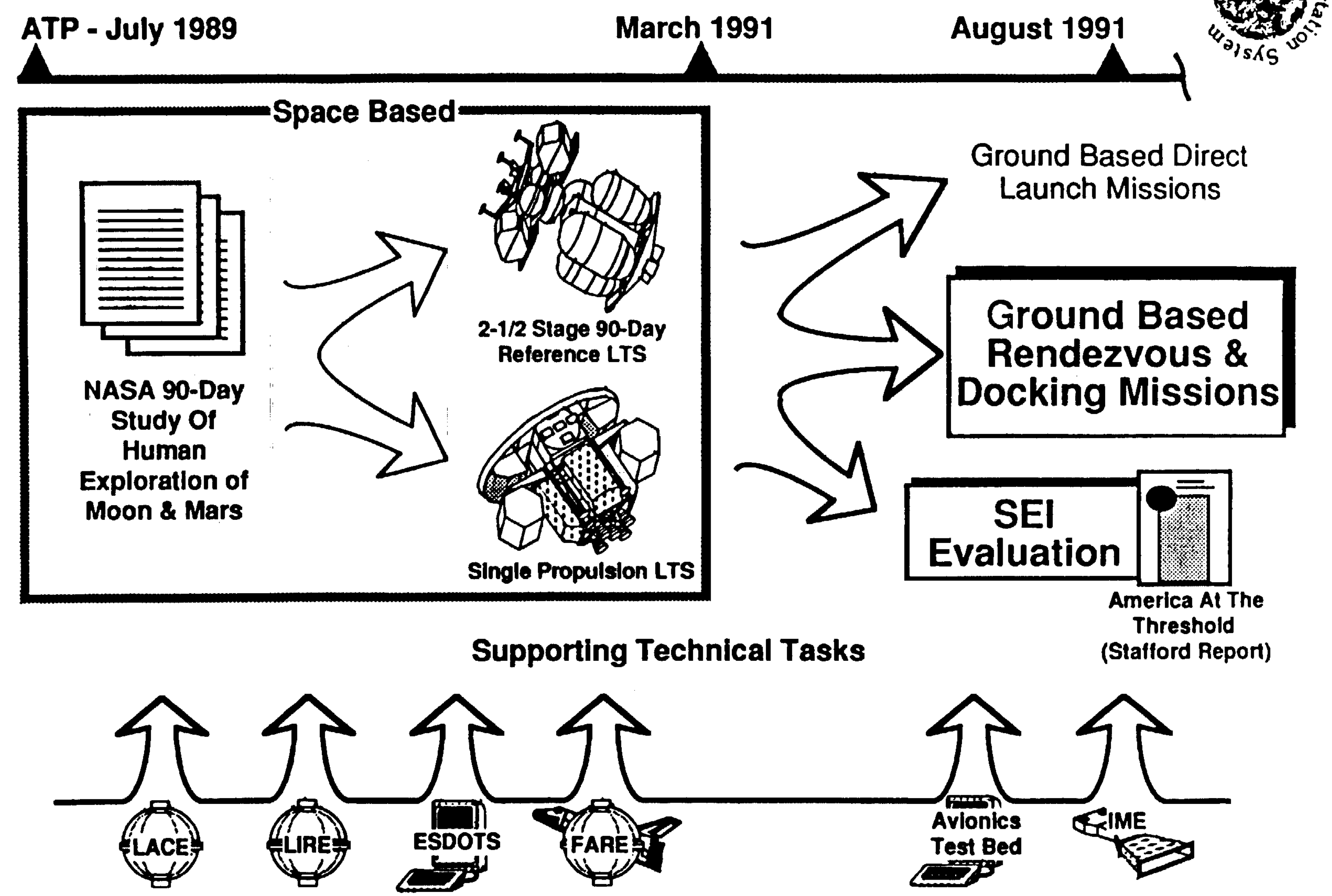
Study Derived From Space-based LTS



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Accomplishments Bounded By Key Objectives

Lunar Transportation System

- Show "Rendezvous & Docking" Is Feasible For Lunar Missions

- Simple But Innovative

- Utilization of Existing Databases And Lessons Learned
  - Phase I STV
  - Apollo

- Joint Development and Ownership of Groundrules
  - HLLV
    - Performance Provided By MSFC (Mass & Volume)
    - Two Flights per Mission
  - System optimized Across HLLV Family
  - No In-Space or Lunar Surface Services
  - No Heat Shield Penetrations
  - Chemical Propellants
  - Engines Isp's
    - RL10B-2 = 468 sec (e =330)
    - RL10C-1 = 468 sec (e =400)
    - RL10A-4 = 449 sec (e =84)

- Parametric Results Supportive of Planned and Future Efforts.

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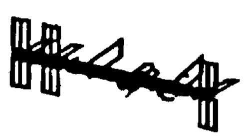
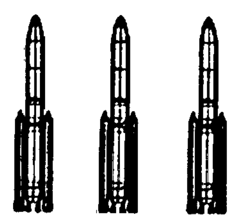
Why Evaluate A Rendezvous & Docking Mission

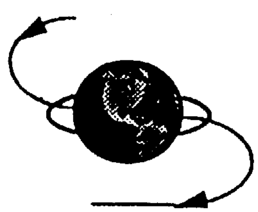
- Space-basing Imposes Critical Requirements On Infrastructure



Extensive SSF Support Required
- Manpower
- Support Equipment
- Assembly Time



2-3 HLLV Launches (Minimum - 120 mt Class)



Technology/Advanced Development Dependence

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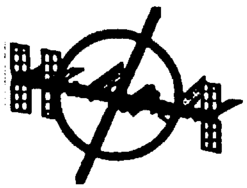
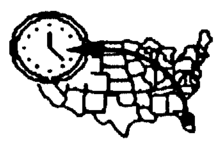
Why Evaluate A Rendezvous & Docking Mission

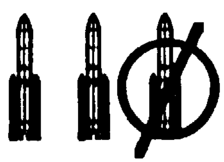
- Attributes Of Rendezvous & Docking Reduces Or Eliminates Critical Infrastructure Requirements



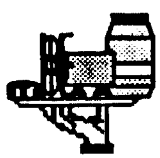
No SSF Support Required



Mission Flexibility



HLLV Quantity/Size Reduction



Utilizes Existing Hardware

Rendezvous & Docking in LLO Extendible to LEO

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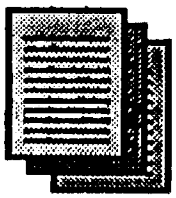
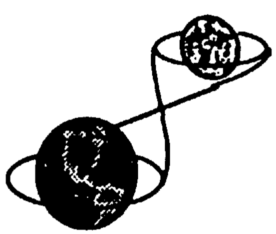
Tasks Key To Study Performance

Identification of Requirements & Interfaces (Hodge, Rauen)



- Requirements Key to System Configuration & Performance
- Groundrules & Assumptions a Collaboration of Contractors & NASA.
- Allocated Mission Functions to System Segments
- Defined Infrastructure Interfaces

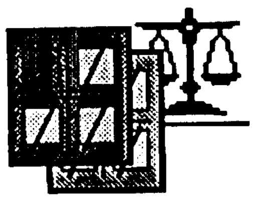
Mission Analysis (Earley, Smith, Joyner)



- Developed Mission Profile
  - "ΔV" Budget (LEO, LLO, etc.)
  - Timeline
- Optimized LEO and LLO Orbit Altitude
- Trajectories

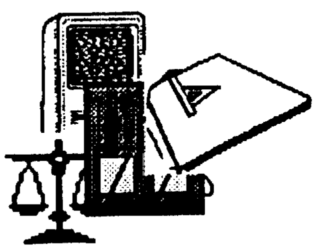
Tasks Key To Study Performance

Initial Concept Downselect (Rauen, Earley, Spencer)



- Optimized and Implemented a Process and Criteria
- Identified LTS Candidates and Bounding HLLV Options
- Relative Cost Parametrics Developed
  - Lunar Surface Mass
  - Mission Model
  - HLLV Options
- Screened Candidate Configurations to Four

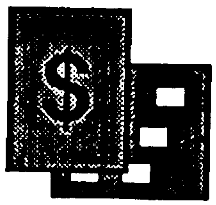
Final Concept Selection & Definition (Spencer, Earley, Rauen)



- Defined Payload Ranges (Piloted & Cargo) Across HLLV Options
- Configuration Recommended - Optimized Across Four HLLV Options
- LTS Defined in System and Subsystem Design
  - Vehicle
  - Propulsion
  - Avionics
  - Life Support

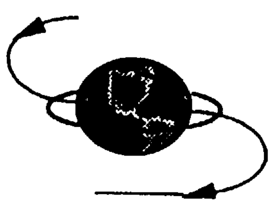
Tasks Key To Study Performance

Operations and Programmatics (Cathcart, Rauen)



- Operational Process Unique to Rendezvous & Docking Mission
- Ground Processing Approach Recommended
  - Key Facility Interfaces
  - Timelines
  - Manpower Skill Requirements
- Formalization of "Rendezvous & Docking" Program Plan
  - Study
  - Test
  - Development
  - Operations
- Definition of Cost Sensitivities

Technology/Advanced Development (McKinnis, Kiefel)



- Continuing Cost/Performance Benefits Assessment
- Key Technology Sensitivities Defined Through "DOE" Analysis Method

System Requirements & Interfaces

Lunar Transportation System

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Requirements Provide Study Direction

Level I - Mission Objective/Statement

- Support the Exploration and Habitation of the Lunar Surface Using a Transportation System That Utilizes a Rendezvous and Docking Approach to LEO Assembly

Level II - Architecture/Transportation System Requirements

- Manned Ops
- Schedule
- Environments
- Interfaces
- Verification
- Flight Rate
- Delivery
- Duration

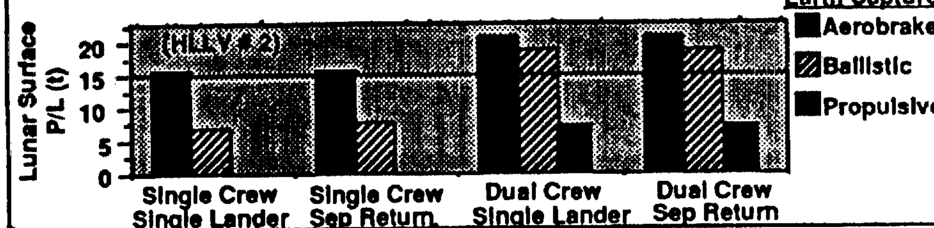
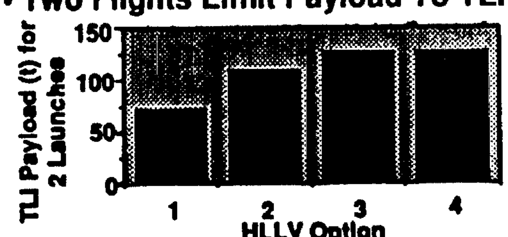
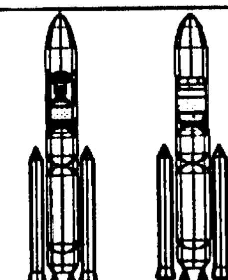
Derived From:
- NASA Specifications/Standards
- Study Groundrules
- STV Phase I Study Results

Level III - System Design Requirements

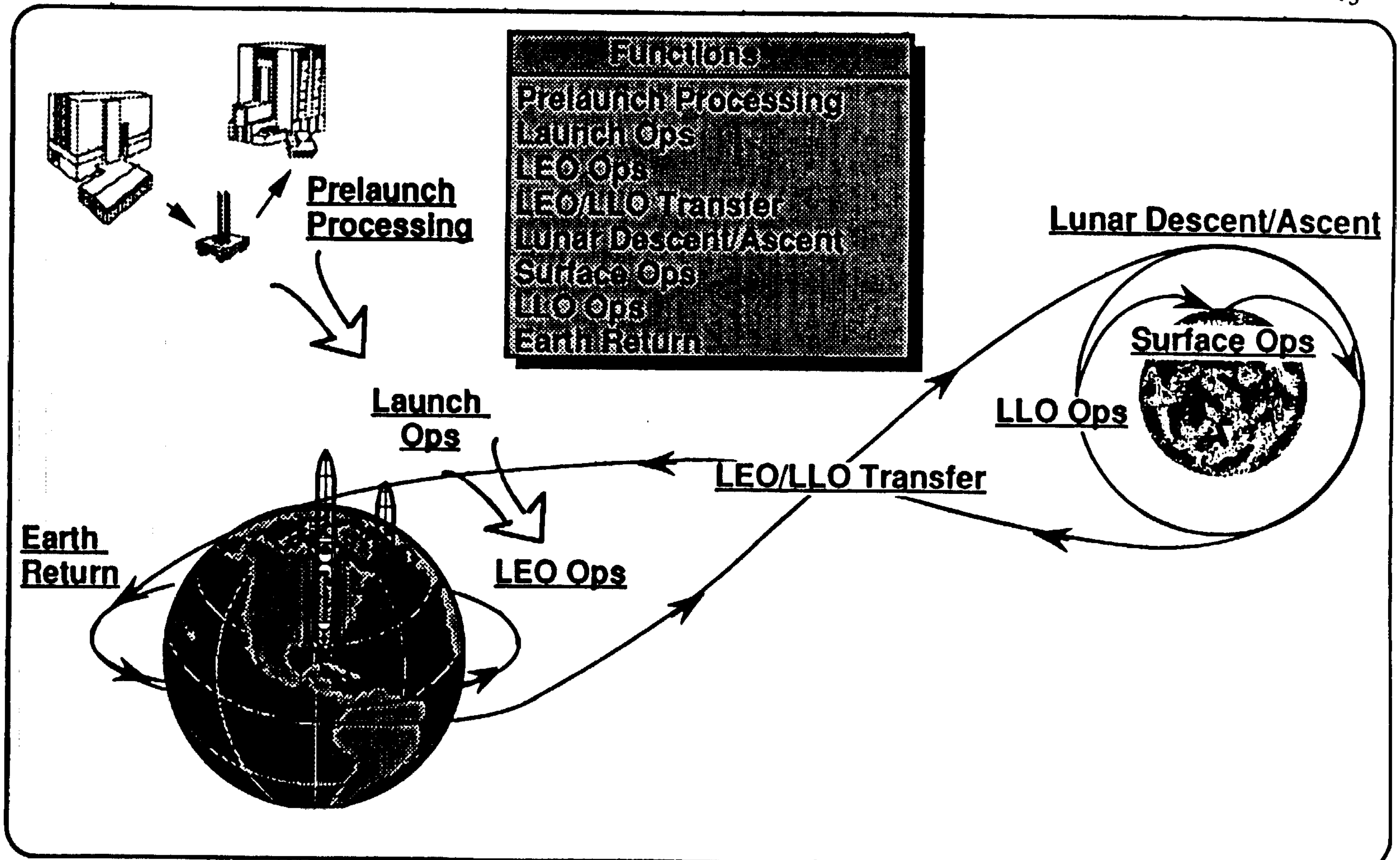
- Prelaunch Processing
- Launch Ops
- LEO Ops
- Lunar Transfer
- Surface Ops
- Earth Return

Derived From:
- Functional Analysis
- Performance
- Operational Analysis

Key Requirements Impact System Definition

Level II System Requirement	Design Impact/Issues
System Assembly Performed by LEO Rendezvous &Docking	• LEO Stationkeeping and Autonomous Space Operations Increase Propulsion and Avionics Complexity
Manned Missions Deliver a Crew of Four People and Between 0 & 15 Tonnes of Cargo to Lunar Surface	• Capabilities Vary for Each Vehicle - Not All Vehicles Can Complete the Mission 
System IOC Is 2003 For Cargo Missions and 2006 For Manned Missions	• Technology Must Be At Level 6 By 1998 • Schedule Risk Mitigation May Require Block Changes to Support System Evolution
HLLV Flights Limited to Two Per Lunar Mission	• Two Flights Limit Payload To TLI  
Mission Duration Will Include a 72 Hour Transit Time With a 30-180 Day Lunar Surface Stay For Manned Missions	• Crew Cab(s) Must Accommodate LSS & Provisions For Mission • Tanks Must Be Designed to Account For Propellant Boil Off

Mission Defines Top Level Functions



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Top Level Functions Allocated to Segments

Functions	Space Segment Hardware	Space Segment Software	Ground Segment Hardware	Ground Segment Software	Ground Segment Personnel	Ground Segment Facilities
Prelaunch Processing						
Package, Handle, Transport	√		√		√	
Receive & Inspect	√		√		√	√
Assemble, Int. & C/O lunar transportation elements	√	√	√	√	√	√
Integrate & C/O Payload (Cargo/Crew) with lunar elements	√	√	√	√	√	√
Integrate lunar system with LV	√	√	√	√	√	
Launch Ops						
Provide prelaunch environment for Lunar elements/payload	√		√			
Monitor LTS		√		√		
Separate from LV	√	√		√		
LEO Ops						

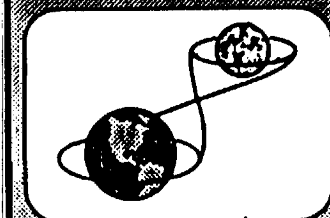
Roadmap For WBS and System/Subsystem Specification and Operations Concept Development. First Step In Development of Function/Hardware Traceability

System Level Interfaces Identified

Lunar Transportation System

Functions \ System Interfaces	Range Safety	Launch System	Mission Control	Payload (Cargo/Crew)	Crew Transport	Lunar Surface
Prelaunch Processing						
Package, Handle, Transport						
Receive & Inspect						
Assemble, Integrate and C/O lunar transportation elements						
Integrate & C/O Cargo/ Crew with lunar transp. elements				√		
Int LTS with launch vehicle		√				
Launch Ops						
Provide prelaunch environ.	√	√	√			
Monitor LTS	√	√	√			
Separate from launch vehicle	[illegible]	[illegible]	[illegible]			

Provides Basis For System IRD and ICDs. Supports System Traceability Process



Mission Analysis

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Mission Analysis - Topics

Lunar Transportation System

- ΔV Allocation
- Typical Mission Timeline
- Earth Orbit Rendezvous
- Earth-Lunar Transfer
- Lunar Ascent
- Lunar Orbit Rendezvous
- Summary

ΔV Allocation (Maximums)

Lunar Transportation System

Event	ΔV Allocation (m/s)	Comments
LEO Circularization	70	Circularize 56 x 300 km Orbit
LEO Altitude Maintenance	20	30 days at 300 km Altitude
LEO Node Change	125	15 min. Launch Window
LEO Rendezvous & Docking	105	Execute Rendezvous & Dock
Trans-Lunar Injection	3135 *	76 hr. "Free Return" Trajectory
Trans-Lunar TCMs	10	Make Necessary Corrections
Lunar Orbit Insertion	915	300 km Altitude
LLO Inclination Change #1	130	5° of Capability, If Needed
De-Orbit #1	215	Target "Free Return" Descent
Lunar Descent	1900	Execute Landing
Lunar Ascent	1985	To 37 x 280 km + Circularize
LLO Node Change	5	Align Orbital Planes
LLO Δ True Anomaly	65	10 min. Launch Window
LLO Rendezvous & Docking	50	Execute Rendezvous & Dock
De-Orbit #2	65	De-Orbit Landing Stage
LLO Inclination Change #2	130	5° of Capability, If Needed
Trans-Earth Injection	915	76 hr. Earth Return Trajectory
Trans-Earth TCMs	10	Make Necessary Corrections
Ballistic Entry Control †	10	Control Ballistic Earth Entry
Aeromaneuver Control §	5	Control Aerobrake Maneuver
LEO Circularization §	40	Circularize at 185 km

† Ballistic Returns

§ Aerobraked Returns

* Not Including Losses

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Typical Top-Level Mission Timeline (Outbound)

Lunar Transportation System

Event	Time (dd:hh:mm)	Comments
HLLV #1 Launch	00:00:00	HLLV #1 Lift-Off
HLLV #1 Initial Orbit Insertion	00:00:15	56 x 300 km Earth Orbit
HLLV #1 Circularization	00:00:59	Circularize at 300 km
TLI Stage Altitude Maintenance	01:00:59	Daily Reboost
HLLV #2 Launch	29:03:13	HLLV #2 Lift-Off
HLLV #2 Initial Orbit Insertion	29:03:20	56 x 230 km Earth Orbit
HLLV #2 Circularization	29:04:03	Circularizes at 230 km
LEO Node Change	29:06:57	Aligns Orbital Planes
LEO Rendezvous Phasing	29:23:17	Orbital Phasing & Transfer
LEO Terminal Rendezvous	29:23:50	Complete Final Approach
LEO Station Keeping	30:00:34	Prepare for Proximity Ops
LEO Proximity Ops & Docking	30:00:59	Docking Complete
Begin TLI Burn	30:13:00	TLI Phasing & C/O Complete
Trajectory Correction Maneuver	33:03:00	Target Free Return
Initiate LOI Burn	33:16:50	Circular LLO at 300 km
Begin LLO De-Orbit	34:04:20	Target Descent Trajectory
Begin Terminal Descent	34:04:55	Begin Terminal Descent
Land on Lunar Surface	34:05:07	Lunar Surface Touch Down

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Typical Top-Level Mission Timeline (Inbound)

Lunar Transportation System

Event	Time (dd:hh:mm)	Comments
Lift-Off from Lunar Surface	214:05:07	Launch Lunar Ascent Stage
Initial Lunar Orbit Insertion	214:05:13	37 x 280 km Lunar Orbit
LLO Circularization	214:06:15	Circularize at 280 km
LLO Node Change	214:08:33	Aligns Orbital Planes
Rendezvous Phasing	214:18:57	Orbital Phasing & Transfer
LLO Terminal Rendezvous	214:19:48	Complete Final Approach
Station Keeping	214:20:57	Prepare for Proximity Ops
Proximity Ops & Docking	214:21:22	Docking Complete
Lander Stage Separation	214:23:40	Separate Lander & TEI Stage
Lander Stage De-Orbit	215:01:58	Target Impact Zone
Initiate TEI	215:04:16	TEI Phasing & C/O Complete
Trajectory Correction Maneuver	215:17:52	Target Earth Entry Point
Earth Entry	218:08:16	Begin Ballistic Entry

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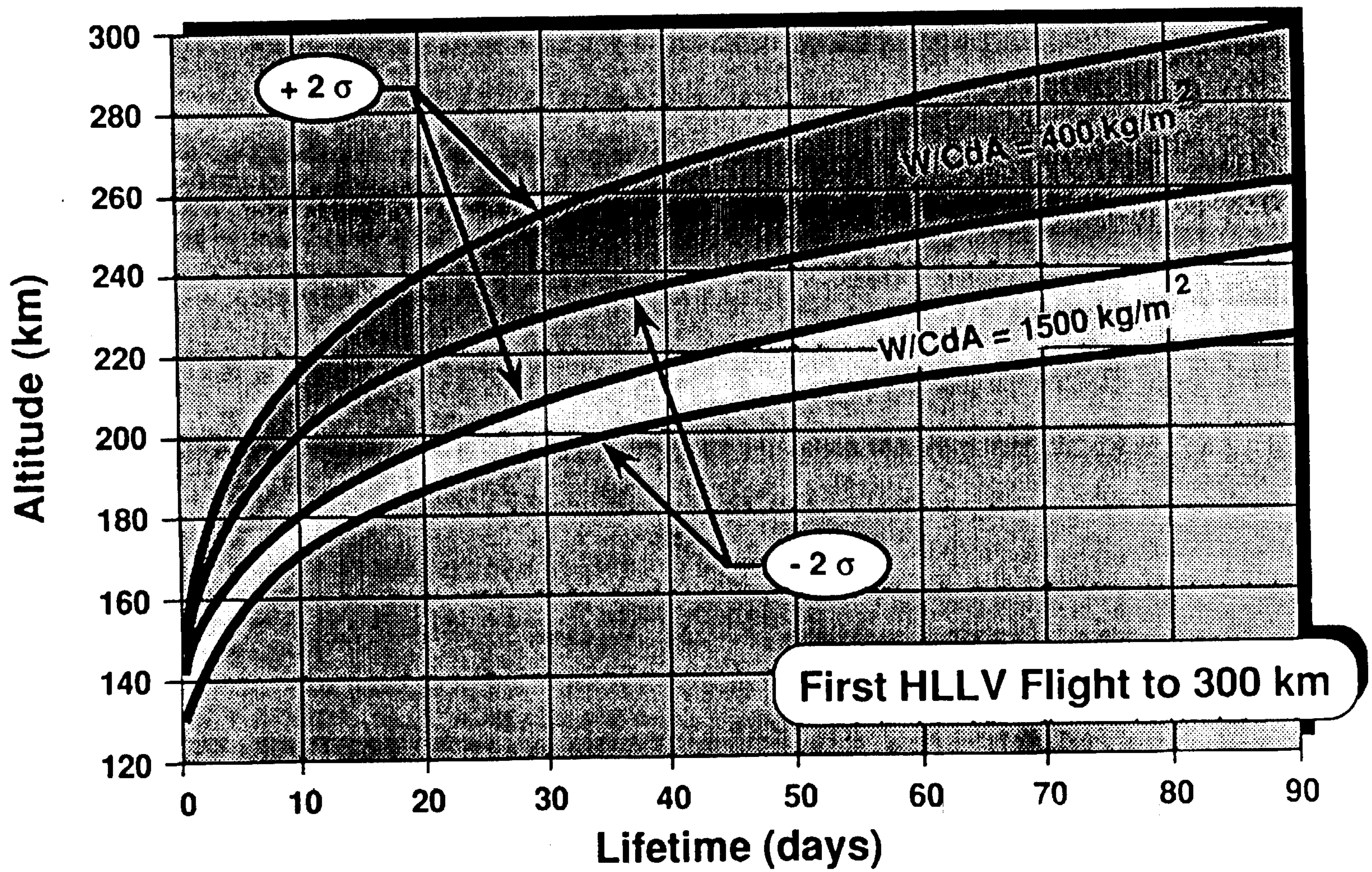
Lifetime Analysis - Solar Max (2001)



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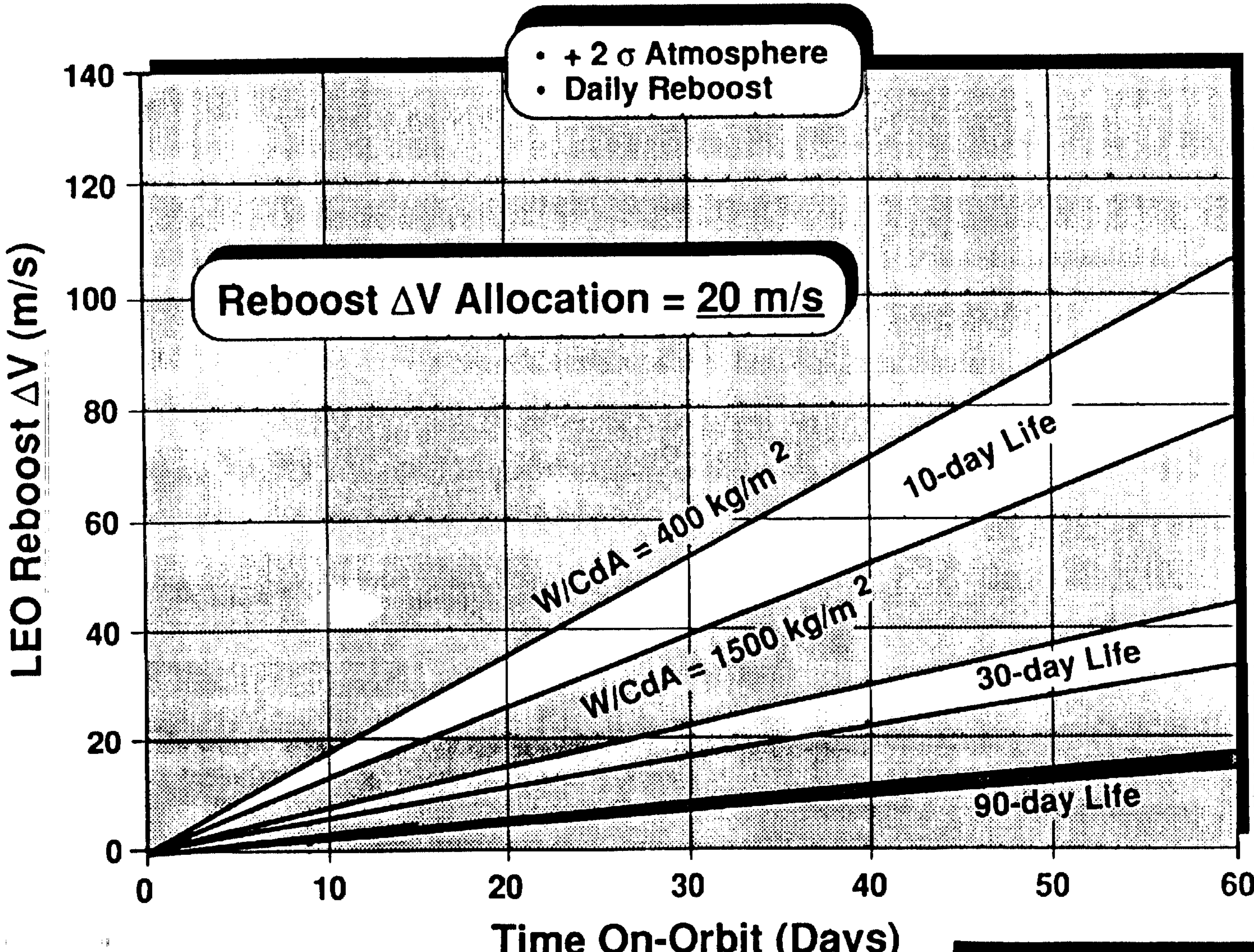
Reboost Requirements Due to Orbit Decay



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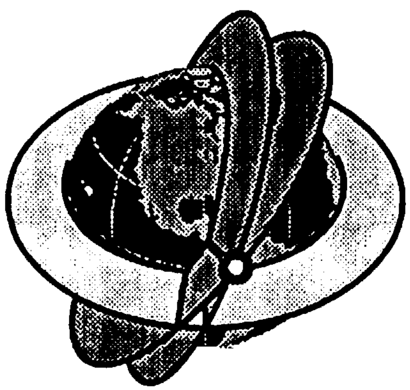
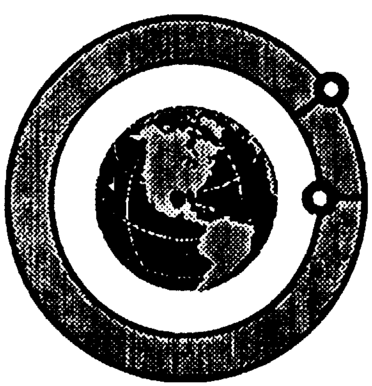
Ascending Node & True Anomaly Changes



Ascending Node Change

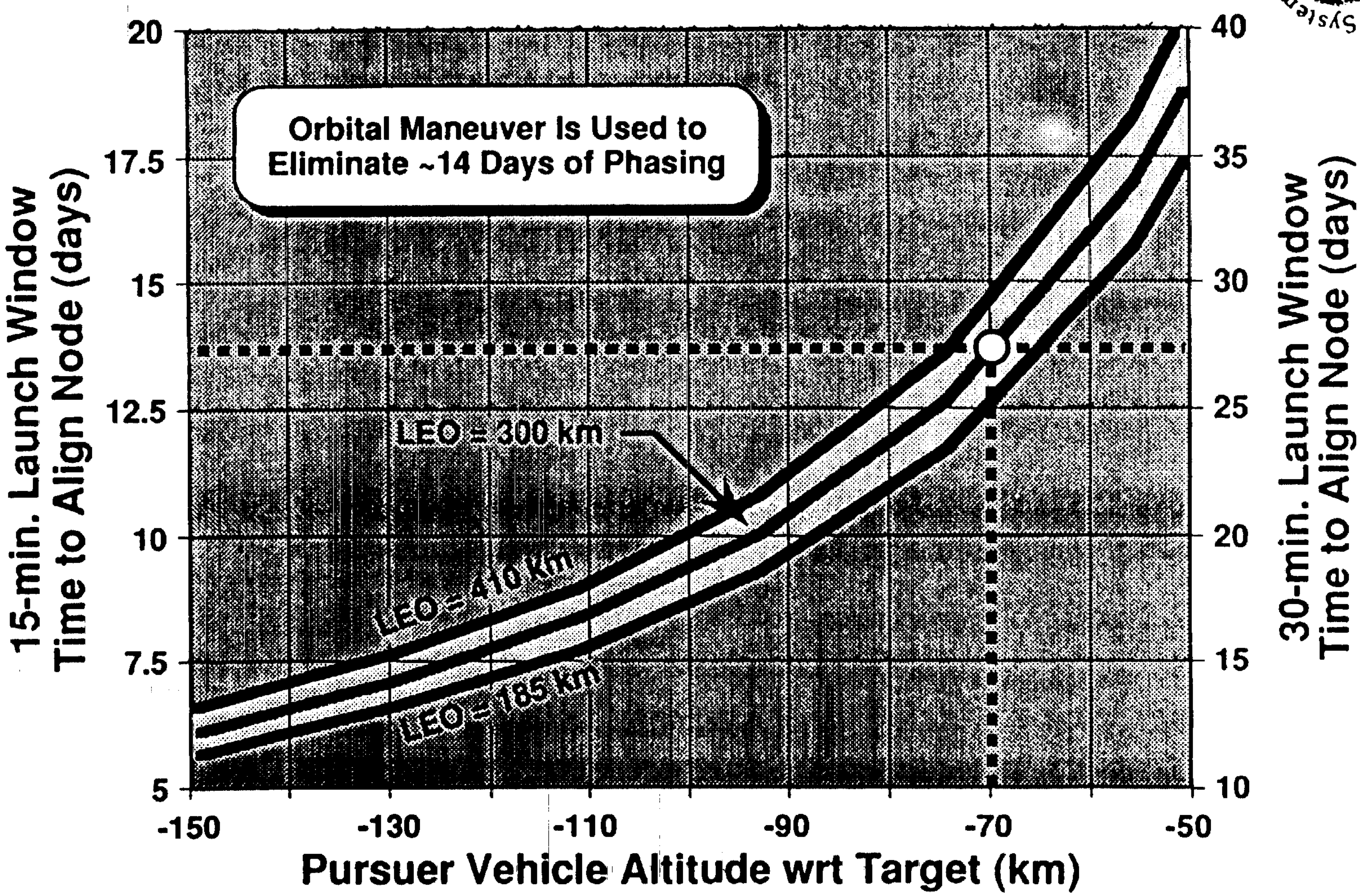
- Largely Determined by the Launch Window
- Earth's Oblateness Has Long Term Effects
- Must Be Aligned Along with <u>Inclination</u> for the Orbits to be "Coplanar"



True Anomaly Change

- Largely Determined by the Launch Window
- Differential Orbital Velocities Cause the Pursuing Vehicle to "Catch-Up" or be Caught
- The Pursuing Vehicle Must Be at the Proper True Anomaly Before It Can Execute the Rendezvous

ETO Launch Window Impact on Ascending Node



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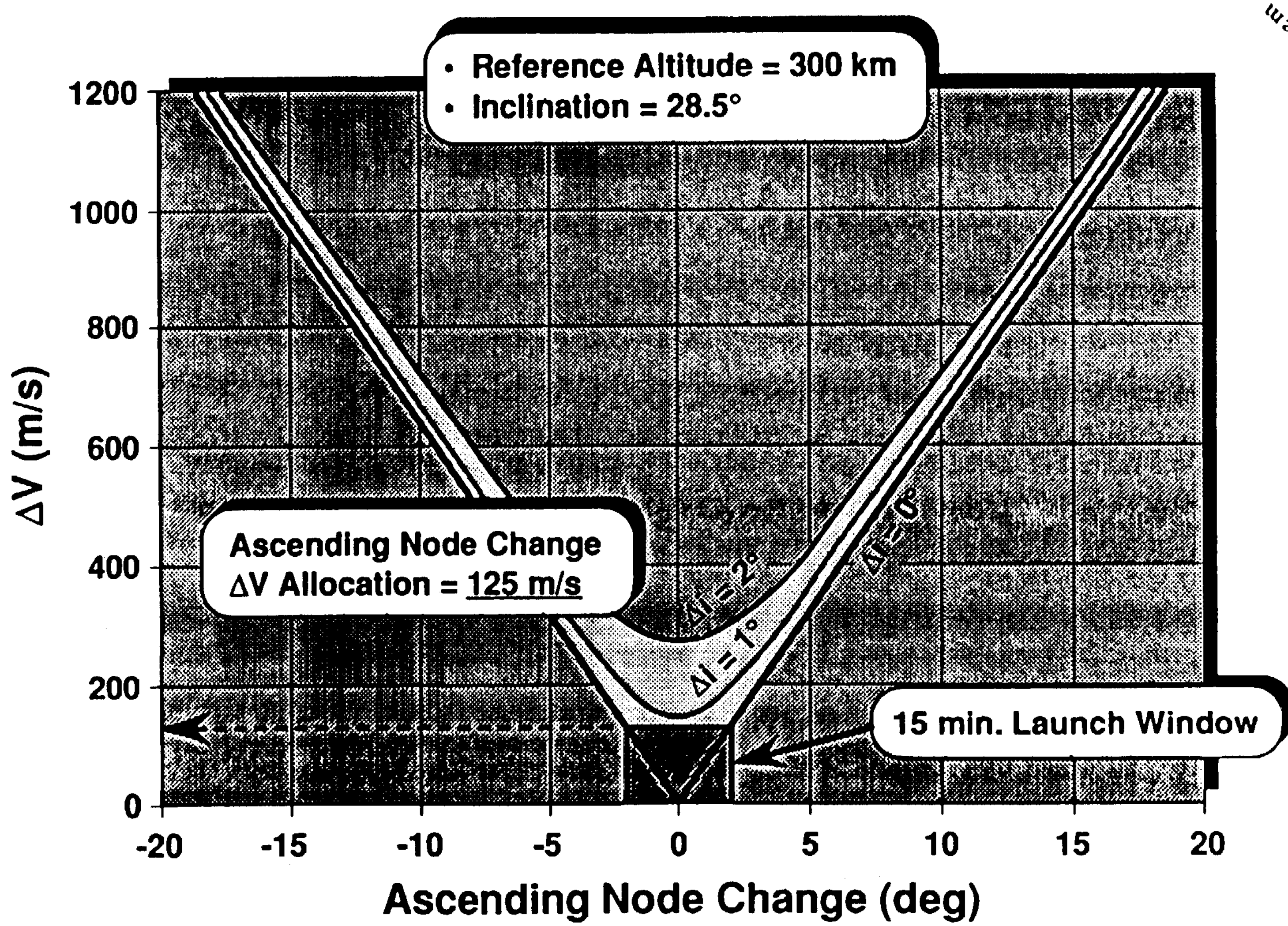
LEO Node Alignment ΔV



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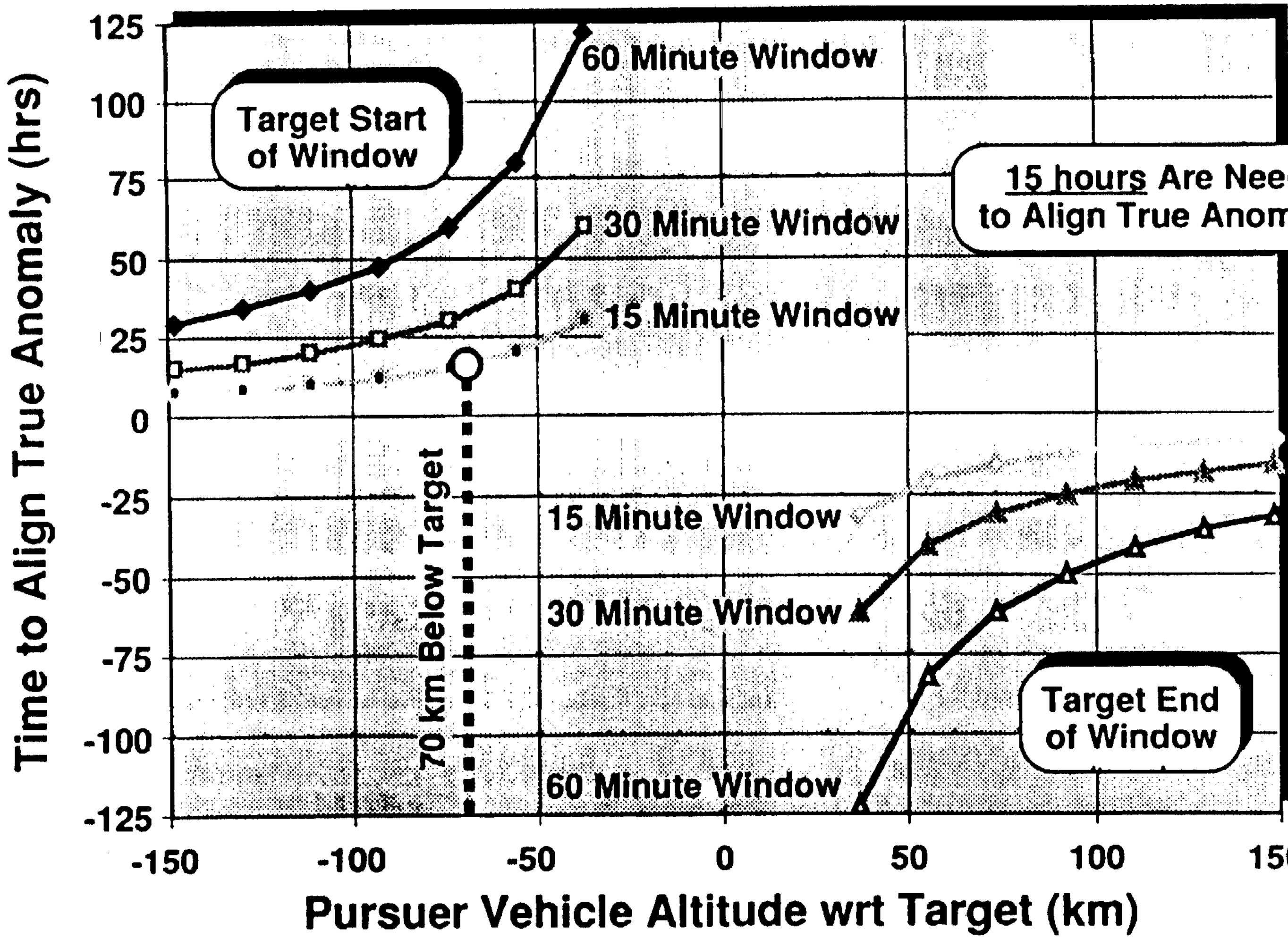
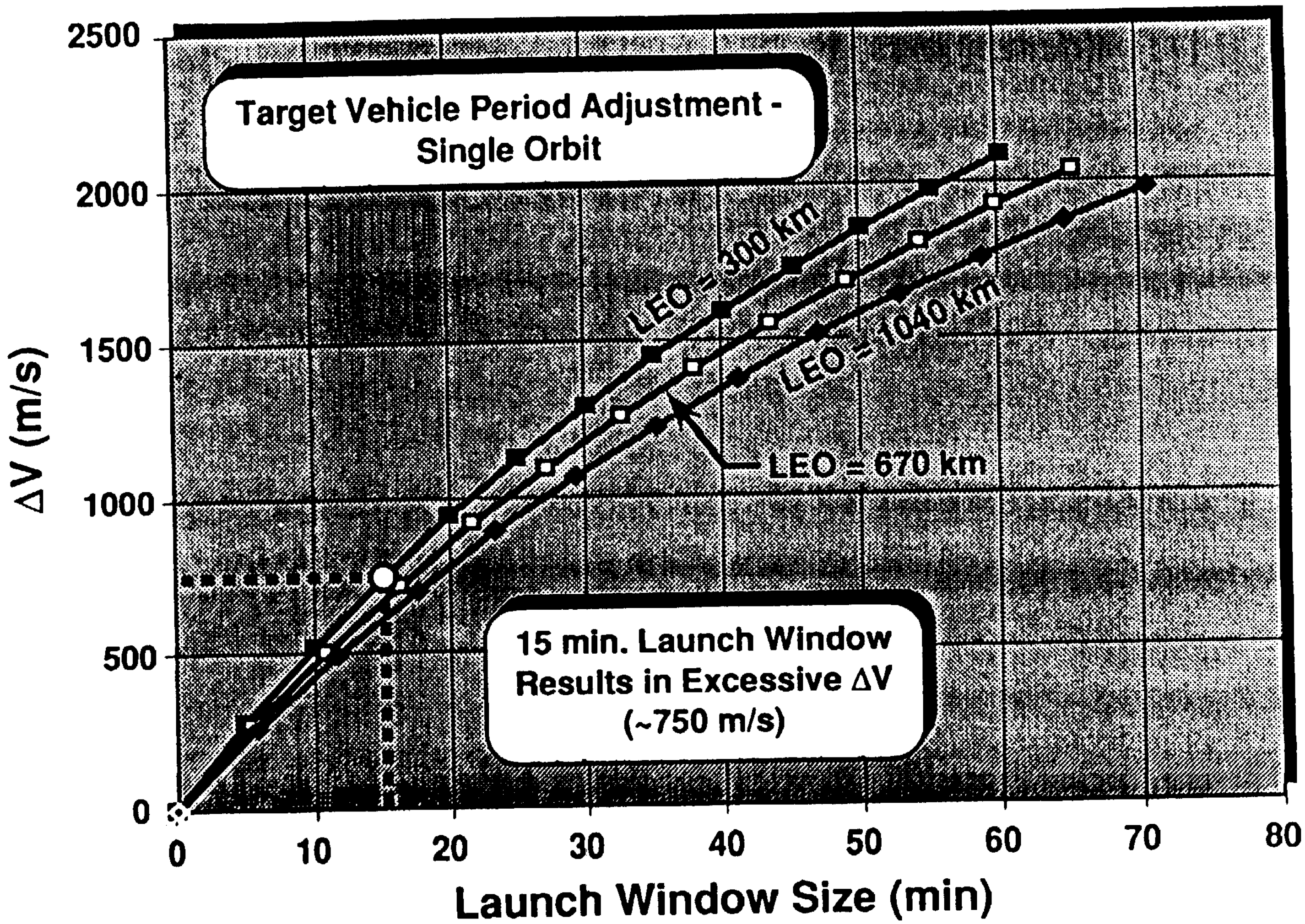
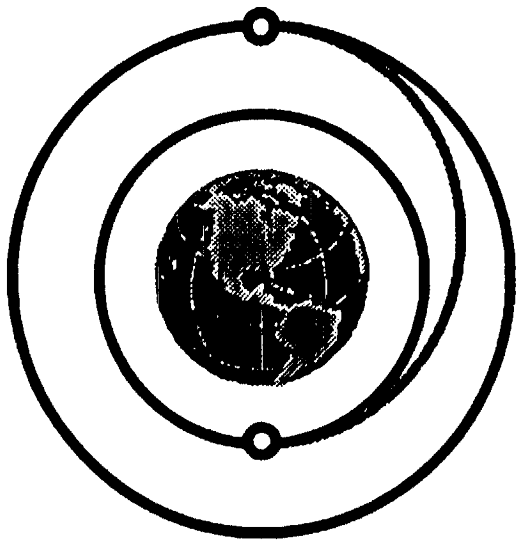
Launch Window Impact on True Anomaly



LEO True Anomaly Alignment ΔV

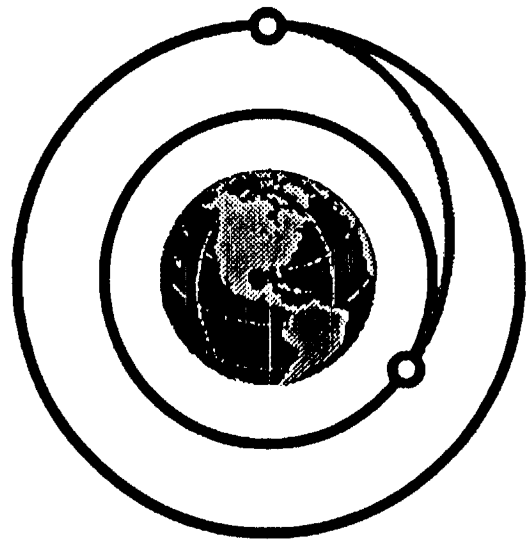


180° & 130° Orbital Transfers



<u>180° Orbital Transfer</u>

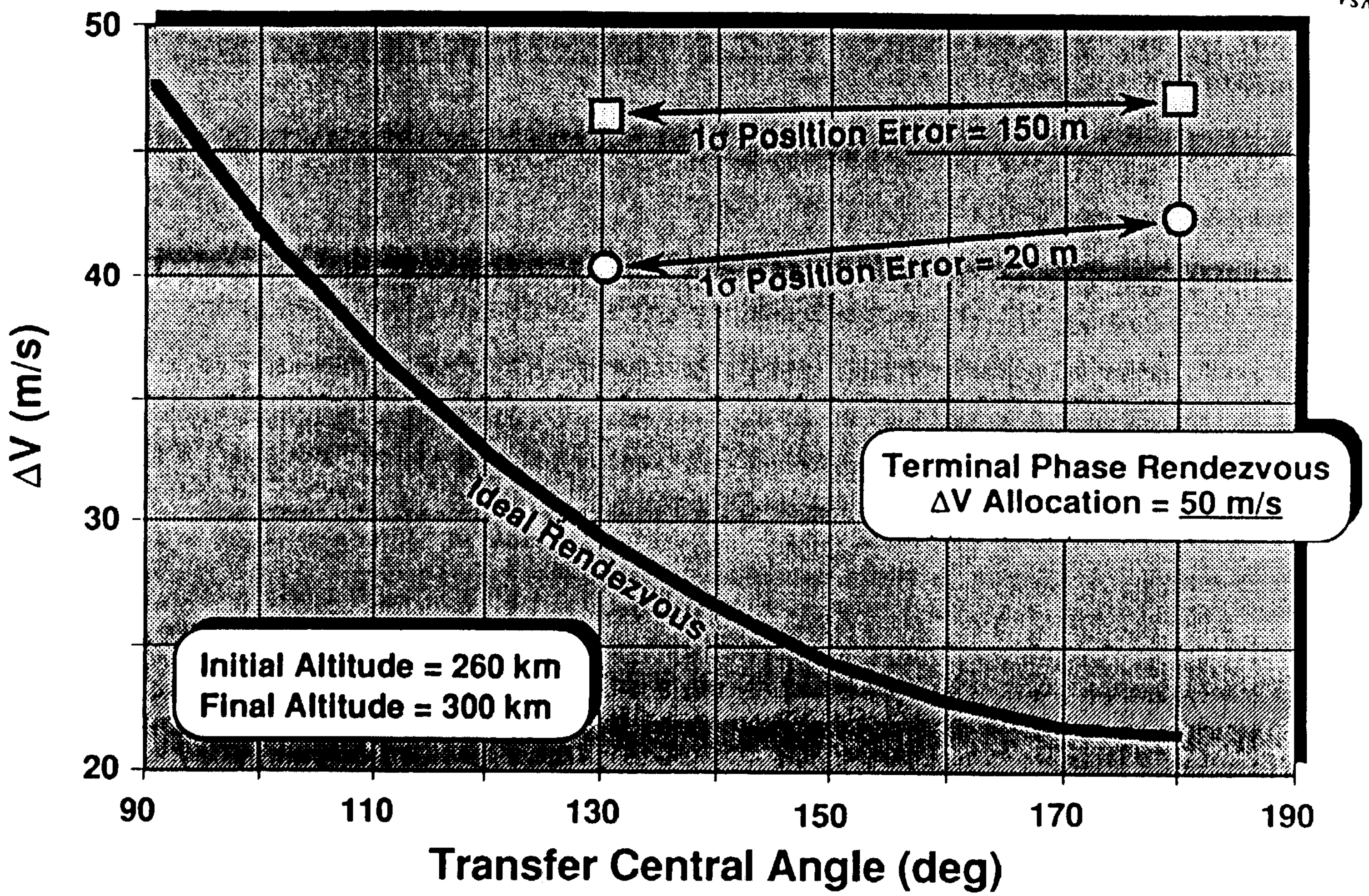
- Most Efficient, <u>Ideally</u>
- Longer Transfer Times
- Less Accurate



<u>130° Orbital Transfer</u>

- Less Efficient Ideally
- Shorter Transfer Times
- More Accurate

Terminal Phase Rendezvous ΔV



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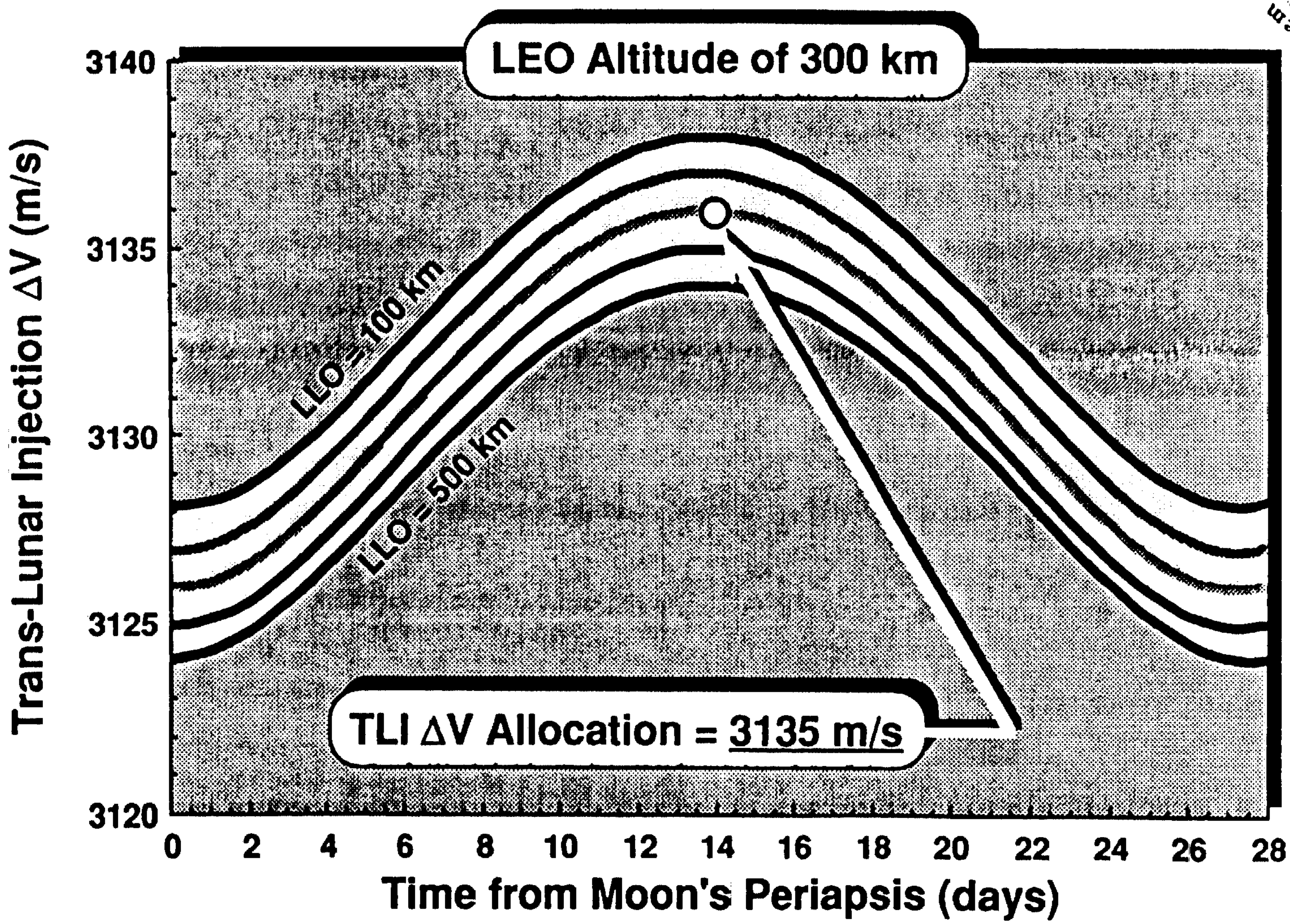
Trans-Lunar Injection ΔV vs Moon's Position



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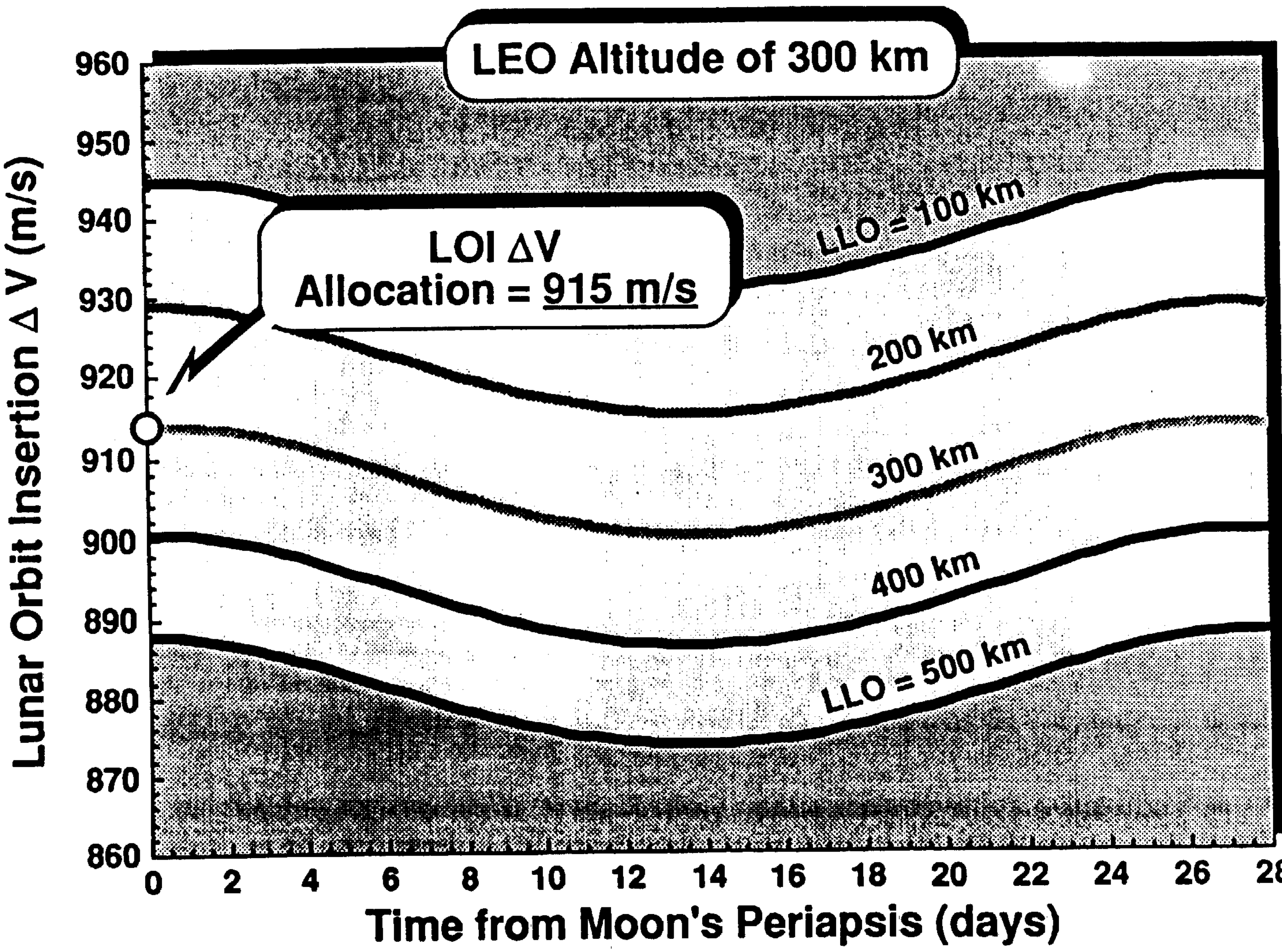
SE910812-01A

Lunar Orbit Insertion ΔV vs Moon's Position



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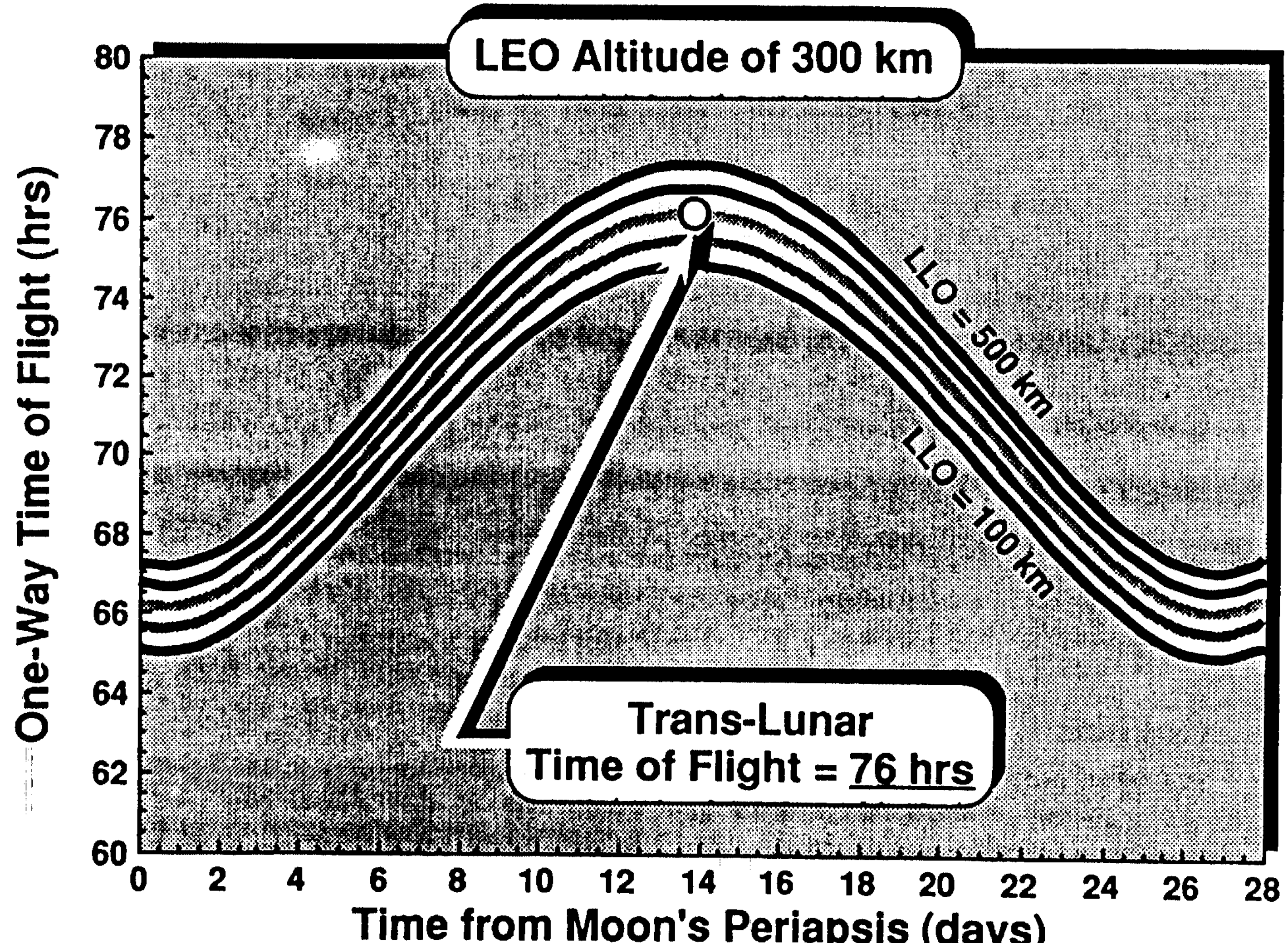
Trans-Lunar Time of Flight vs Moon's Position

Lunar Transportation System



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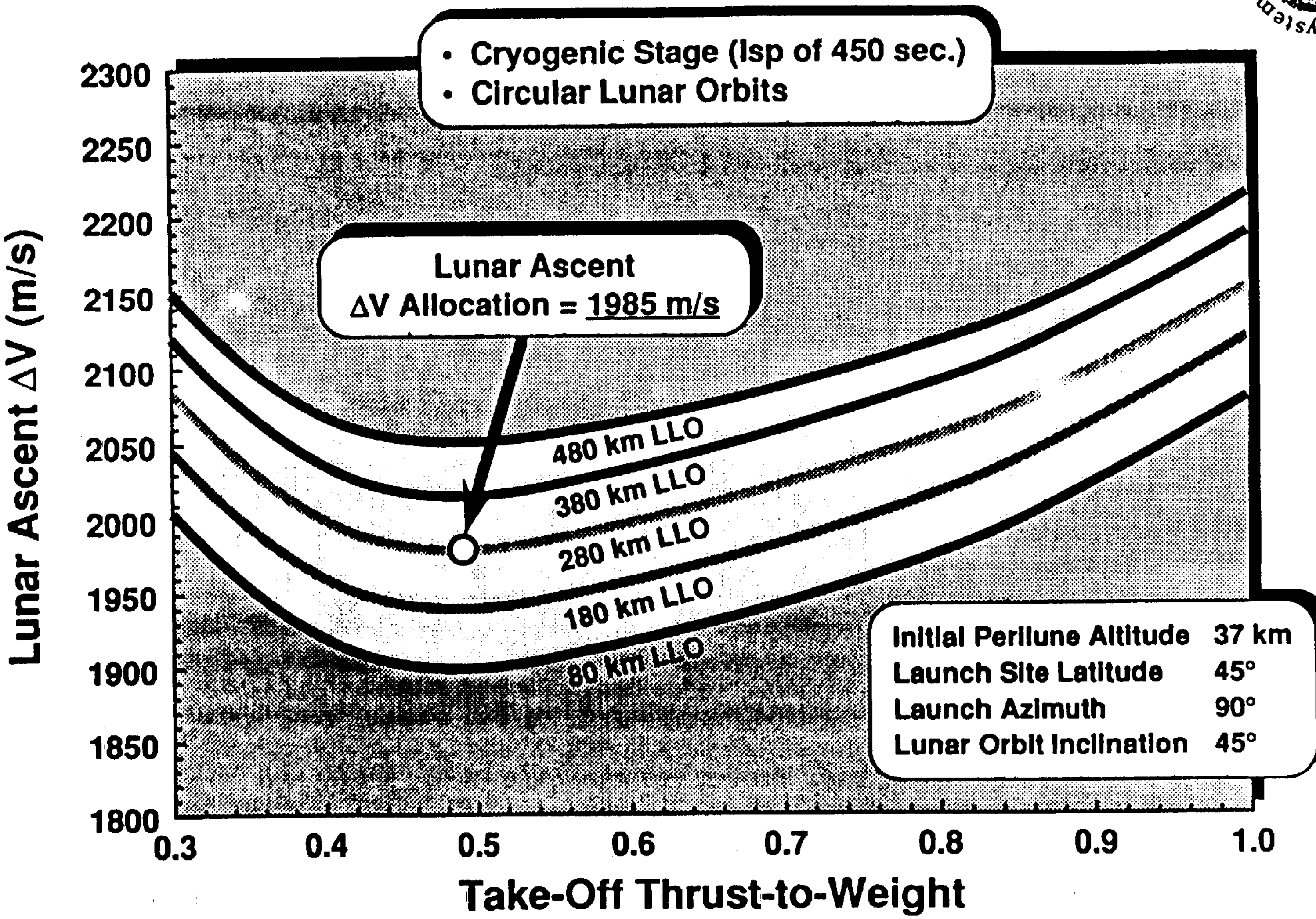
Lunar Surface Ascent ΔV vs Take-Off T/W



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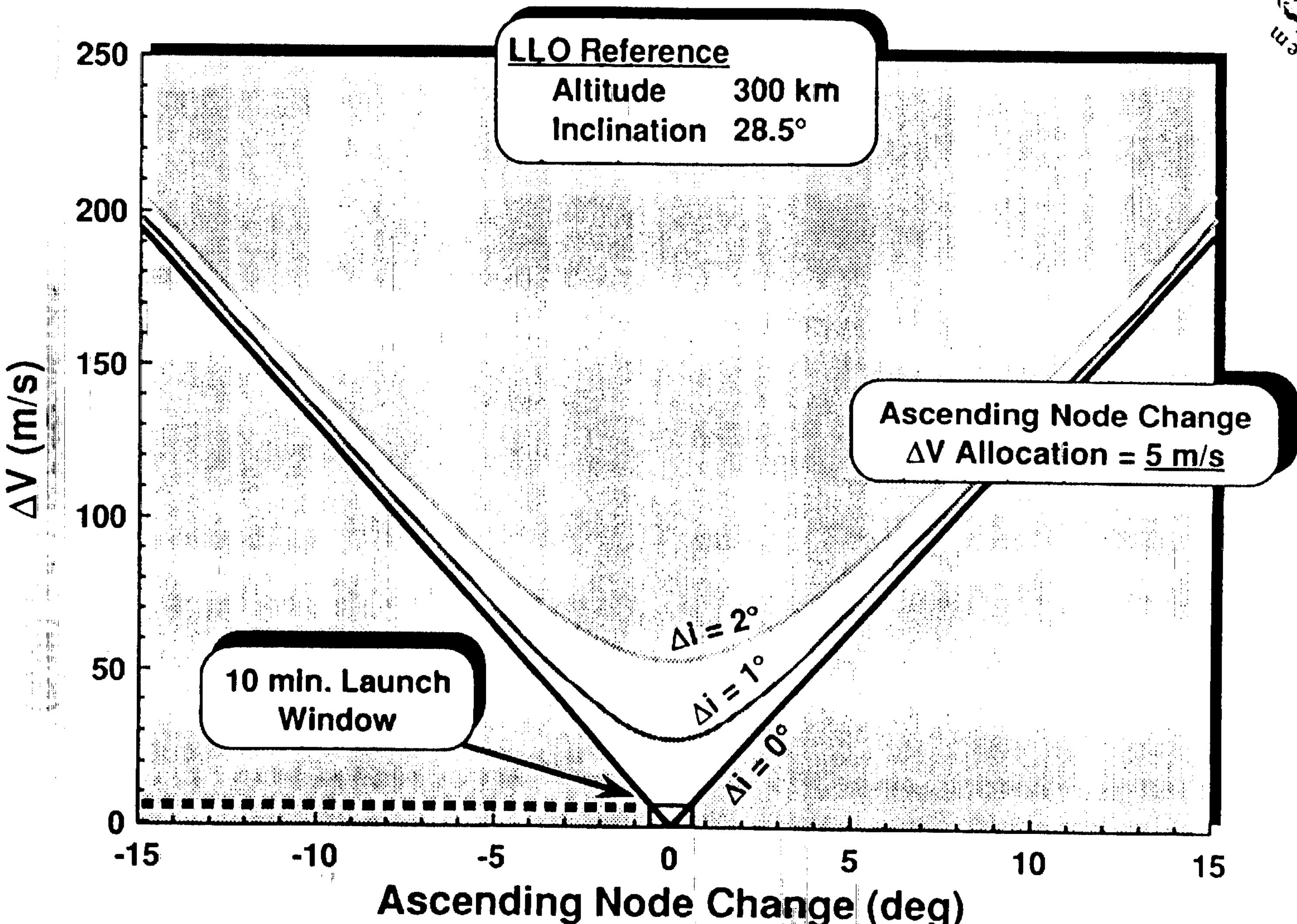
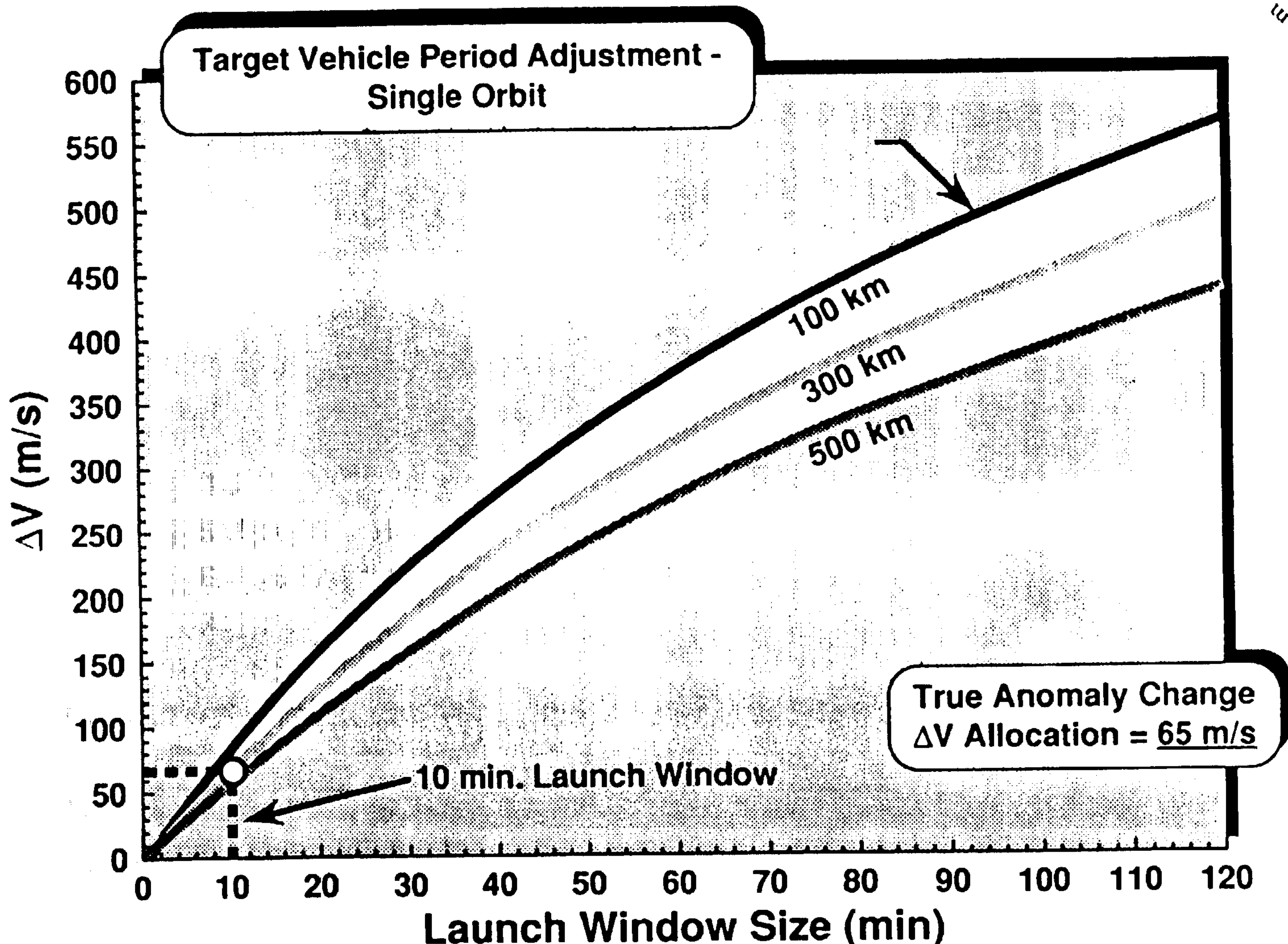
LLO Node Alignment ΔV



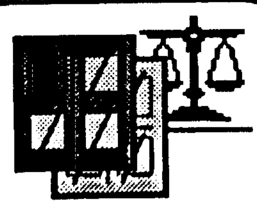
LLO True Anomaly Alignment ΔV



Mission Analysis - Summary

- A Fairly Comprehensive Lunar Mission ΔV Allocation and a Top-Level Mission Timeline Have Been Developed

- An Extensive Lunar Mission Analysis Parametric Database Has Been Generated
    - Earth-to-Orbit
    - Rendezvous & Docking
    - Earth/Lunar Transfer
    - Lunar Descent & Ascent



Initial Concept Downselect

Lori Rauen
(303-977-5760)
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JH910813-10A

Initial Concept Selection Topics

Lunar Transportation System

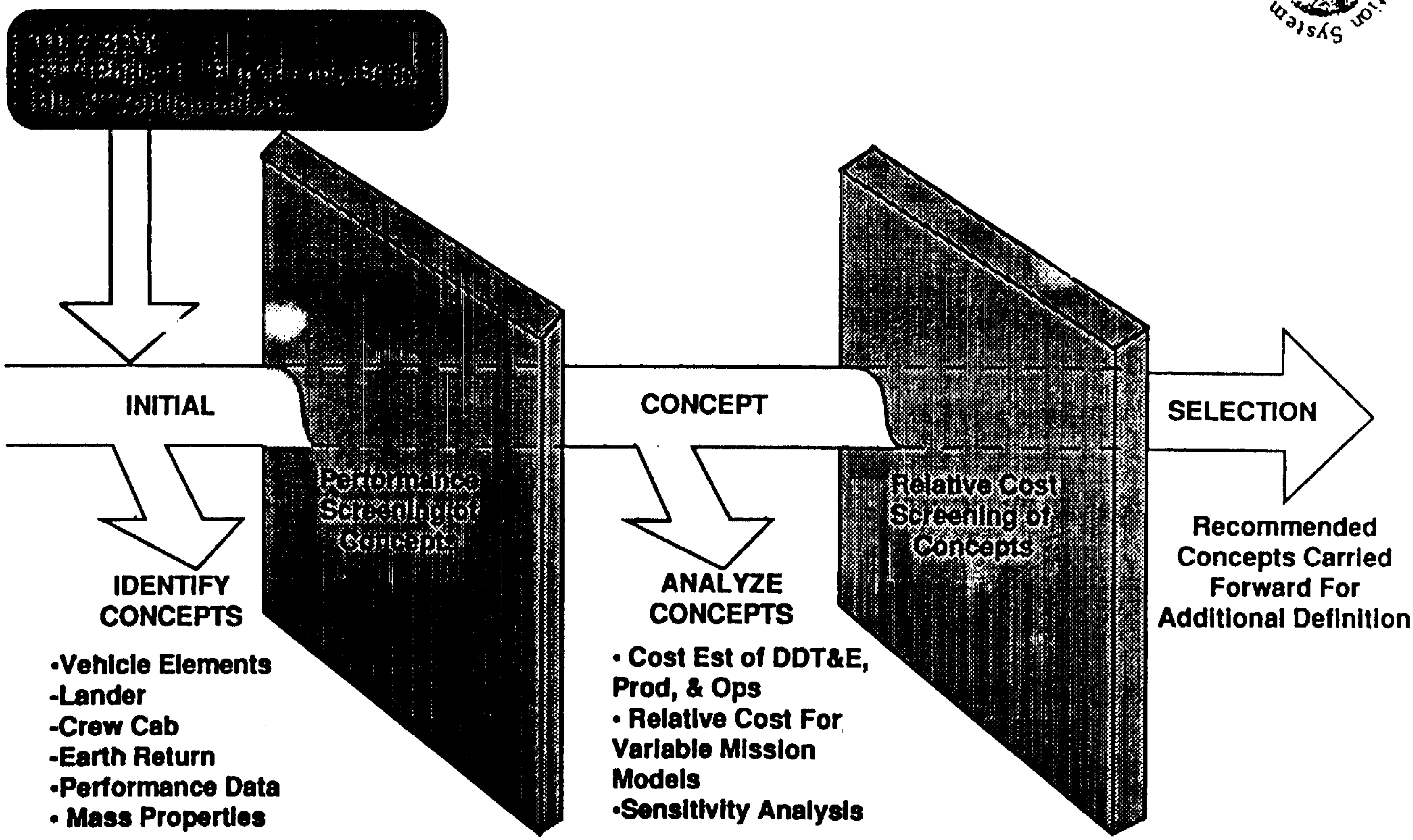
- Goals and Objectives
- Selection Process
- Selection Criteria
- Results
  - Concept Identification
  - ETO Summary
  - Normalized Data Summary
  - Cost Screening of Concepts
  - ID Top Concepts Using HLLV # 1, 2, 3
  - General Cost Analysis Results
  - Recommended Concepts

Two Objectives Identified in Initial Concept Selection Process

Lunar Transportation System

- Systematically Identify and Evaluate LTS Concepts For Rendezvous and Docking Approach to Lunar Transportation
- Identify Top Candidates Associated With Each HLLV Option to Carry Forward for Additional Study and Definition

Selection Process Quickly Identifies Top LTS Concepts

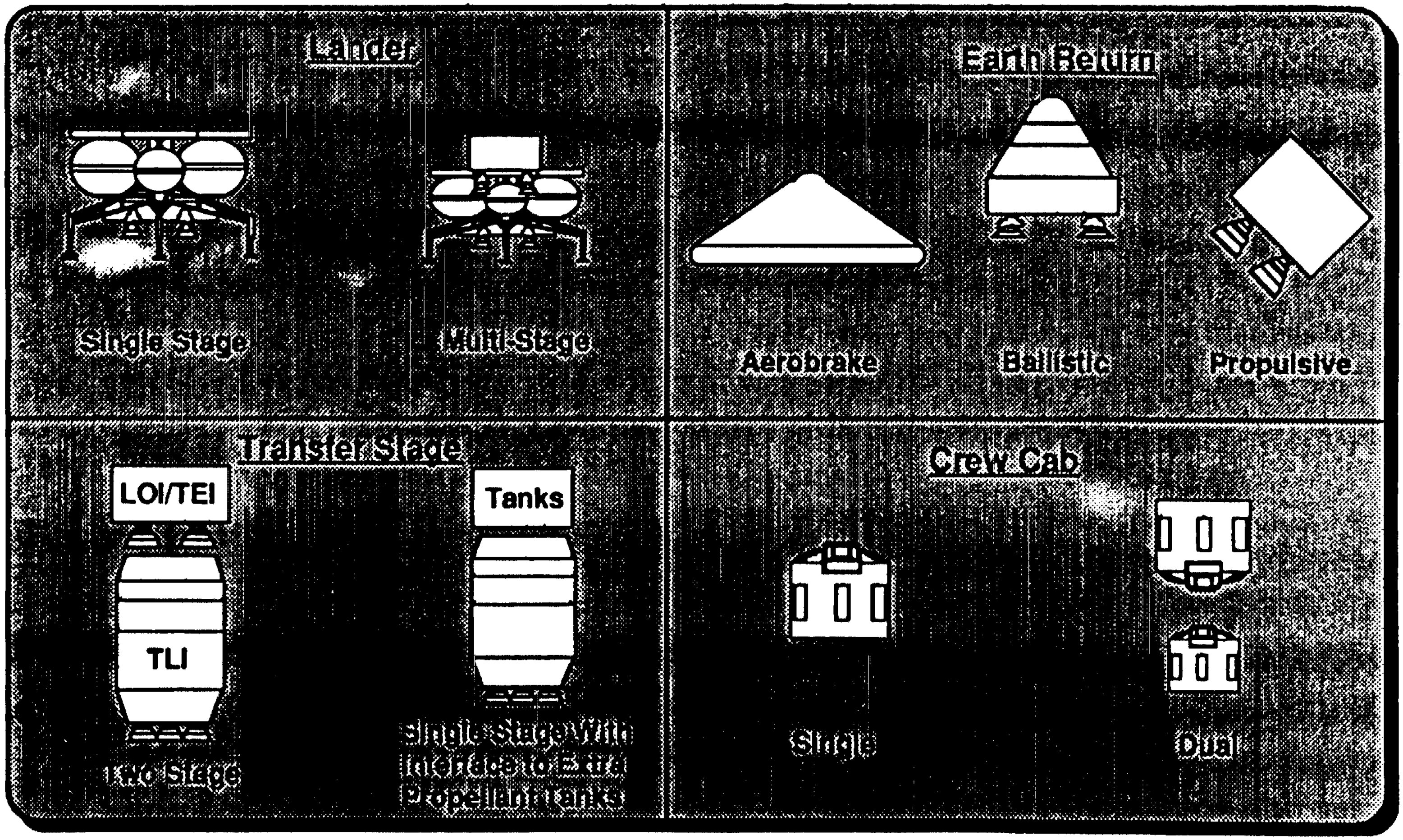


Performance and Cost Were Primary Evaluation Critieria

Performance
- Must Deliver Crew of Four to Lunar Surface

Cost/Unit Mass
- Considers All Measures of Effectiveness in Statement of Work
  - Performance
  - Cost
  - Operations
- Provides Effective Method of Comparison
  - Difficult to Compare Total Cost with Evolving Mission Model
  - Difficult to Compare Performance Capabilities Directly Since Several Concepts Met the Cargo Delivery Requirements, but Each Had a Different Capability
  - Considers All Relevant Cost Elements
- Provides Sensitivity to Variable Mission Model
  - Ratio of Piloted Flights to Cargo Flights Varies from 10:1 to 1:10

Building Block Approach Used In Concept Identification Task



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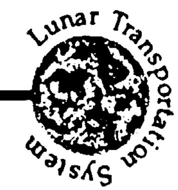
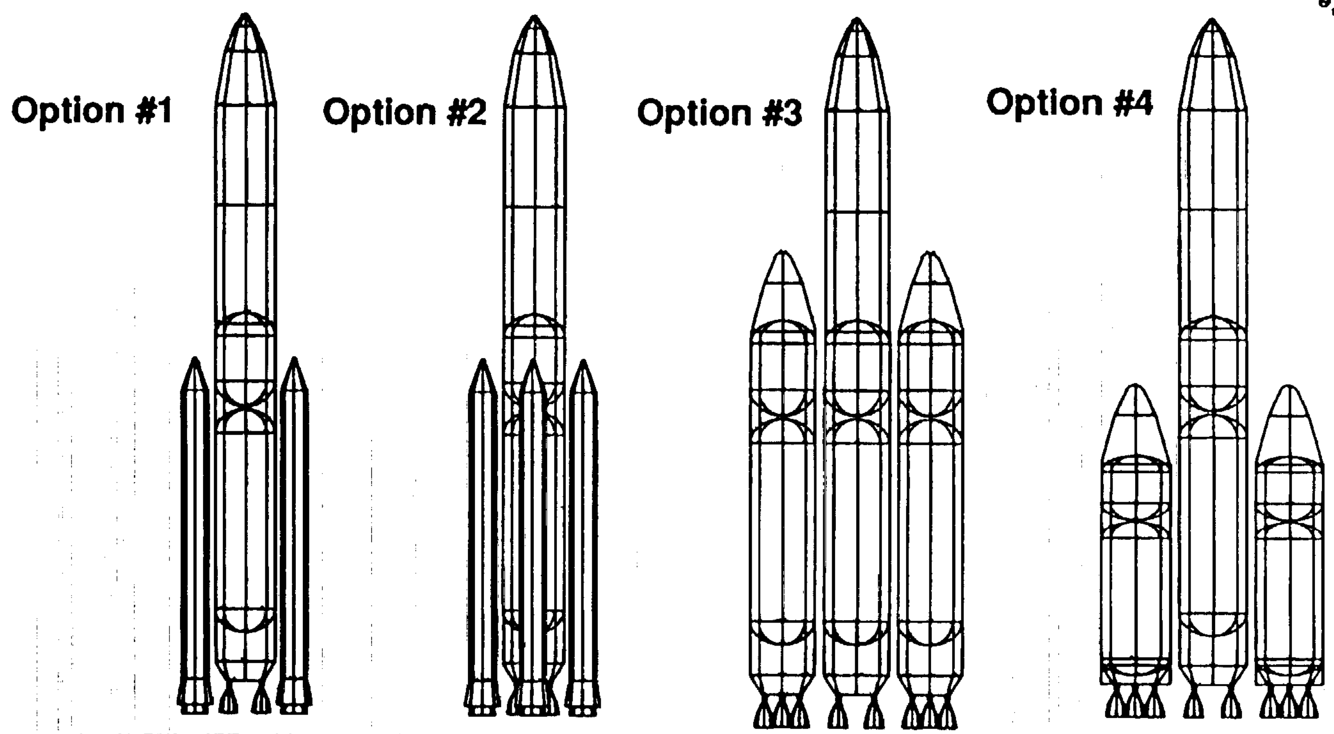
Multiple LTS Concepts Defined for Each HLLV Option





P/L to SSF (t) (Storable Kickstage)	66	105	135	137
P/L to TLI (t) (Single Launch)	35	45	60	62
P/L to TLI (t) (Dual Launch)	73	105	126	127

Provided by MSFC

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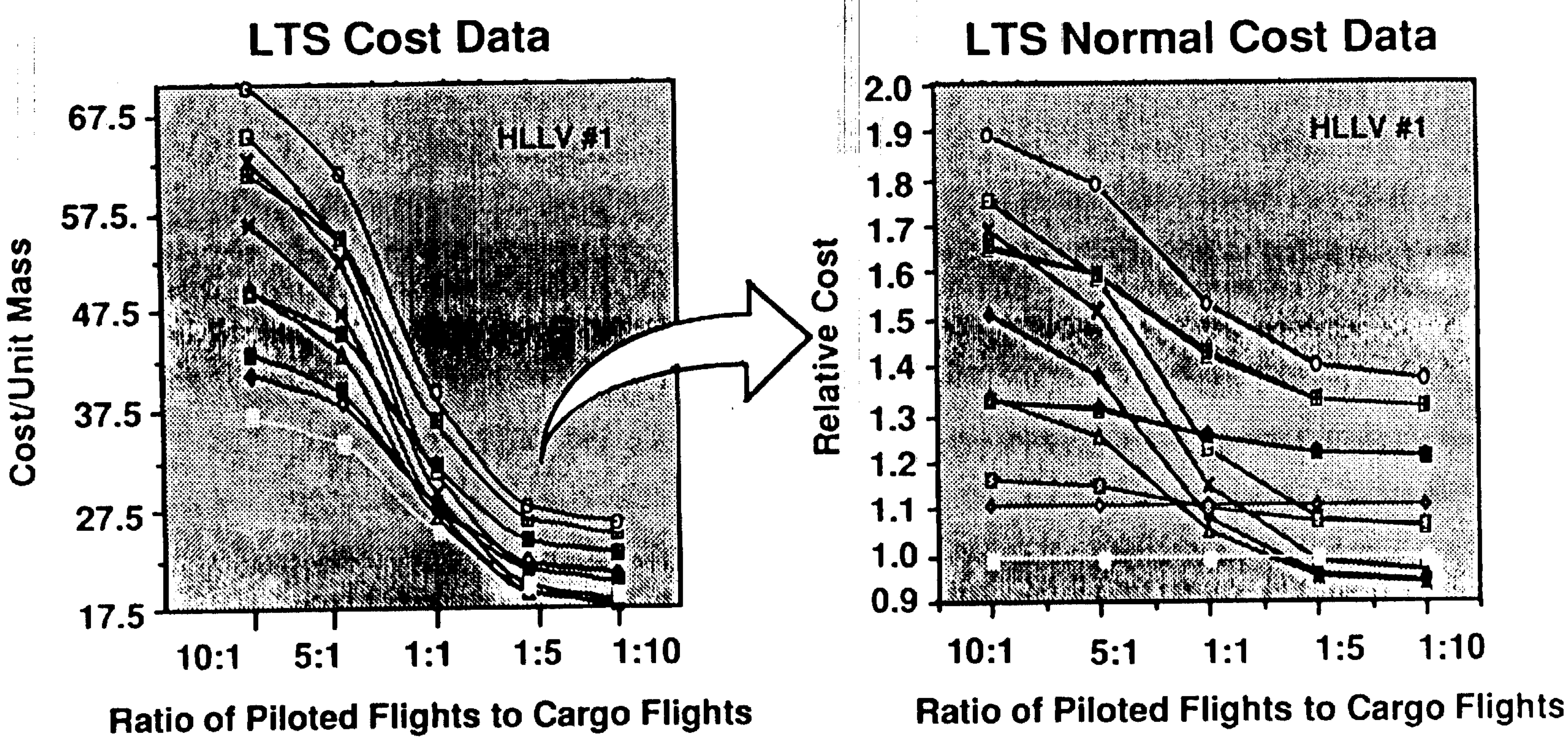
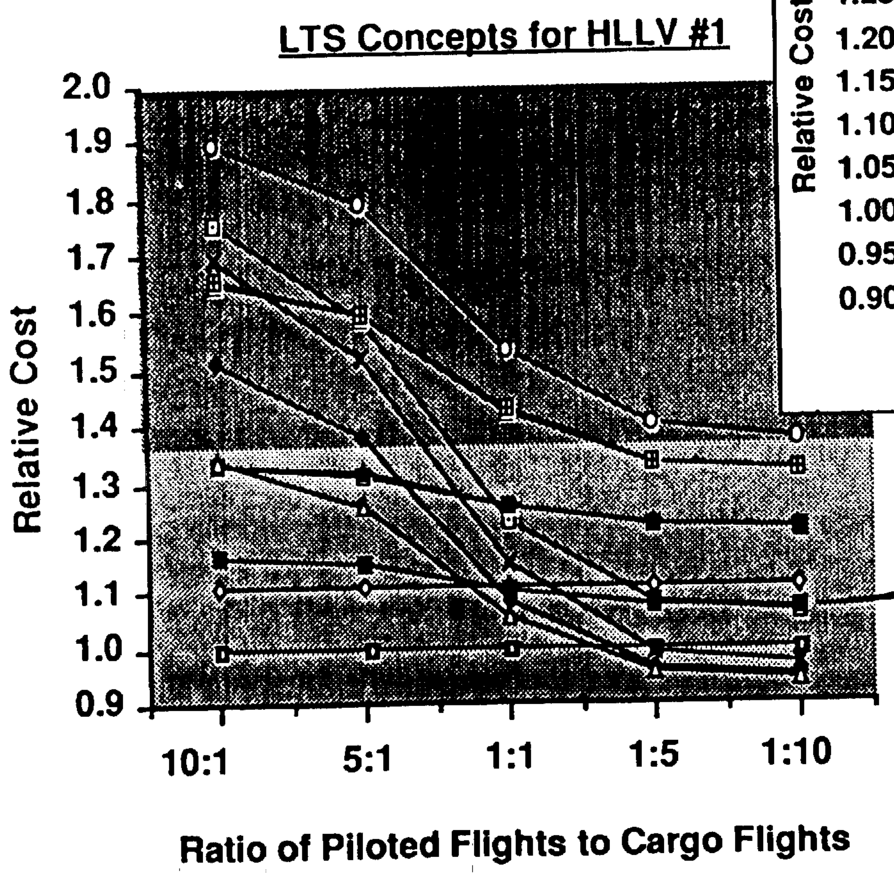
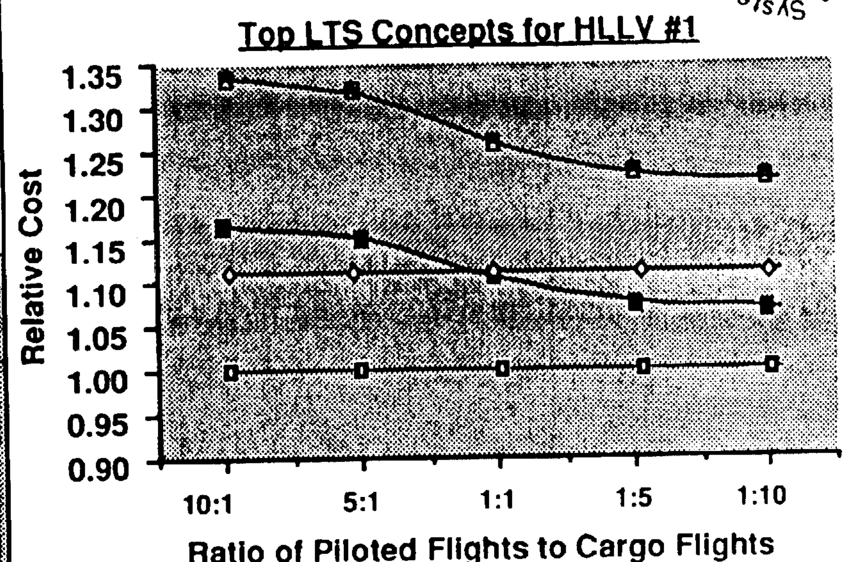
LR910729-06

Normalized Data Facilitates Cost Evaluation



Cost Screening Identifies Top Concepts





Top Candidates Display Low Cost and Small Changes Across the Variable Mission Model

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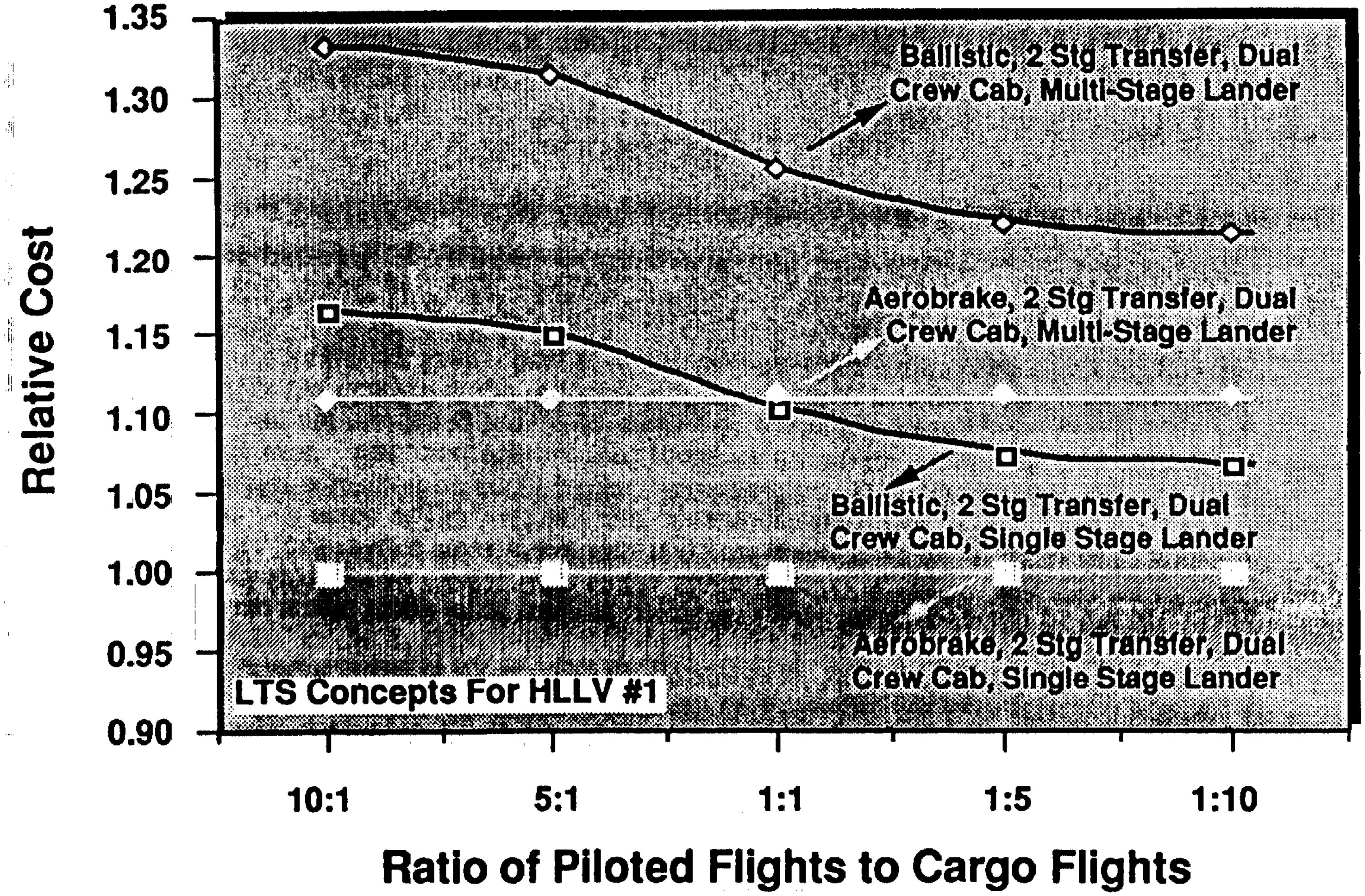
LR910729-08

Top LTS Concepts for HLLV #1 Use Two Stage Lunar Transfer Element



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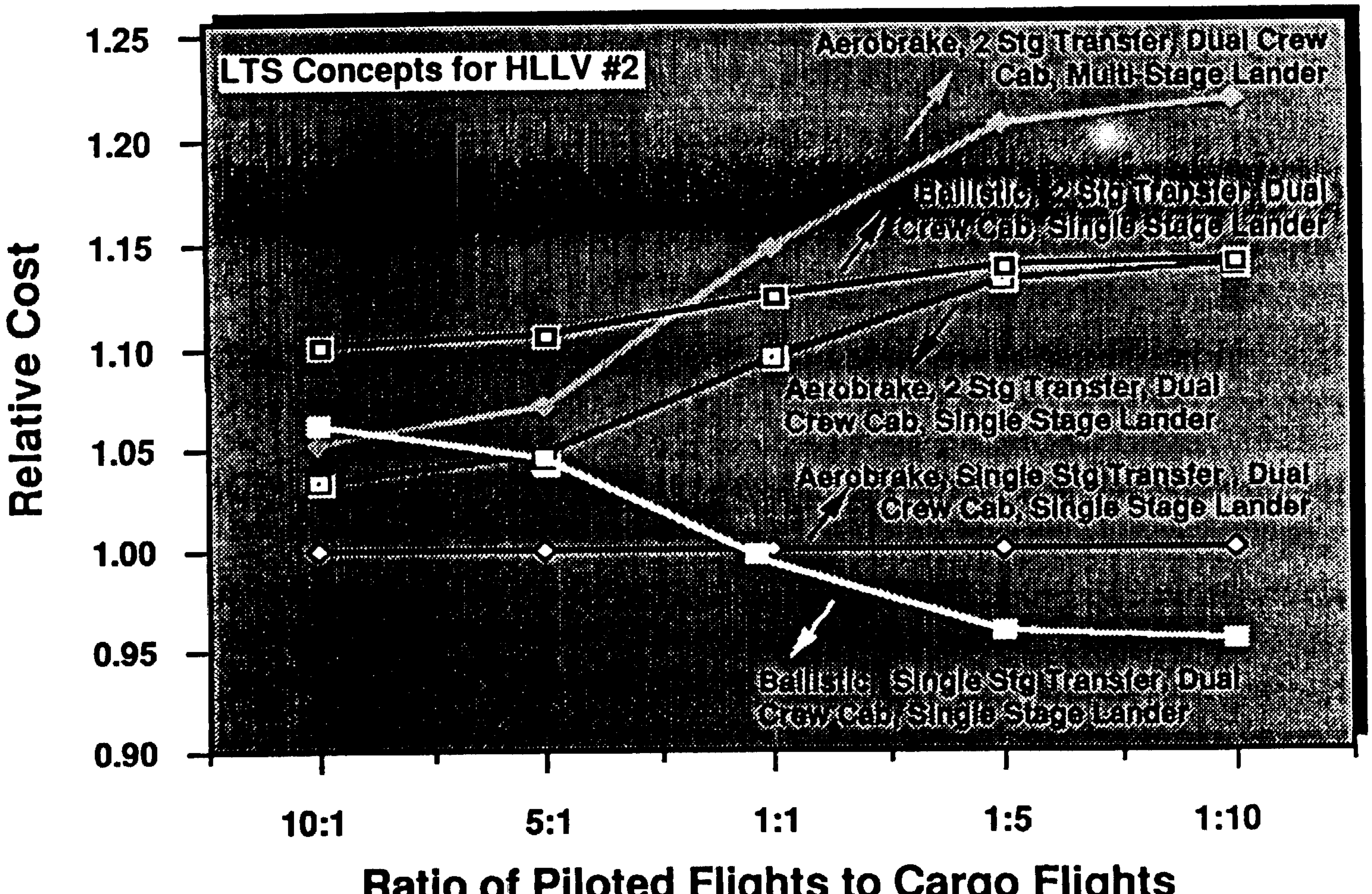
Single Stage Lunar Transfer Concepts Recommended for HLLV #2



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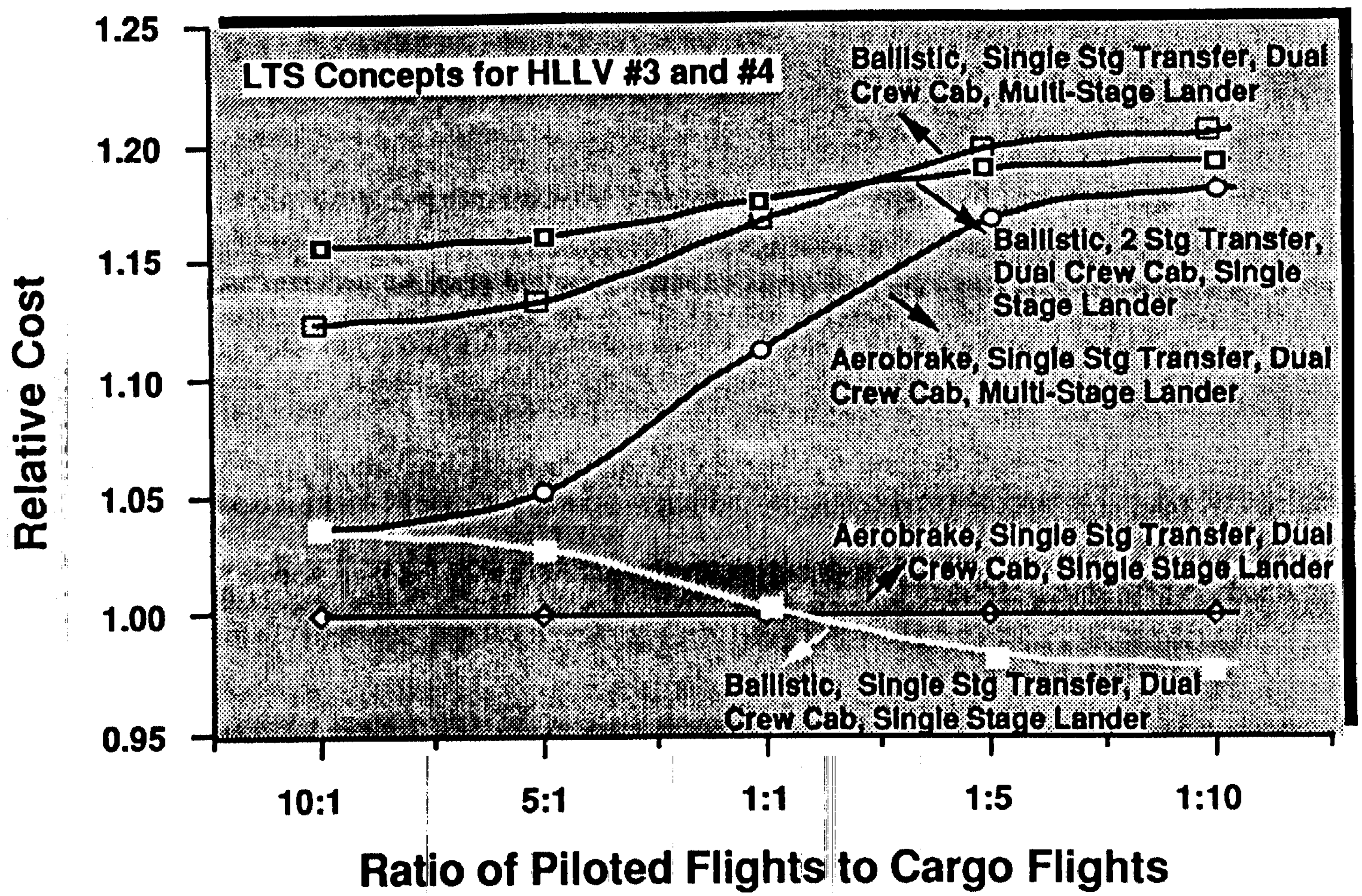
Lowest Relative Cost Concepts Are the Same for HLLV #2 , #3, and #4



Low Cost LTS Options Identified

Lunar Transportation System

Element	Alternatives	Results & Understanding
Earth Return	Aerobrake Ballistic Propulsive	•Propulsive Concept Cost High Due to Significantly Increased Vehicle Size
Crew Cab	Single Cab Dual Cab	•Single Cab Cost Higher Due to Limited Cargo Delivery, Especially For Piloted Flights
Lander	Single Stage Multiple Stage	•Single Stage Lower Cost Due to Decreased Mass & Increased Cargo Delivery
Lunar Transfer	Two Stage – (TLI & LLO Stage) Single Stage – (TLI Stage with Interface to Extra Propellant)	• 2 Stage Transfer Most Cost Effective For HLLV #1 – Single Stage Requires Too Much Propellant Mass and Cannot Always Meet Minimum Delivery Requirement • Single Stage Transfer More Cost Effective with HLLV #2, 3, 4 Because the Extra Propellant Costs Less Than the LOI/TEI Stage

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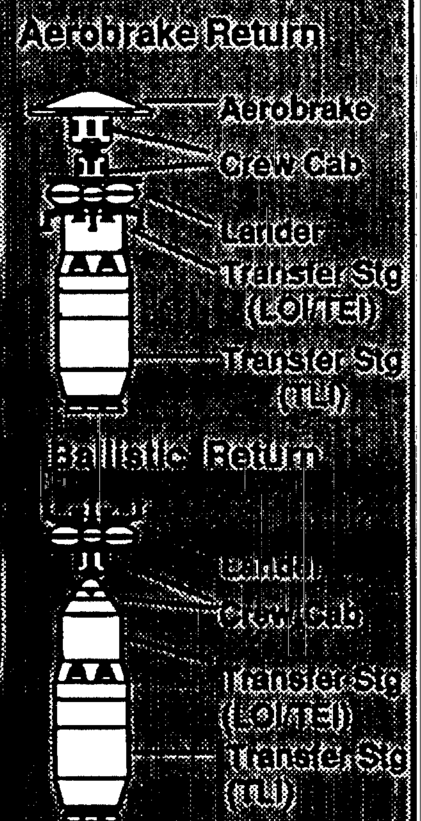
Dual Crew Cab/Single Stage Lander Concepts Recommended for Further Study

Two Stage Lunar Transfer Recommended for HLLV #1

P/L Capability

Piloted – 11 t
Cargo – 22 t



P/L Capability

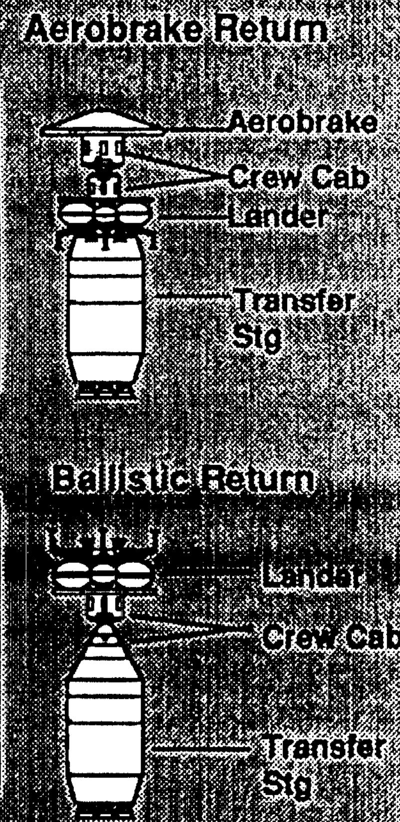
Piloted – 9 t
Cargo – 20 t

Single Stage Lunar Transfer Recommended for HLLV #2,3,4

P/L Capability

With HLLV #2:
Piloted – 20 t
Cargo – 35 t

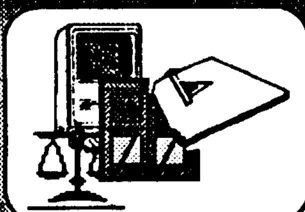
With HLLV #3/4:
Piloted – 31 t
Cargo – 48 t



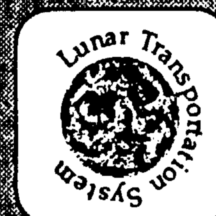
P/L Capability

With HLLV #2:
Piloted – 18 t
Cargo – 36 t

With HLLV #3/4:
Piloted – 29 t
Cargo – 48 t



Final Concept Selection & Definition



Bob Spencer
(303-971-4530)

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JH910813-11A

Final Concept Selection & Definition - Topics

- Initial Downselect Review
- Aerobrake vs Ballistic Trade
  - Performance
  - Risk & Operational Assessment
- Detailed Concept Definition
  - HLLV Top Candidates & Rational
  - Sequential Mass Breakdown (LEO, LLO, Descent, Surface, LLO, Return)
  - Detailed Configuration Layout
    - TLI Stage
    - LOI Stage
    - Lander Stage
    - Top Level Layout
- Summary

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Aerobrake vs Ballistic Performance Comparison

Lunar Transportation System

- 25 ft Payload Diameter & No On Orbit Servicing
  - A 25 ft Diameter Rigid Aerobrake was baselined as a result of STV Phase I Work
  - A One-Piece Rigid Aerobrake Eliminates All On-Orbit Assembly and Checkout Associated with a Flexible or Multi-piece Rigid Brake
  - Launch Vehicle Payload Diameter is 25 ft (7.62 m) Maximum
- Aerobrake Wake Impingement Angle
  - 22° Wake Angle Generates No Impingement (Phase I STV Angle)
  - 33° Wake Angle Limits the Trans. Cab - Excr. Cab Interface Diameter to 8 ft (2.44 m) (AIAA Paper 91-1371 "On the Computation of Near Wake, Aerobrake Flowfields, NASA Langley)
- STV Phase I Ballistic Coefficient vs Aerobrake Capability
  - Desired W/CdA=10-15 lb/ft^2
  - With Customer Supplied Transfer Cab Mass + Aerobrake Calculated @ 15% of Braked Mass, W/CdA Becomes ≈ 22 lb/ft^2
- Ballistic Coefficient of 22 lb/ft^2 and Phase I Data on FRCI Tile
  - Multi Flight is Not an Option - Surface Temperature above Range
  - Single Flight is Border Line without Geometry Effects
  - Geometry Effects Increase the Heat Flux by ≈ 30% and Surface Temp Increases ≈ 10%
- Possible Options
  - Shuttle Carbon-Carbon Material (Heavy)
  - Multi-Pass Aerobraking (Duration too Long, Increases Consumable Mass)
  - Advanced Material Development (Costly)
  - Ablative Surface (Heavy and Expendable Brake)
  - Ablative Direct Return (Heavier, Possible Re-use)

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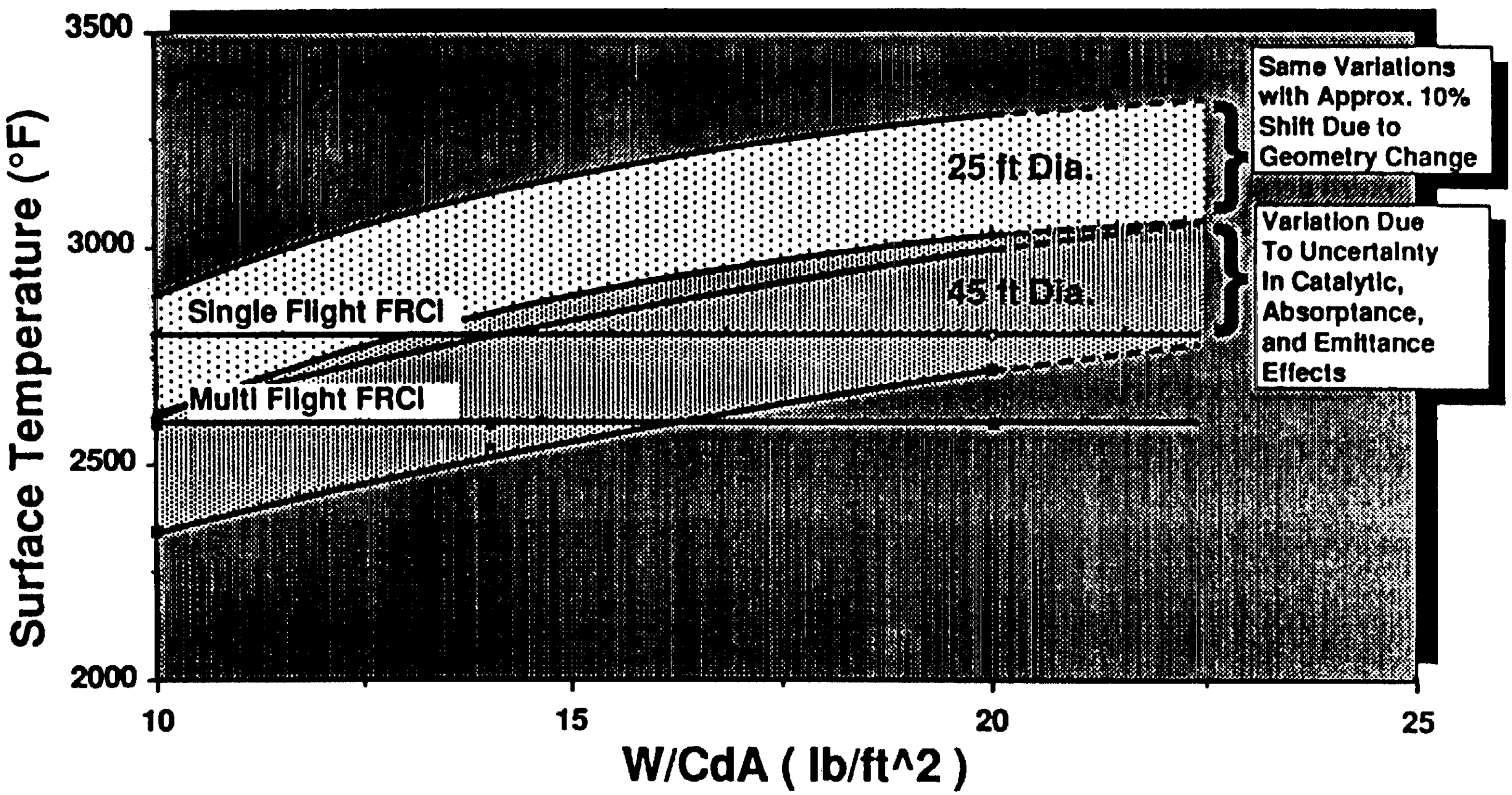
25 ft Dia. Brake Surface Temperature Sensitivities Prove Too High For Application



- Brake Radius / Nose Radius = 1.0

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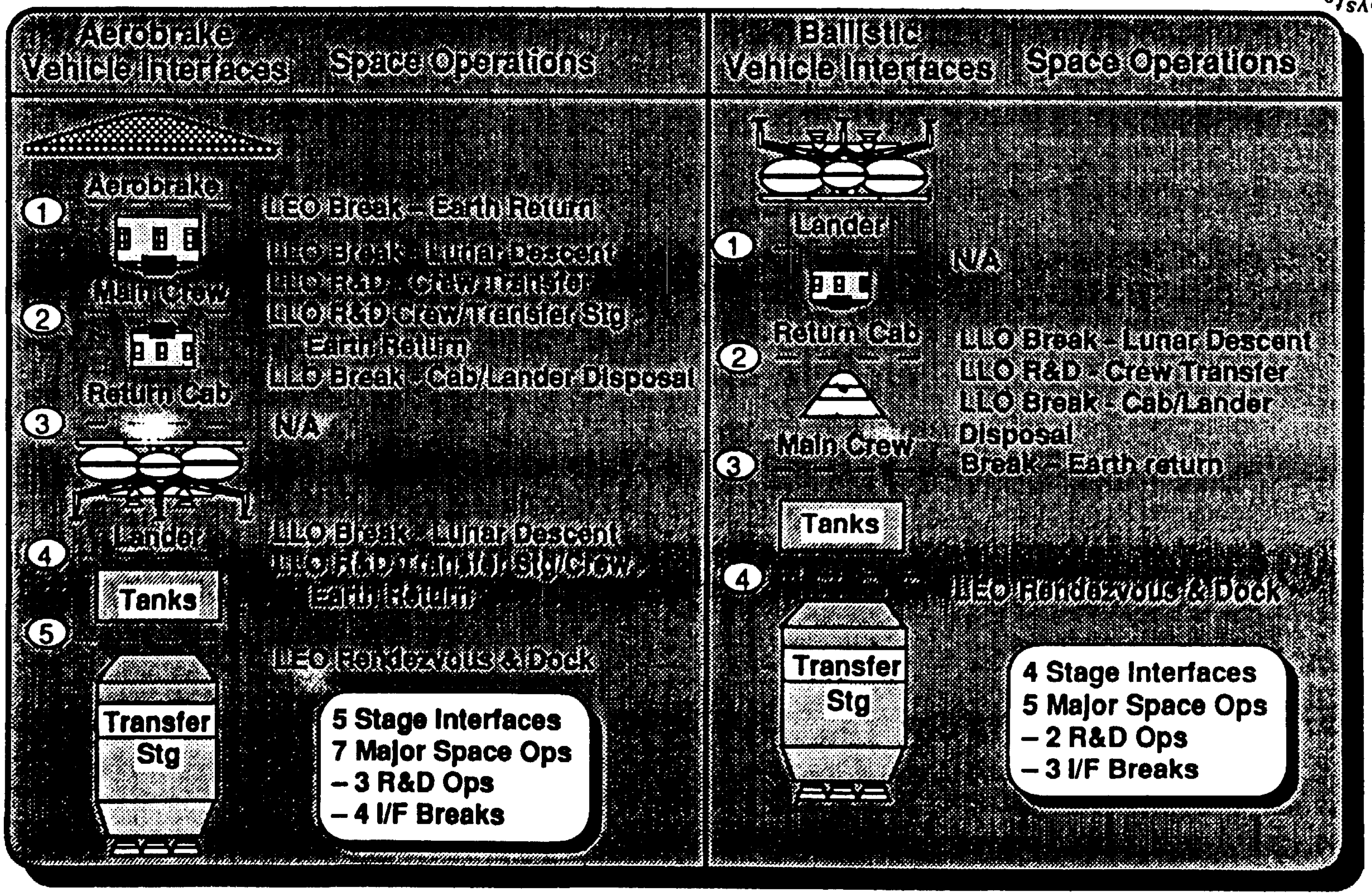
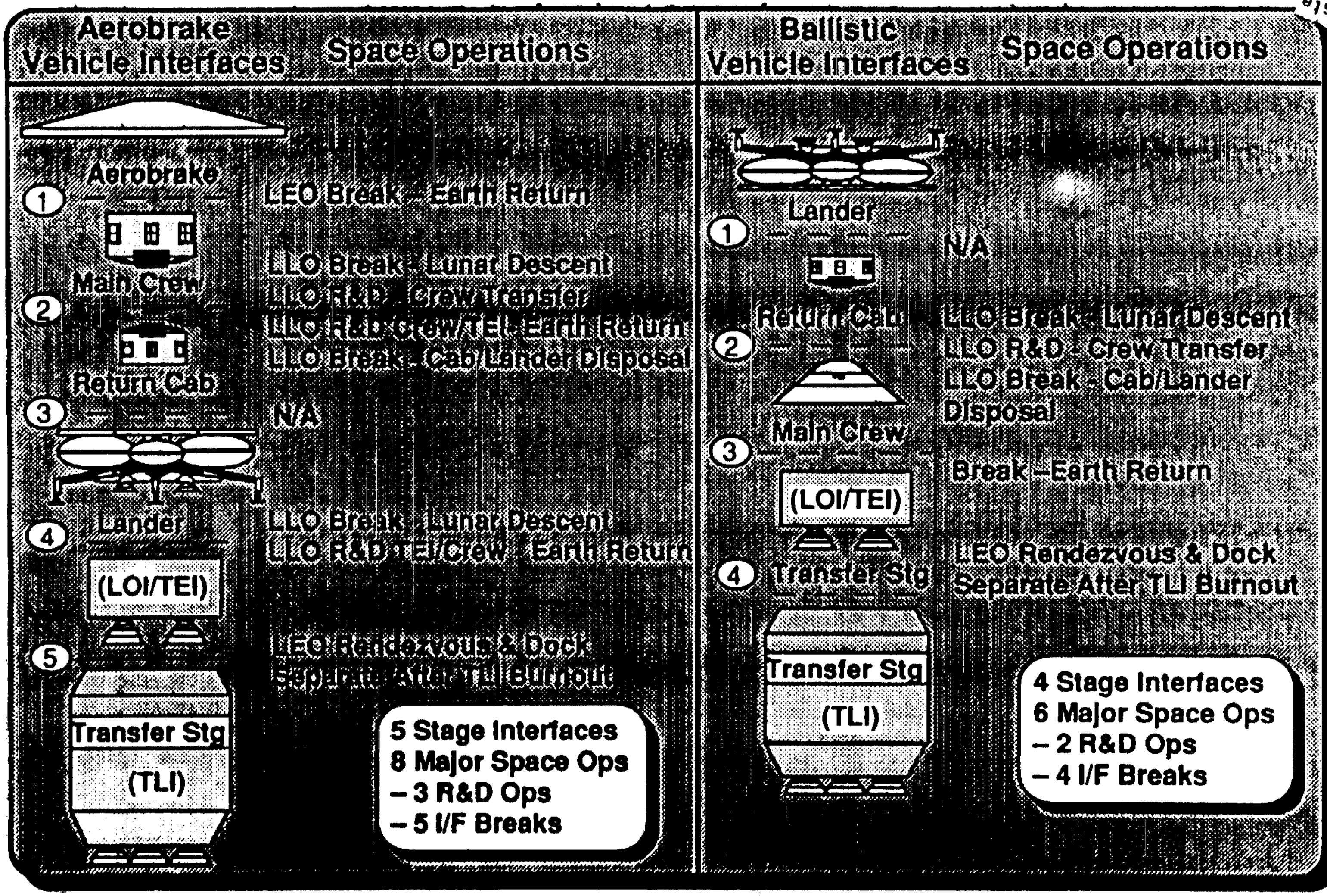
Ballistic Configuration Reduces Interfaces & Space Ops - Single Stage Transfer



Ballistic Config. Reduces Interfaces & Space Ops - Two Stage Transfer



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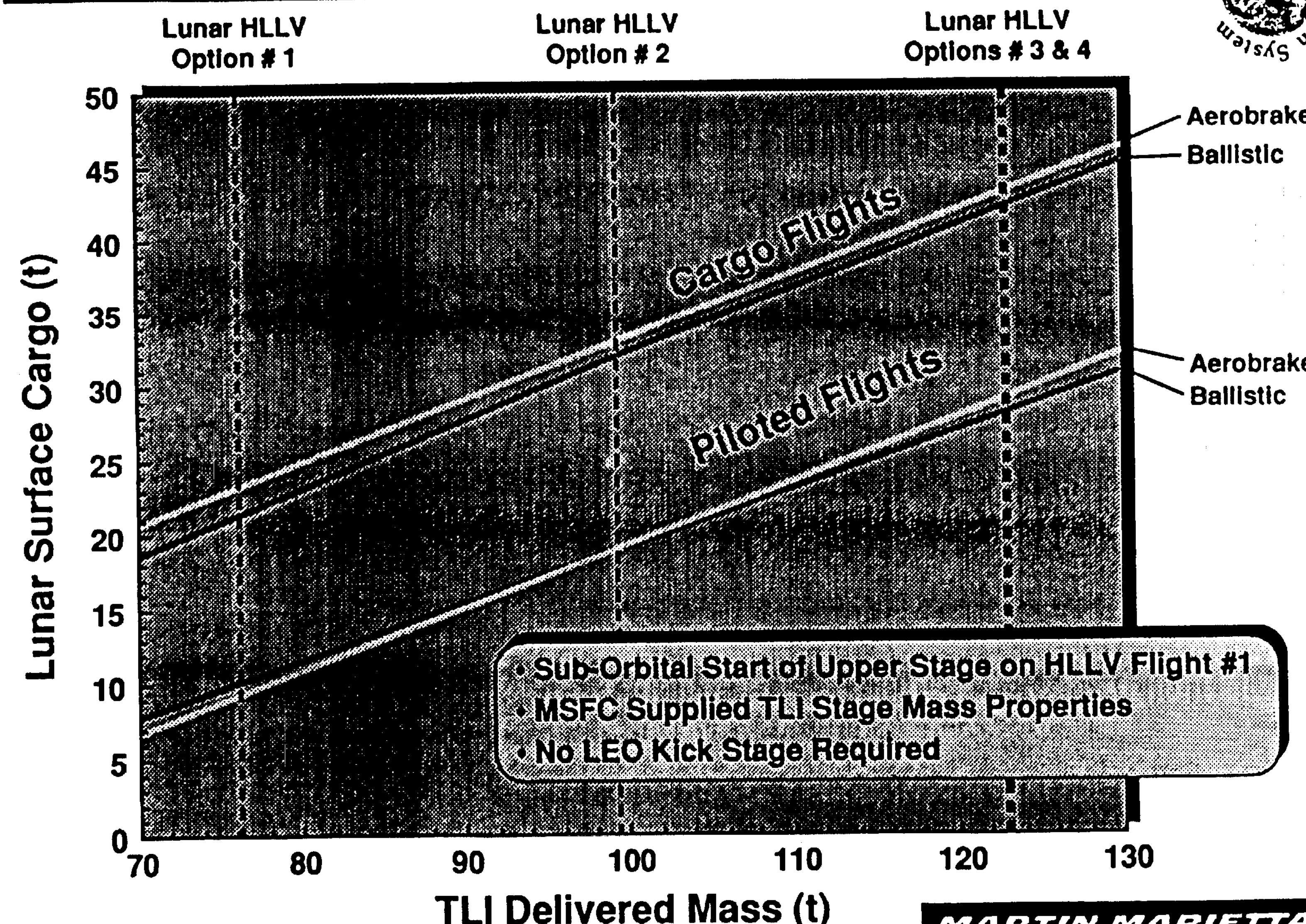
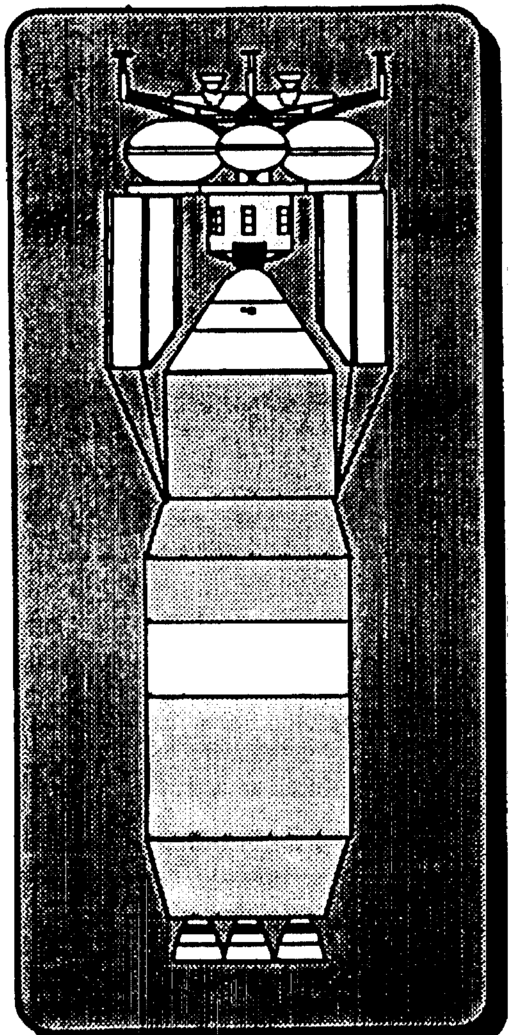
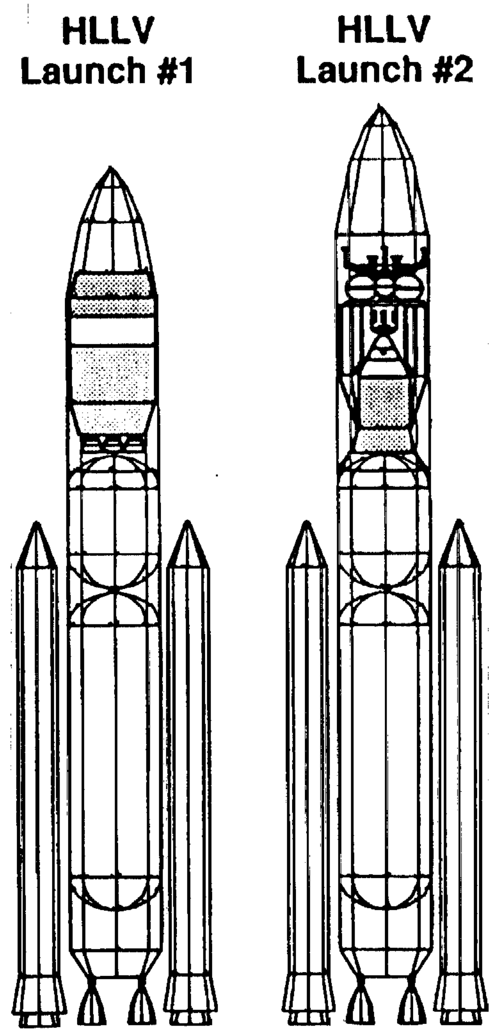
Two Stage Transfer, Single Stage Lander - Lunar Cargo vs TLI Delivered Mass - (Preliminary Data)



Top Candidate For HLLV Option #1



Two Stage Transfer, Single Stage Lander Vehicle



Vehicle Summary

Performance:	
Lunar Surface	
› Piloted	5.0 t
› Cargo	23.1 t
Hardware Reuse	Optional
Advl Tech.	Minimal

Decision Factors

(Compared to Aerobrake Vehicle)

Space Ops.	2 Fewer Ops
PayLoad Dia	7.62 m
On Orbit Serv.	None

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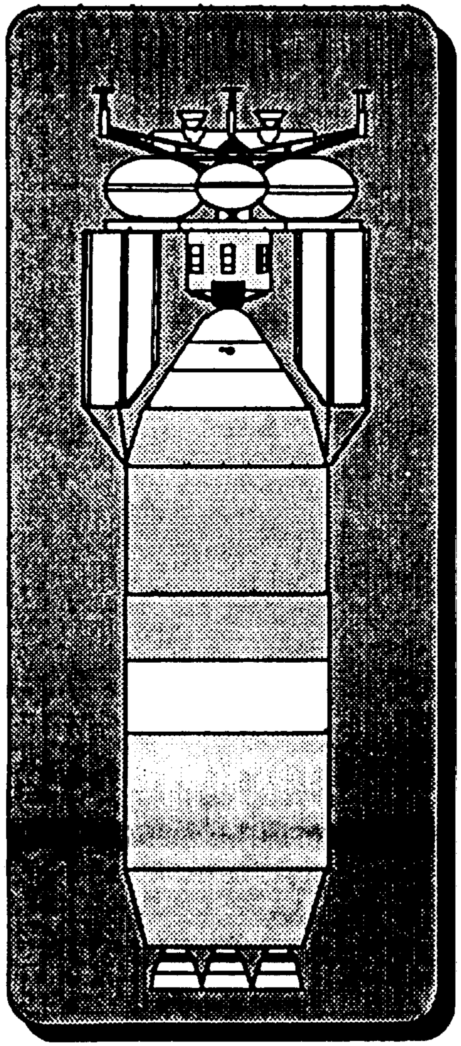
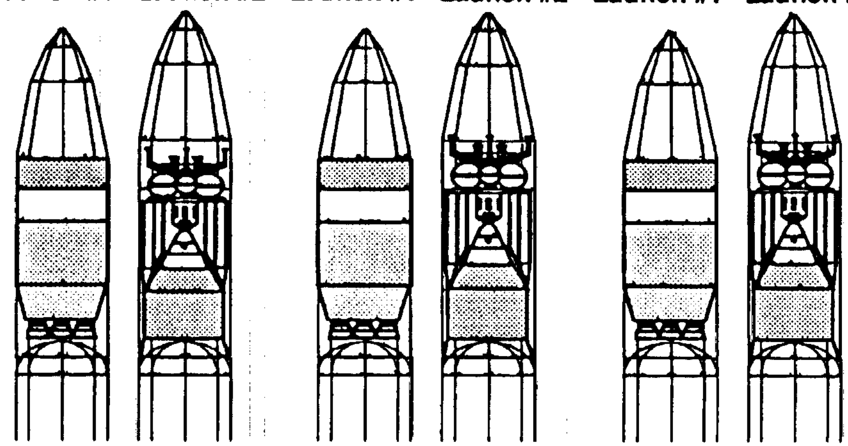
Top Candidate For HLLV # 2,3, &4



Single Stage Transfer, Single Stage Lander



Vehicle Summary

	HLLV #2	HLLV #3	HLLV #4
Performance: Lunar Surface • Piloted • Cargo	 12.3 t 38.1 t	 20.4 t 45.7 t	 20.5 t 45.8 t
Hardware Reuse	Optional	Optional	Optional
Adv. Tech.	Minimal	Minimal	Minimal

Decision Factors
(Compared to Aerobrake Vehicle)

Space Ops.	2 Fewer Ops
Payload Dia.	7.62 m
On Orbit Serv.	None

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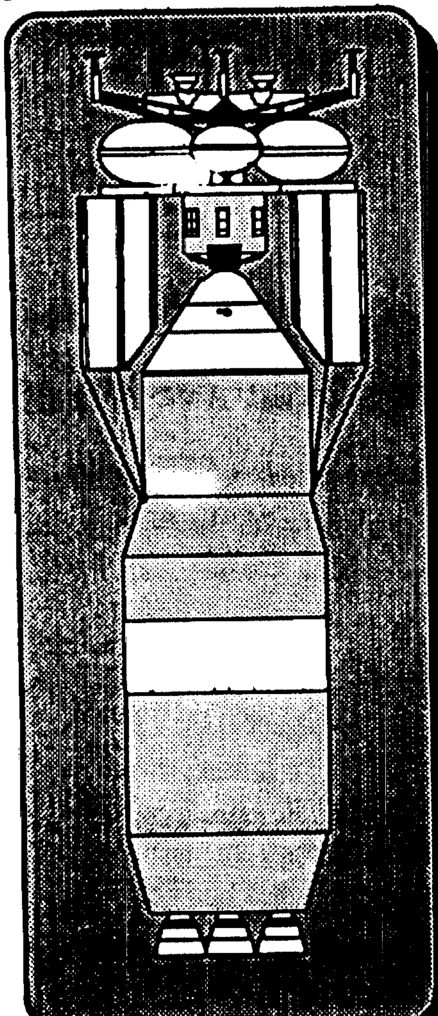
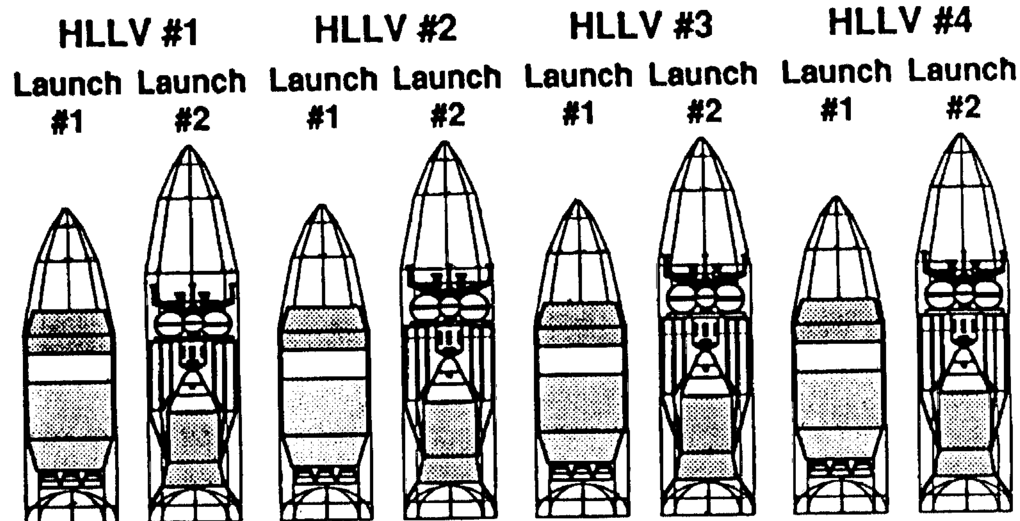
RS910729-03A

Most Consistent Top Candidate Over All HLLV Options



Two Stage Transfer, Single Stage Lander



HLLV Option	#1	#2	#3	#4
Stack Mass	288.0 t	313.0 t	330.4 t	330.6 t
Performance				
Piloted	5.0 t	18.8 t	26.6 t	26.7 t
Cargo	23.1 t	38.9 t	47.6 t	47.8 t
Launch Limit	#2	#1&#2	#1	#1
Cost Comp				

Consistently Low Relative Cost For All HLLV Options

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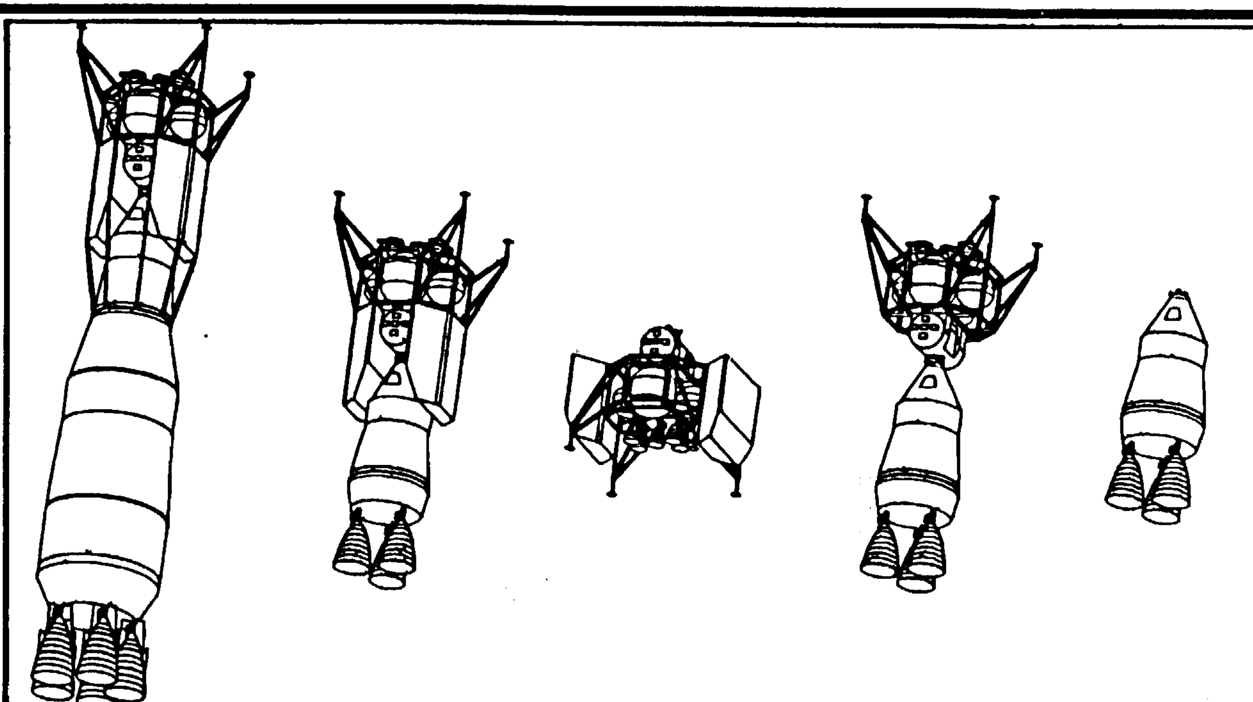
RS910730-01A

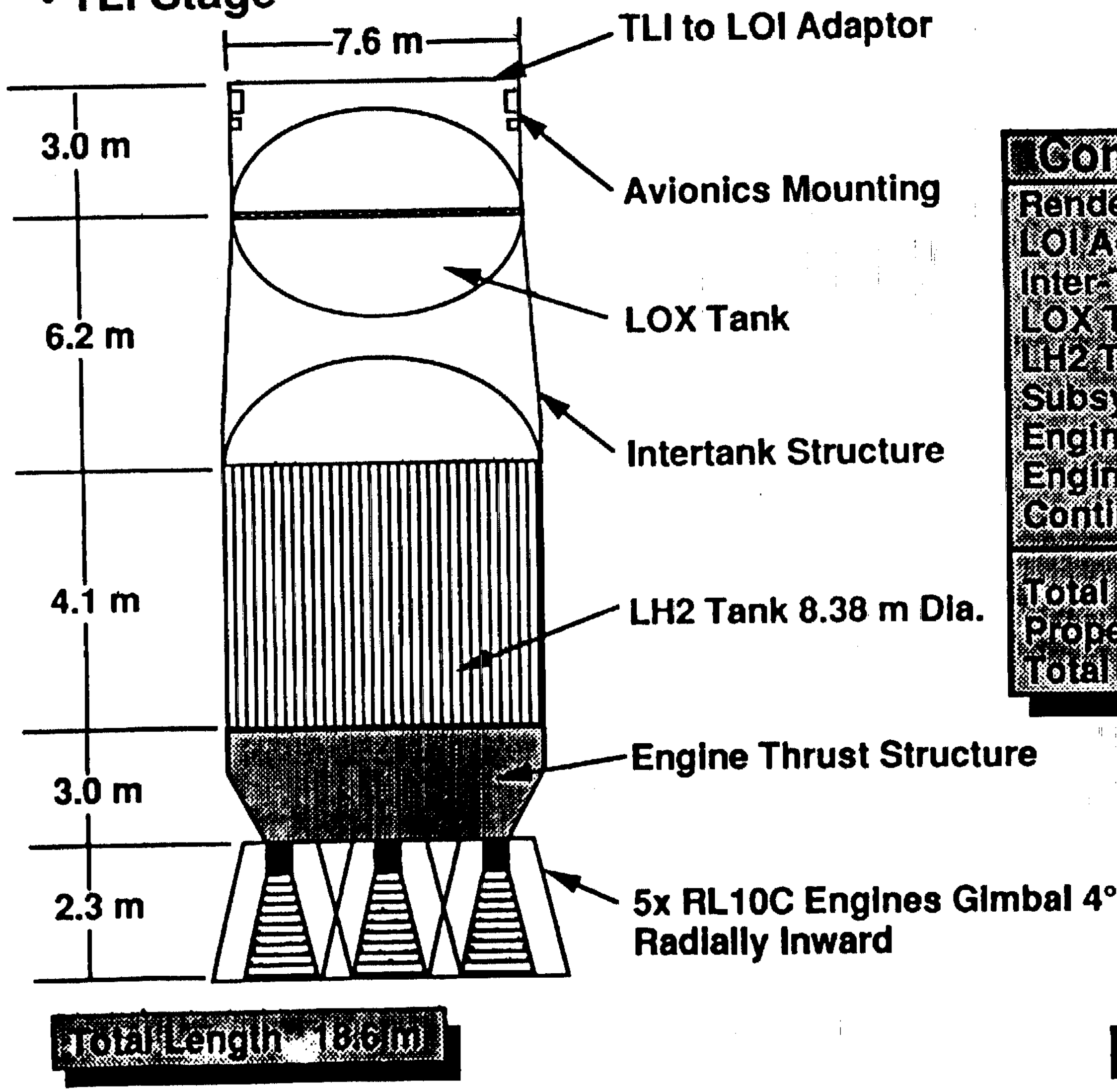
Two Stage Transfer Single Stage Lander - Mass Properties @ Diff. Points in The Mission



Element \ Position	Pre - TLI LEO (t)	Pre - LOI LLO (t)	Pre - Unload Surface (t)	Pre - TEI LLO (t)	Pre - Return Return (t)
Lander Stage	6.1	6.1	6.1	6.1	-
Lander Prop	21.0	20.9	7.5	.3	-
Excr. Cab	2.5	2.5	2.5	2.5	-
Transfer Cab	8.6	8.6		8.6	8.6
LOI/TEI Stage	8.4	5.6		5.6	5.6
LOI/TEI Prop	19.6	19.4		4.0	.4
TLI Stage	14.9				
TLI Prop	[illegible]				
Payload	6.0	6.0	6.0		
Total	[illegible]	[illegible]	21.1	27.1	14.6

Common TLI Stage - 210 t Propellant

- TLI Stage



Component	t
Rendezvous & Dock	0.14
LOI Adaptor	0.67
Inter-Tank	1.85
LOX Tank	1.01
LH2 Tank	3.94
Subsystems	1.89
Engine Strct	1.63
Engines	1.82
Contingency (15%)	1.94
Total Dry	14.89
Propellant	210.0
Total Wet	224.9

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RS910711-02B

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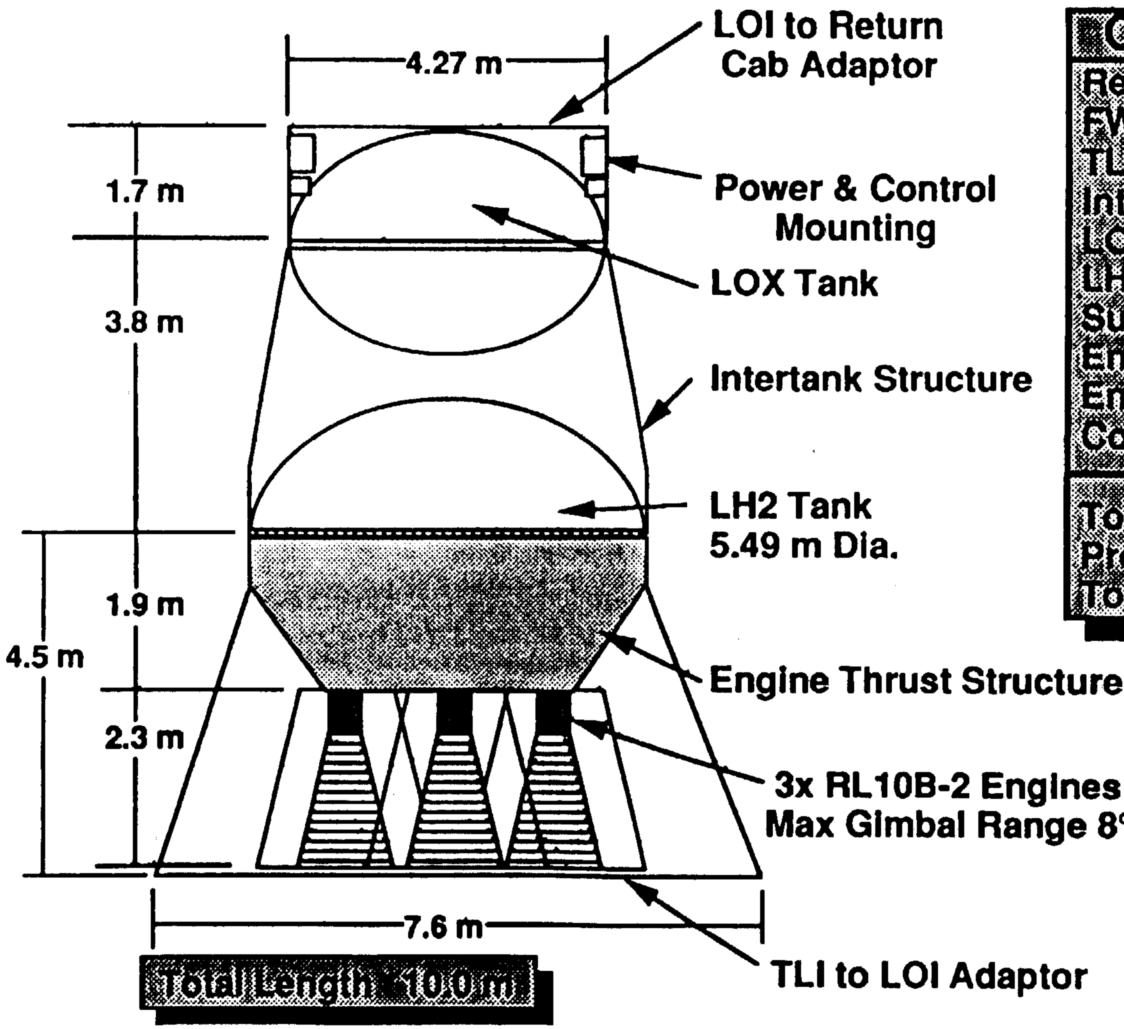
HLLV Opt #1 LOI Stage - 23 t Propellant

- LOI Stage



Component	t
Rendezvous & Dock	0.14
FWD Adaptor	0.10
TLI Adaptor	1.80
Inter-Tank	0.19
LOX Tank	0.19
LH2 Tank	0.44
Subsystems	2.83
Engine Strct	0.47
Engines	0.80
Contingency (15%)	0.67
Total Dry	7.63
Propellant	22.67
Total Wet	30.3

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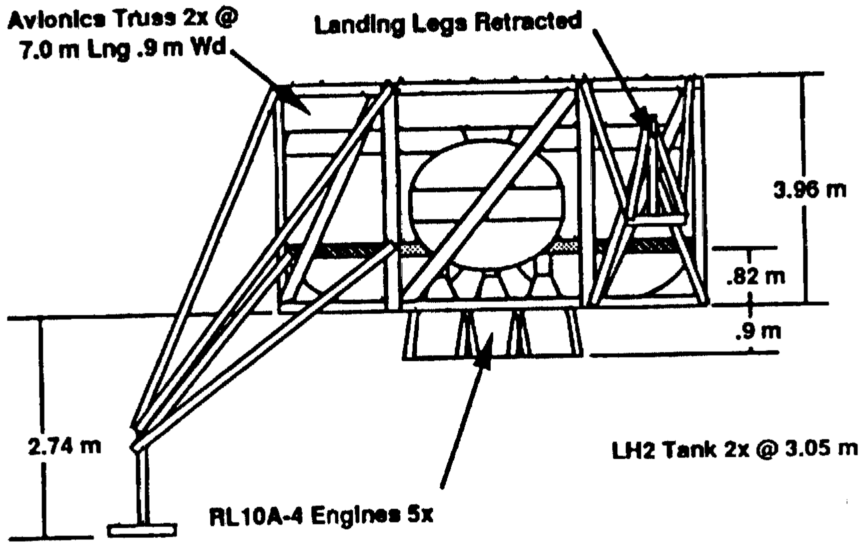
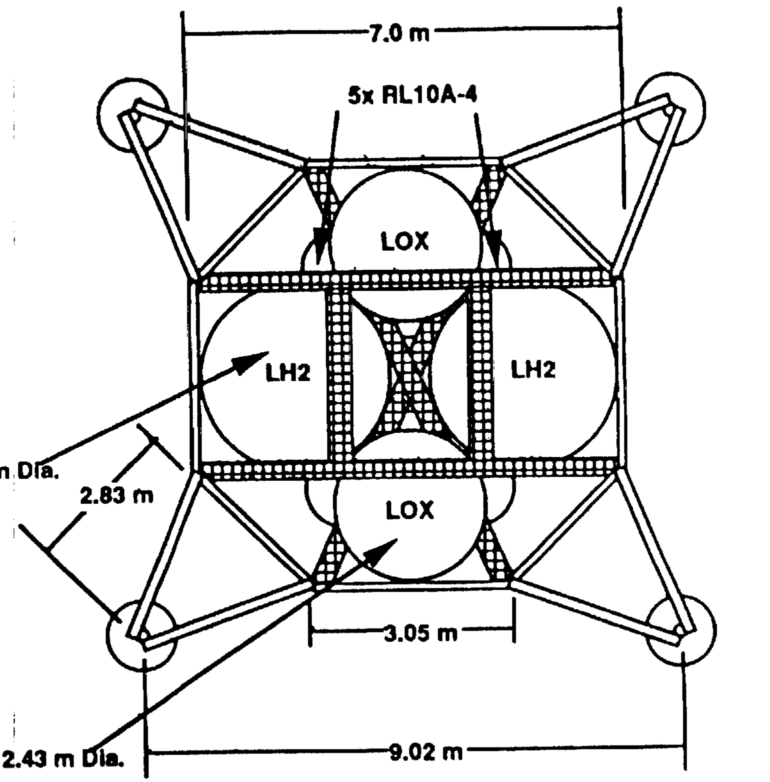
HLLV Opt #1 Lander Stage





Component	t
Structure	1.99
Tanks	0.29
Subsystems	2.44
Engines	0.83
Contingency (15%)	0.51
Total Dry	6.06
Propellant	19.59
Total Wet	25.65

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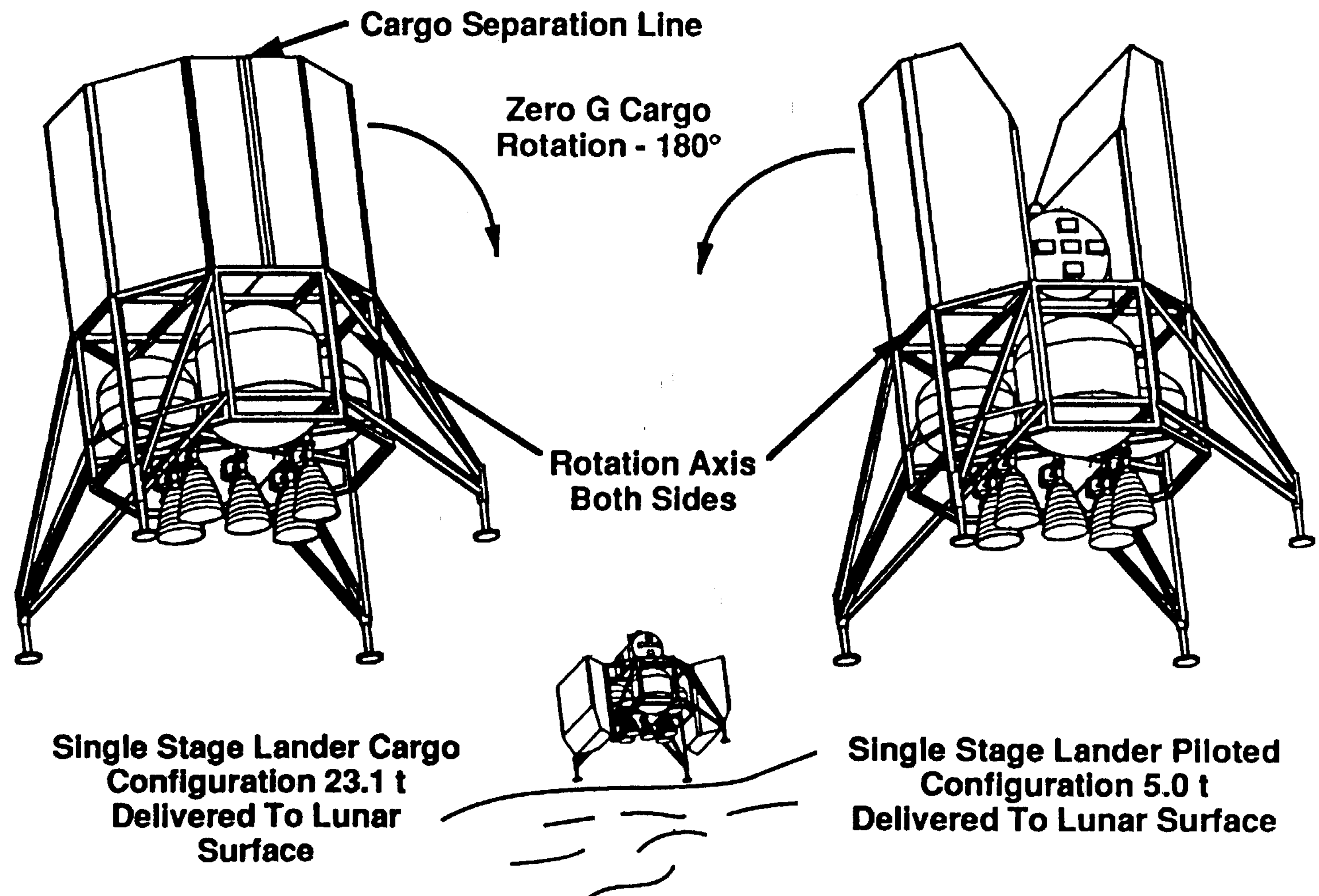
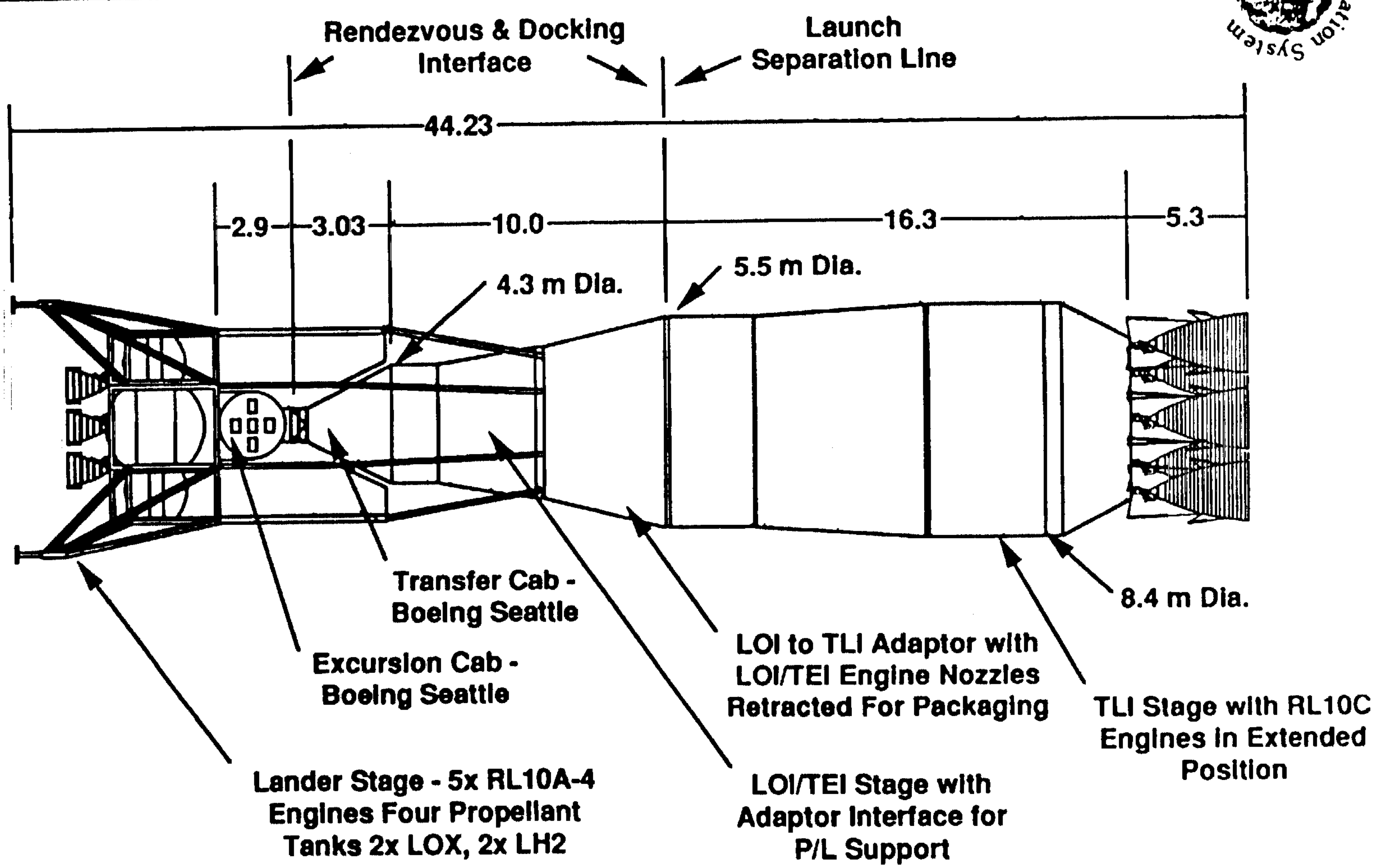
RS910710-01B

HLLV #1 Lander Stage - Piloted & Cargo Configurations

- **Landers Shown in LLO Prior To Cargo Rotation and Descent**



Dimensioned Layout of Two Stage Transfer, Single Stage Lander Configuration



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RS910816-01A

Final Concept Selection & Definition Summary

Lunar Transportation System

- Direct Earth Return (Ballistic) is Favored Over LEO Return (Aerobrake) For This Study Only, Due To Lower Risk, Fewer Space Ops, Maximum Payload Dia. 25 ft (7.26 m), and no On Orbit Assembly

- Selected Concepts For HLLV#1 (Two Stage Transfer, Single Stage Lander), HLLV#2,3, &4 (Single Stage Transfer, Single Stage Lander

- Common TLI Stage Across All HLLV Options With 210 t Usable Propellant

- Alternate Inverted Lander Launch can Minimize Lander Mass and Maximize Cargo Capability & Usable Mass on Lunar Surface

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RS910718-03A

Operations and Programmatics

Lunar Transportation System

Jim Cathcart
(303-977-7263)
MARTIN MARIETTA

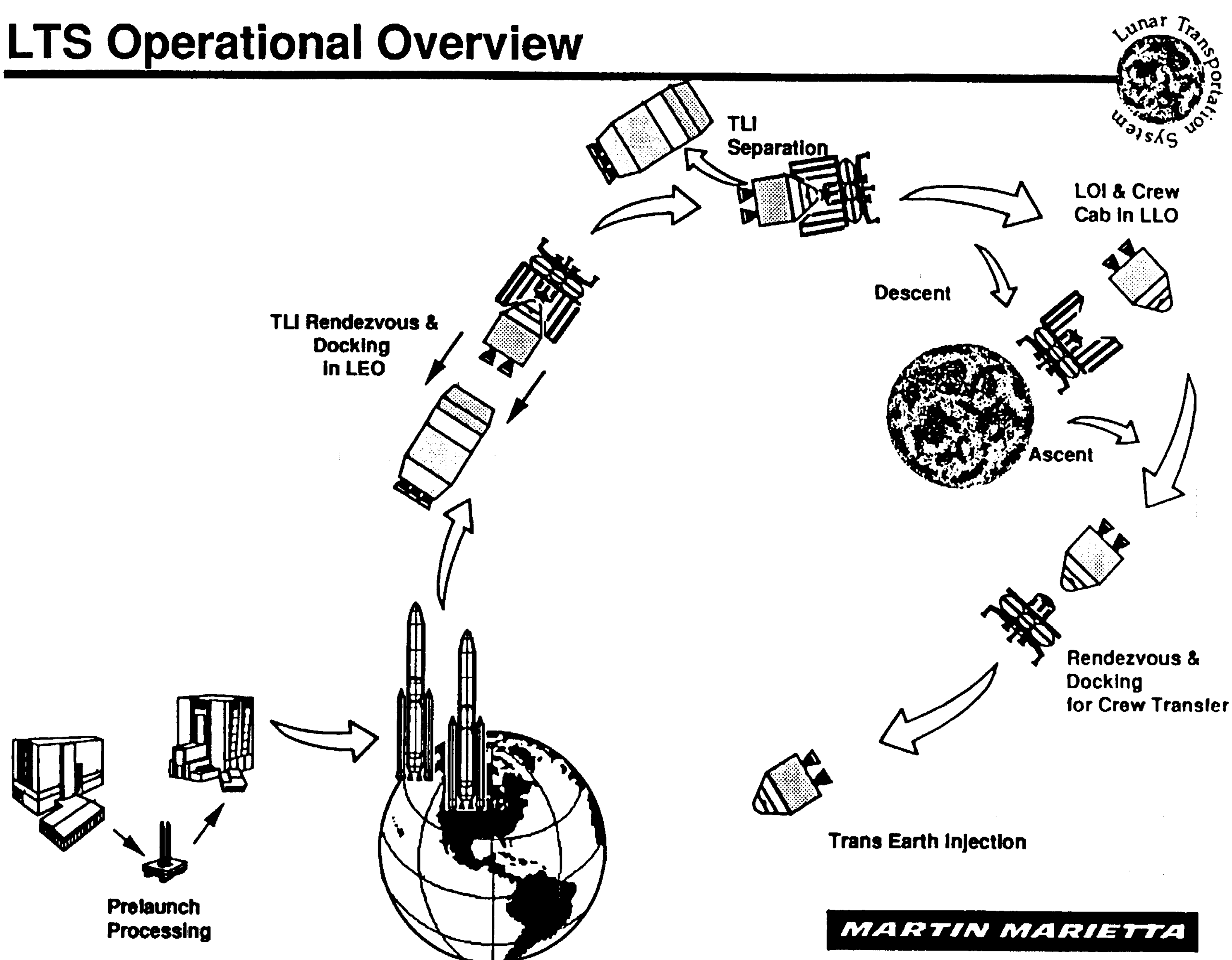
JH910813-12A

Topics

- Operations
  - Overview
  - Facilities
  - Timelines
- Program Planning
  - LTS Development Schedule
  - LTS Flight Test Program
- Cost Analysis
  - Recommended Configuration Costs
- Summary

Operations Overview

LTS Operational Overview



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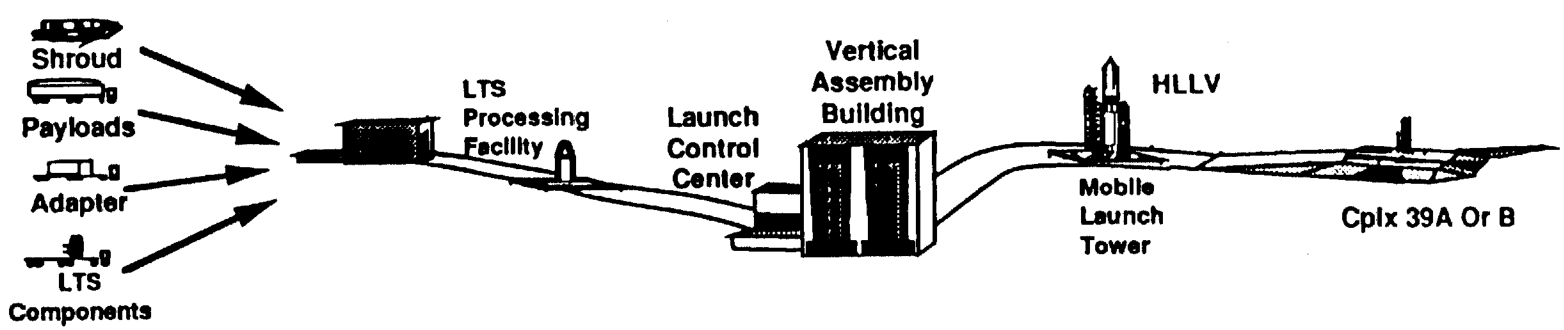
Ground Operations Flow



Ground Operations at KSC Require One New Facility, the LTS Processing Facility Which Functions as an Assembly and Payload Encapsulation Facility

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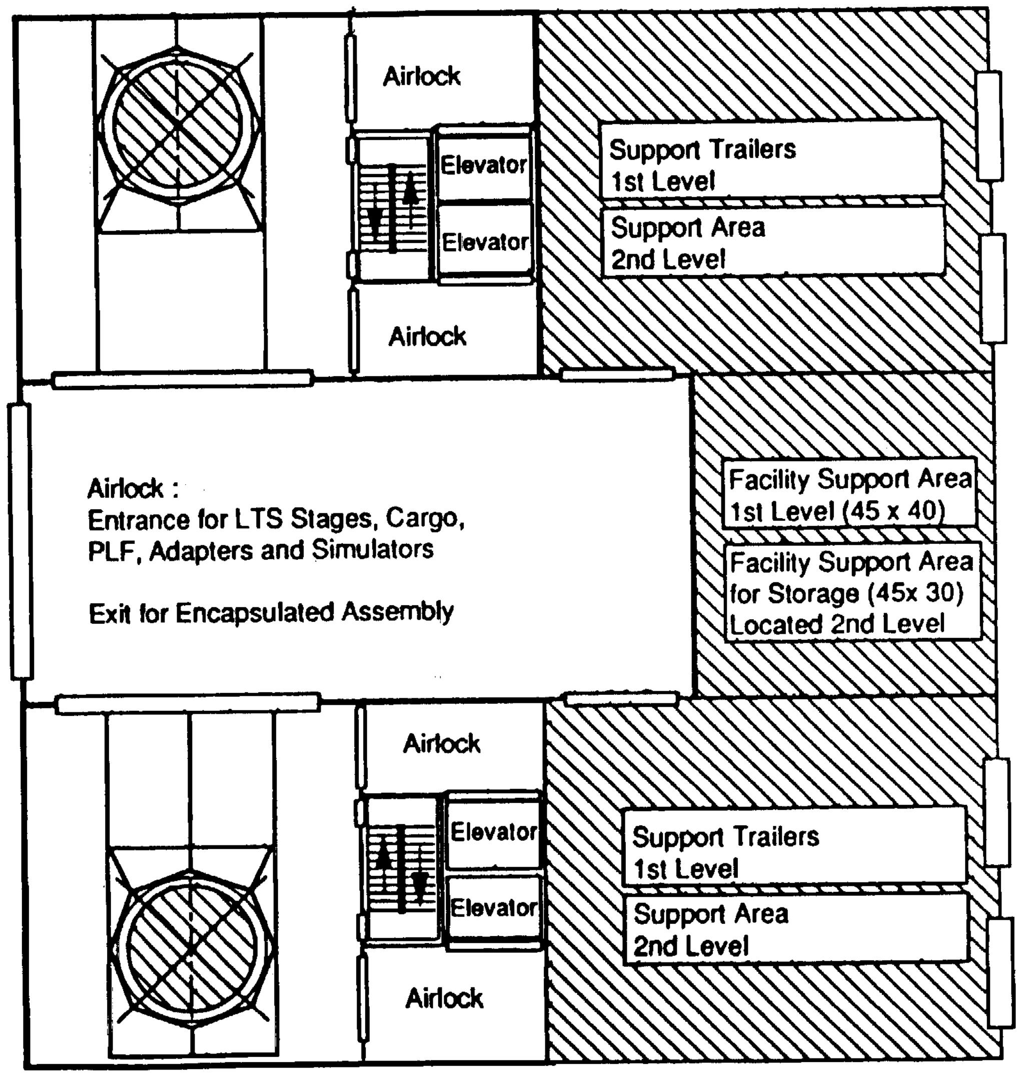
LTS Processing Facility - Plan View



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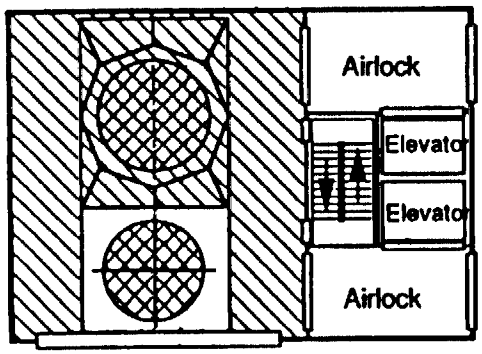
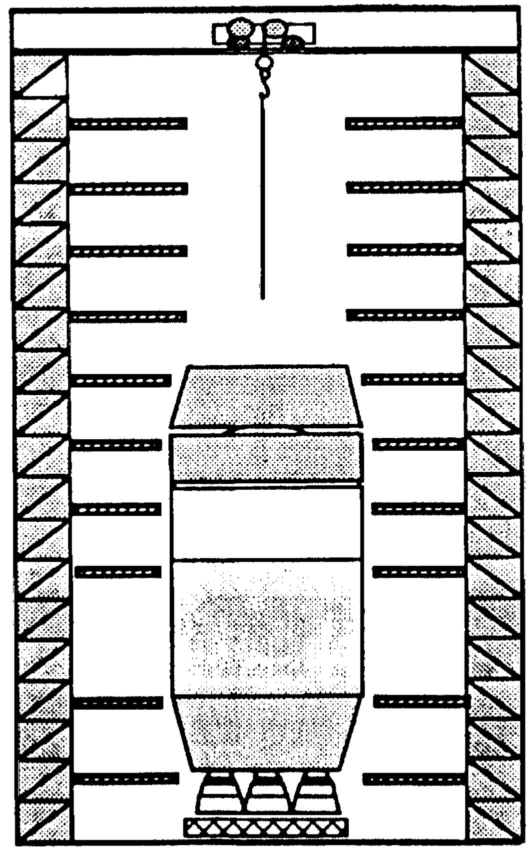
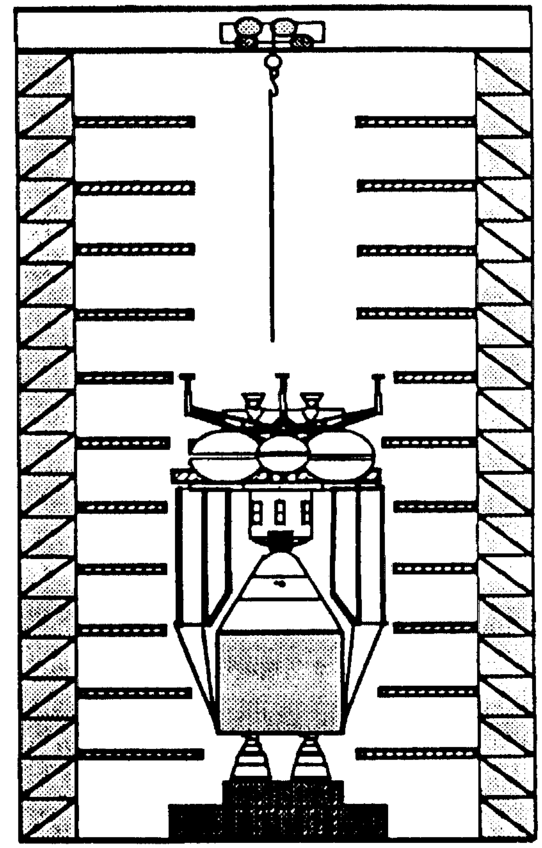
LTSPF Processing Activities



Processing Cell
Configuration



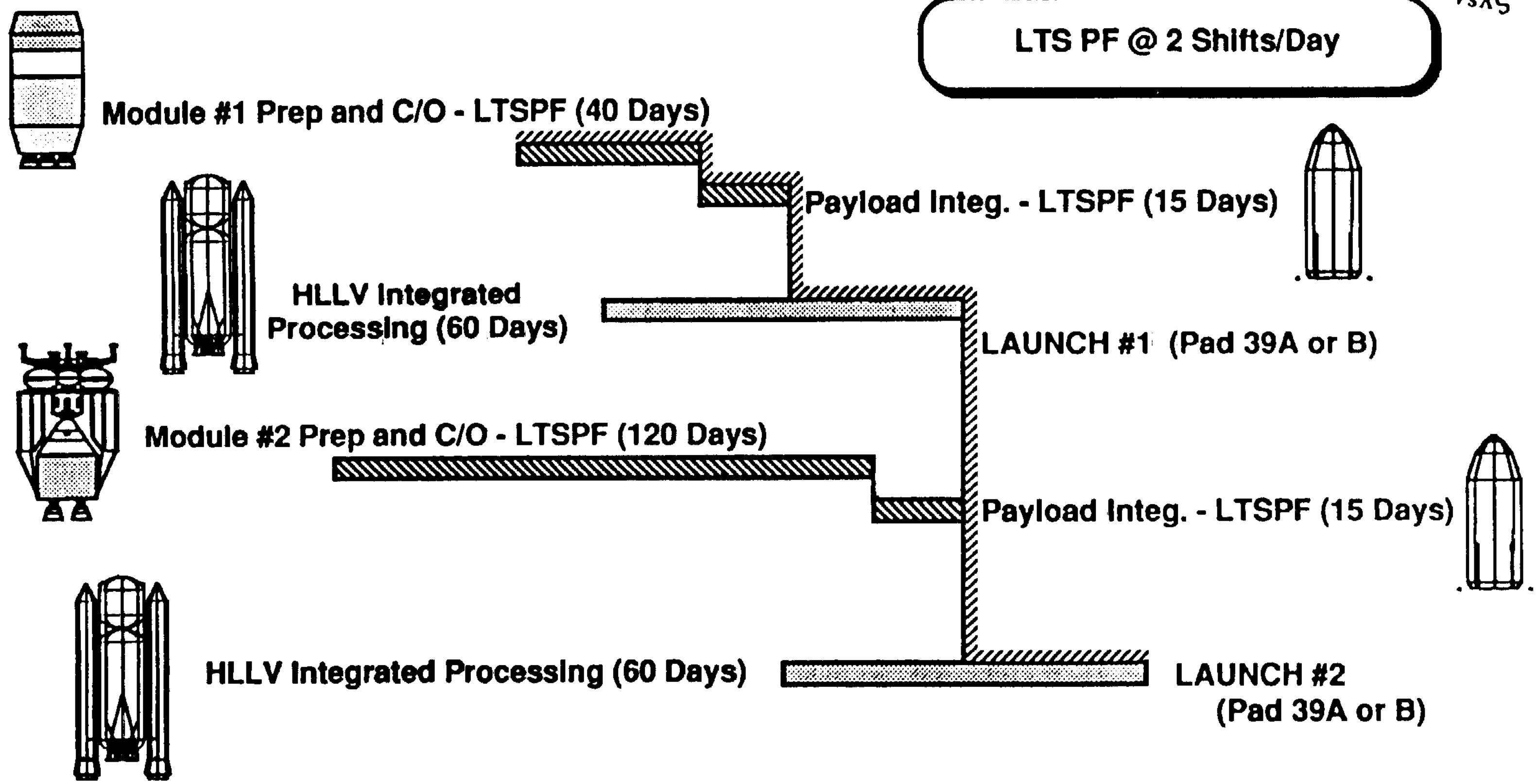
Module #1



Module #2

LTS Processing for 2 HLLV Flights

LTS PF @ 2 Shifts/Day



Module #1 Maintains a Stable Orbit For 30 Days Prior to Rendezvous and Docking Procedure with Module #2

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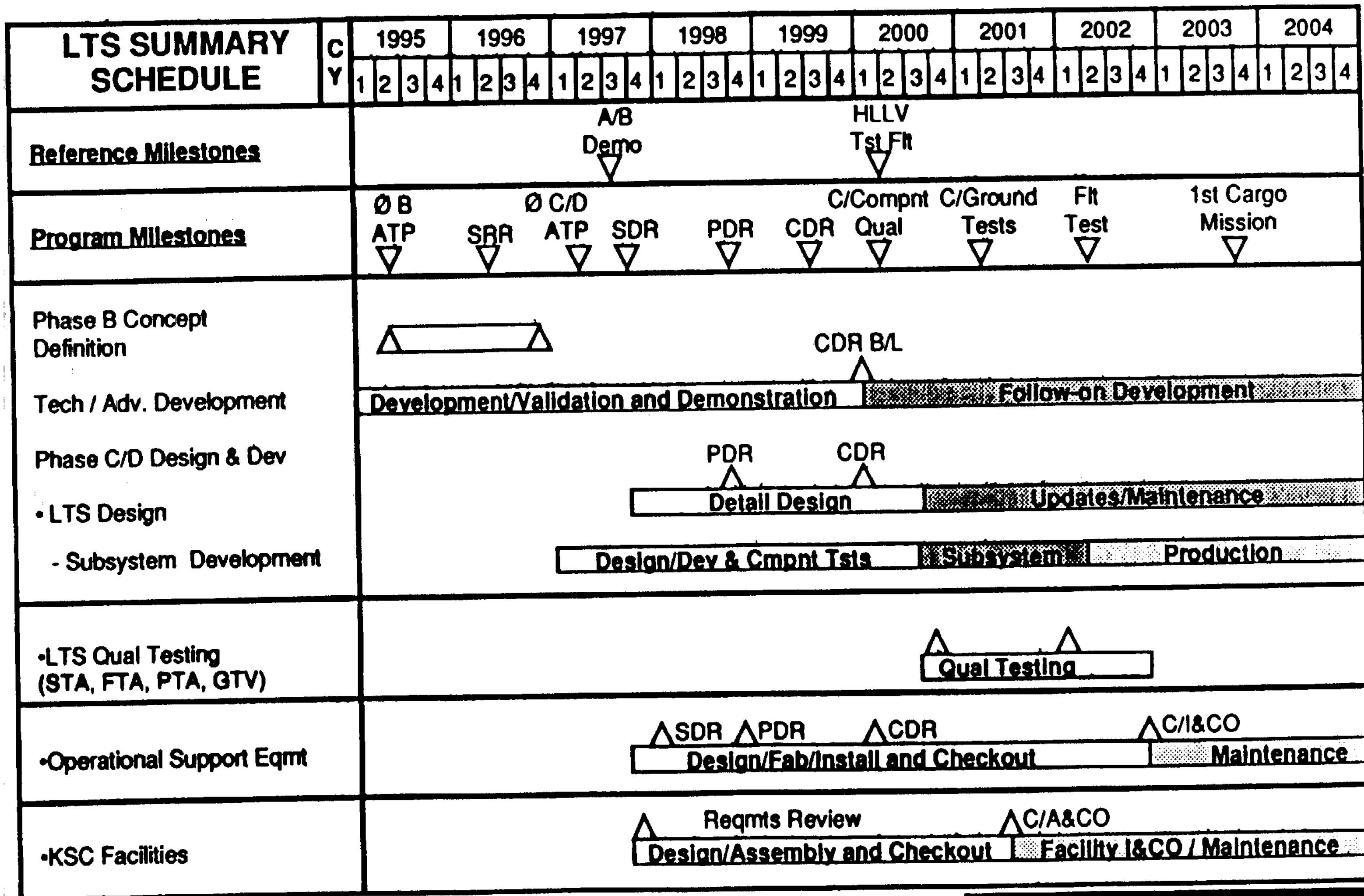
Program Planning Overview

LTS Program Overview

Lunar Transportation System Overview



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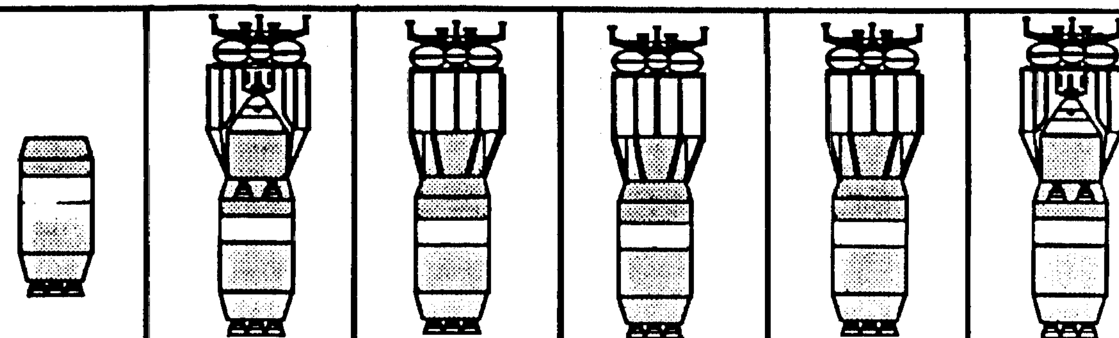
LTS Flight Test Program



Mission Phase \ Test Article	Near Earth Mission	Flight Test Vehicle	1st Cargo Flight	2nd Cargo Flight	3rd Cargo Flight	1st Piloted Flight
Rendezvous and Docking	√	√	√	√	√	√
On-Orbit Verification and Checkout	√	√	√	√	√	√
Trans-Lunar Injection		√	√	√	√	√
Descent		√	√	√	√	√
Ascent		√	EXP	EXP	EXP	√
Trans-Earth Injection		√	EXP	EXP	EXP	√
Ballistic Re-Entry		√	EXP	EXP	EXP	√

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Lunar Transportation System

Cost Analysis Overview

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Cost Analysis Groundrules and Assumptions

Lunar Transportation System

Government Furnished Groundrules:

- All Costs Reported in Millions of 1991 Dollars
- Program Phases Include DDT&E, Production, and Integration and Operations
- A Costed 15% Weight Contingency for Growth
- A 30% Allowance for Requirements Growth
- 8% Allowance for Prime Contractor Fee
- 15% Allowance for Government Support Beyond Scope of Prime Contract
- 0.5% Allowance for DCAS Taxes
- Integration & Assembly Costs for All WBS Levels are Included
- ETO Costs for HLLV at $444 M per Flight ($370 M + 20% For Operations)

Cost Analysis Groundrules and Assumptions

Lunar Transportation System

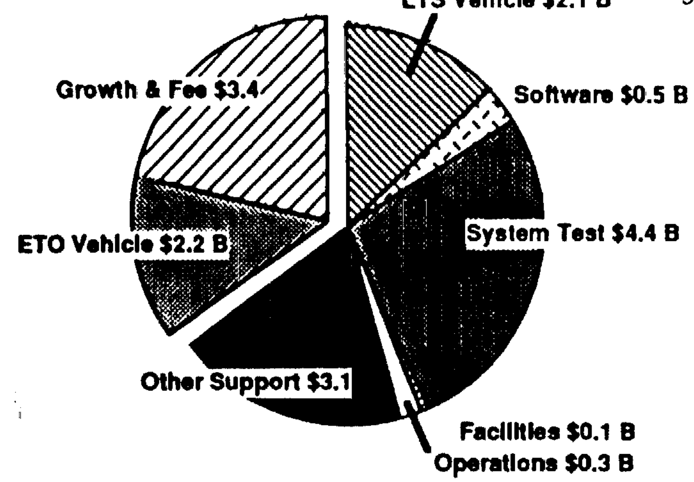
Martin Marietta Assumptions:

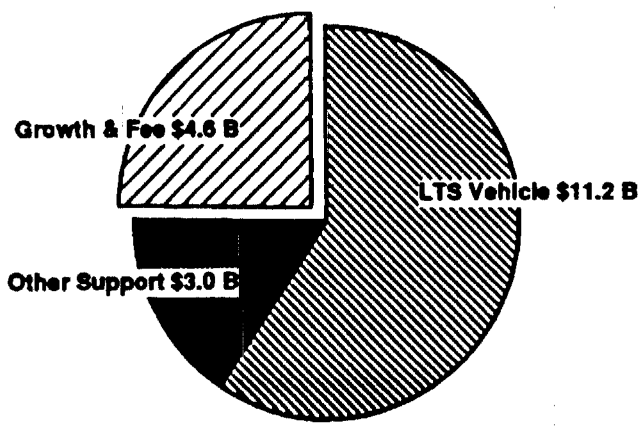
- LTS Initial Launch Configuration in 2003 (Test Flight in 2002)
- Flight Test Hardware and Operations Included in DDT&E Costs
- Technology Development Costs Included in Subsystem DDT&E Costs
- Operations Cost Included for LTS Processing, LTS/Payload Integration, Flight Operations, Spares, and ETO Costs
- Sustaining Engineering and Program Management are Included
- Incremental Development Cost of New Launch Vehicle Included
- Cargo Missions Include 4 Lunar Flights
- Crew Missions Include 21 Lunar Flights
- Vehicle Services Not Required at Lunar Surface
- Expendable Elements are Not Salvaged

LTS Life Cycle Cost Analysis

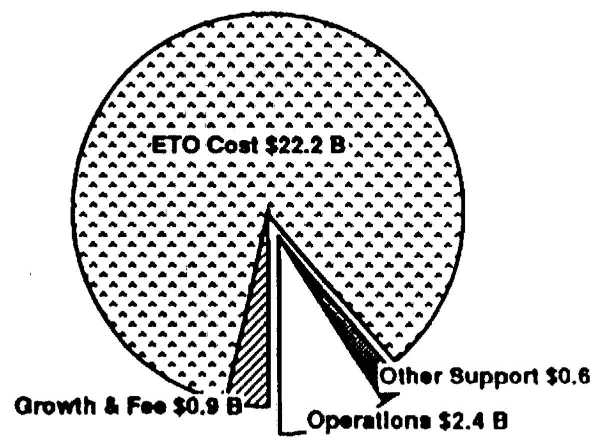
LCC Estimate	
DDT&E	$16.1 B
Prod	$18.8 B
Ops	$ 3.9 B
ETO	$22.2 B
TOTAL	$61.0 B

DDT&E - $23.4 B





Production - $18.8 B



Operations Costs - $26.1 B

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Summary

- Developed Timelines for the LTS Ground Processing Show That the Lunar Missions Can Be Supported With Two HLLV Flights
- LTS Processing Facility Support Concepts Have Been Identified and a Facility Layout Completed
- Development Schedule Has Been Updated to Reflect Changes in the Design and Development Approach
- Final Qualification Test Program Has Been Developed
- Final LCC Estimate and Groundrules Provided

Tasks Produced Key Findings

- "Rendezvous and Docking" is a Feasible Approach To Supporting Lunar Exploration and Habitation.

- Key Rendezvous & Docking/Space Based Commonalties
  - HLLV, Cargo Delivery, Operations Bound System Definition and Cost and Schedule Sensitivities.

- ΔV Budget Consistent With MASE Recommendation
  - Launch Windows & Orbit Altitude Key To Performance Optimization

- Primary Configuration Selection Driven By Cost/Delivery Mass Trends Across Variable Mission Model.
  - HLLV Option #1 Systems Minimizes Cost Fluctuation

Tasks Produced Key Findings

- Recommendation Of A Multi-Stage Configuration
  - TLI
  - LOI/TEI
  - Single Stage Lander
  - Separate Excursion & Transfer Crew Modules (Ballistic)
- Defined Key Ground Processing Timelines and Facility Layouts

Where Do We Go

Lunar Transportation System

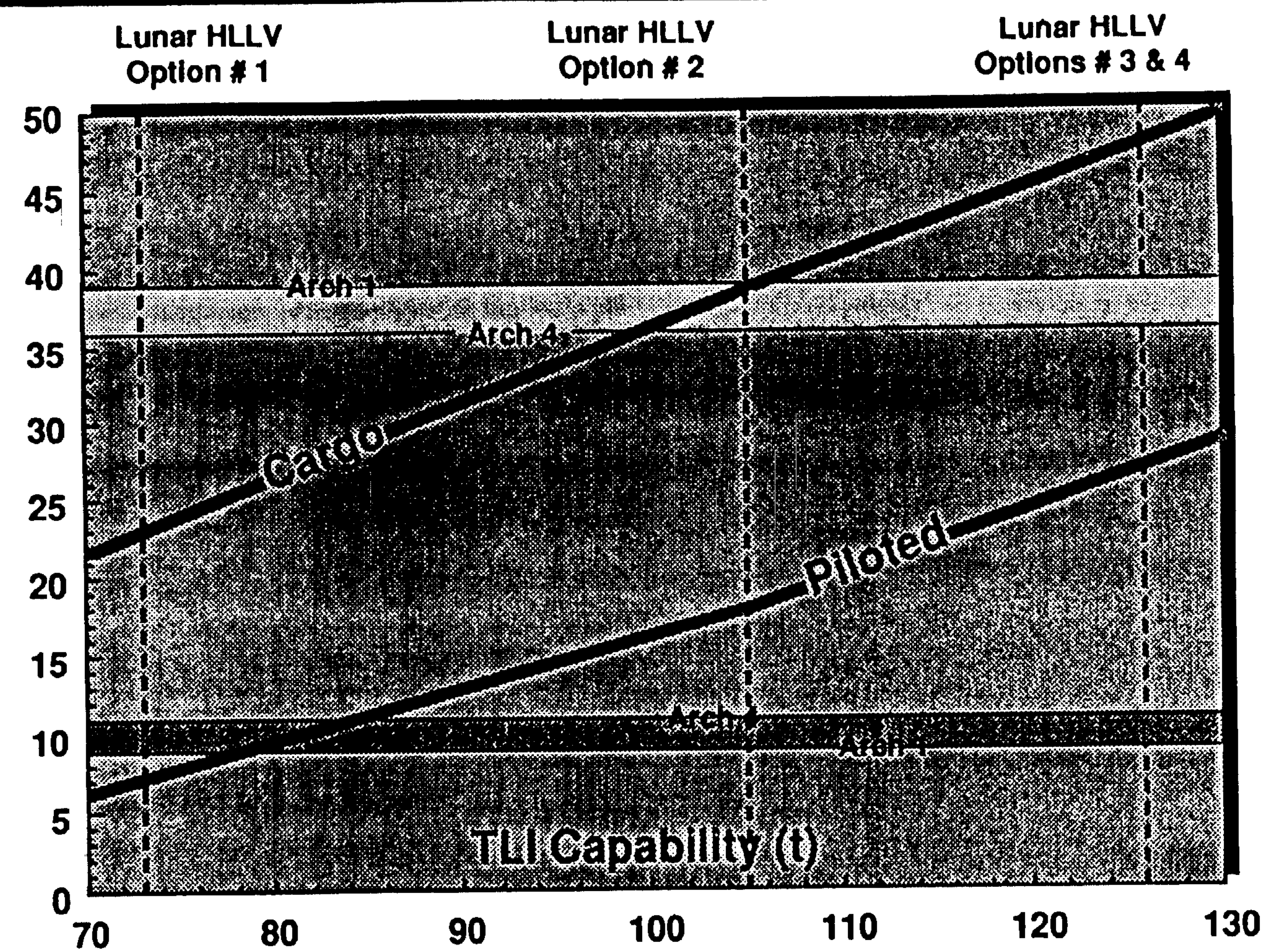
Complete TD07 Requirements

- Level II/MSFC SEI Evaluation Support
  - Augment In-House Lunar/Integration Teams
    - New Data
    - Existing Data
  - Perform Key Studies, Analysis, Design Tasks
- Continue Technology/Advanced Development Assessment

Near Term Activities

- Support Definition of Strategic STV Plan
  - Expansion of Existing Databases
  - Existing/Planned ETO Systems Benefits
  - Integrate Upper Stages Into Transportation Infrastructure
  - Increase Technology/Advanced Development Utilization

Task Data Applicable to SEI Support Efforts



- Sub-Orbital Start of Upper Stage on HLLV Flight #1
- No LEO Kick Stage Required

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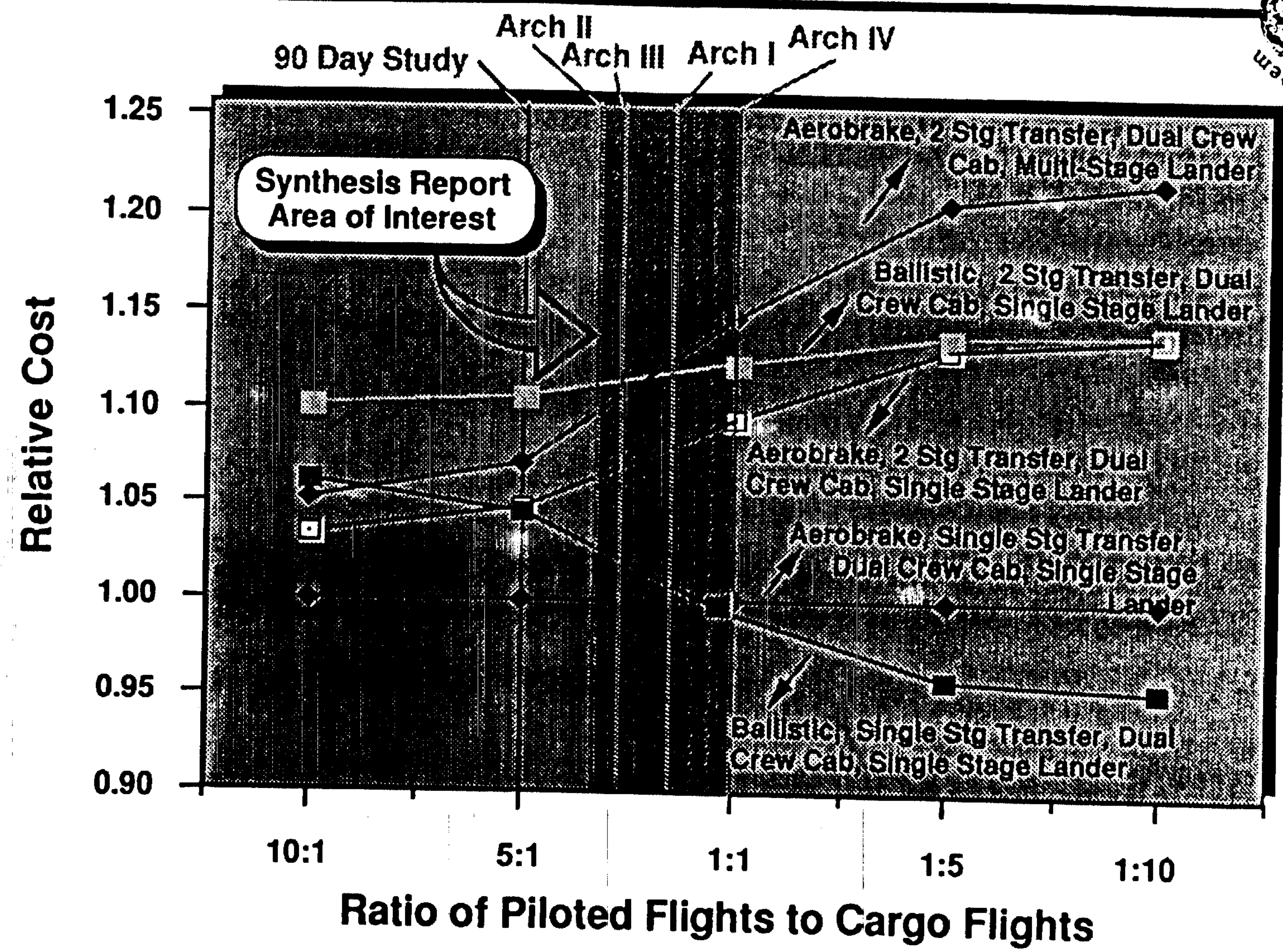
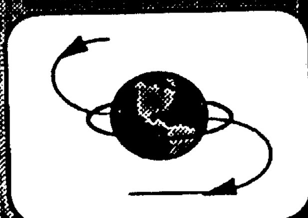
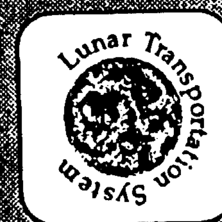
Task Data Applicable to SEI Support Efforts





STV Technology and Advanced Development Benefits Assessment



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STV Tech. & Adv. Dev. Benefits Assessment Tasks

Lunar Transportation System

Task A

- Technology & Advanced Development Assessment
  - Development Plan

Task B

- Technology & Advanced Development Sensitivity Study
  - Vehicle Assessment Tool

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STV Technology & Adv. Dev. Assessment Topics

TASK A - Technology and Advanced Development Assessment

- Approach
- Assessment Criteria
- Key Technologies and Advanced Development Concepts
- Cost and Performance Benefits Analyses and Roadmaps
- Avionics Assessment Model
- Summary

STV Technology & Advanced Dev. Approach

TASK A - Technology and Advanced Develop Assessment

Phase a

- For Key Technology and Advanced Development Concepts:
  - Define Requirements
  - Identify Technology Readiness Levels
  - Assess Cost, Performance, Schedule and Other Benefits
  - Prioritize and Rank

Phase b

- Perform Indepth Analysis of Highest Priority Technology and Advanced Development Concepts Identified in Phase a

Phase c

- Assess Impact of Technology and Advanced Development on Synthesis Group Recommended Architectures

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STV Technology & Advanced Development Areas

Lunar Transportation System

- Cryogenic Fluid Management
- Avionics, Power, Software and Vehicle Health Mgt
- Cryogenic Engines and Propulsion
- Vehicle Structure and Tankage
- Aerobrake
- Flight Operations
- Ground Operations
- Advanced Propulsion
- Vehicle Assembly, Servicing & Processing
- Crew Module
- Environmental Control & Life Support System
- Lunar and Mars Surface Operations

Technology Readiness Levels

Lunar Transportation System

Level	Description	
1 □	Basic Principles Observed and Reported	Technology
2 ◪	Technology Concept and/or Application Formulated	Technology
3 ■	Analytical and Experimental Critical Function and/or Characteristics Demonstration	Technology
4 ◇	Component and/or Brassboard Validated in Laboratory Environment	Technology / Advanced Development
5 ◈	Component and/or Brassboard Validated in Relevant Environment	Technology / Advanced Development
6 ◆	System/Subsystem Model or Prototype Demonstrated in Simulated Environment	Advanced Development
7 ○	System Prototype Demonstrated in Space Environment	Advanced Development
8 ◒	"Flight-Qualified" System	System Development
9 ●	"Flight-Proven" System	System Development

C-3

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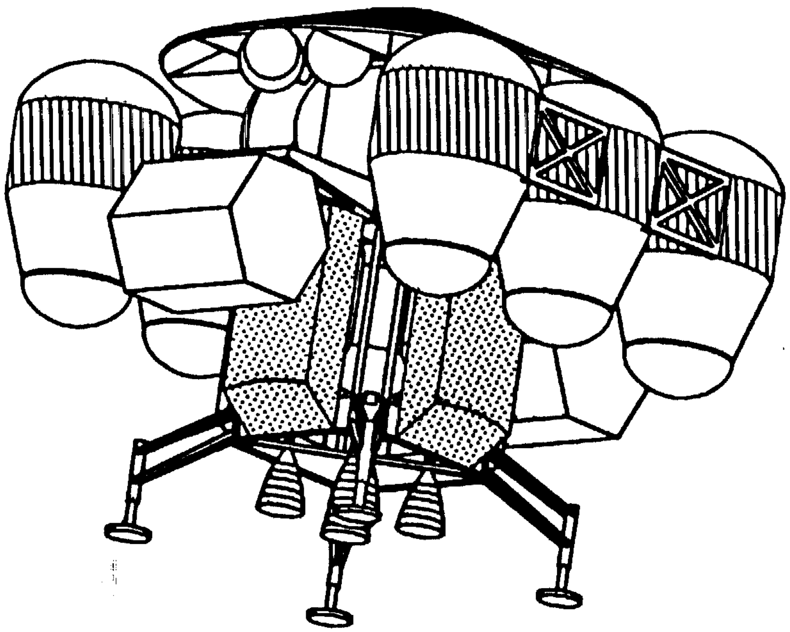
STV Technology & Adv. Dev. Assessment Criteria

• Cost	Life Cycle Cost - Recurring and Nonrecurring Recurring Savings per Vehicle DDT&E and R&T Costs Cost Benefit - LCC/R&T Cost Net Present Value @ 5%
• Performance	Satisfy Operation Requirements Satisfy Safety Requirements Reliability STV Impacts Launch Vehicle and Infrastructure Impacts Robust Design - Large Margins
• Schedule	Readiness Level 6 by STV Preliminary Design Review Risk - Lead Time
• Other	Operational Life - Reusability Producibility Maintainability Adaptability Ability to Man-Rate Fault Tolerance Capability Ability to Space-Base

STV Space-Based Zero Base Technology Concept

Lunar Transportation System

STV Phase 1 Study Reference Vehicle With State-Of-The-Art Technology



- RL10A-4 Engine (Man-Rated & Space-Base Certified)
- Aluminum Tanks and Structure
- Centaur Cryogenic Fluid Management/Wet Tanks
- Off-The-Shelf Aluminum/Mylar MLI
- Space Station Avionics
- Nickel Zinc Batteries
- Apollo Thermal Protection System
- Hydrazine Auxiliary Propulsion System

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STV Technology & Adv. Dev. Assessment - Main Engine

Lunar Transportation System

	RL10A-4	RL10B Derivative	ASE	IME
• Cost ($M) - $\boxed{\$2500/lb\ ETO}$				
LCC Savings	B/L	2500	3700	5400
Recurring Savings per Vehicle	B/L	314.7	545.1	664.8
DDT&E + R&T	150	300	625	450
Cost Benefit (LCC/R&T Cost)	B/L	50	24.7	36.0
Net Present Value @ 5%	39.5	38.6	39.0	38.2
• Performance				
Satisfy Operation Reqmts - Cargo Pilot/Expend	9.8/33.3t	12.8/36.3t	14.6/37.4t	14.2/38.2t
Satisfy Safety Requirements	2	2	2	2
Reliability	1	2	2-3	2
STV Impacts	3	2	1	1
IMLEO / Infrastructure Impacts	243t/3	229t/2	218t/1-2	216t/1
Robust Design / Large Margins	2-3/3	2/2	2/1	2/1
• Schedule				
Readiness Level 6 by STV PDR	1	1	3-4	2
Risk - Lead Time	2	2	1-2	1
• Other				
Operational Life - Reusability	2-3	1-2	1-2	1
Producibility	3	2	1-2	1
Maintainability	4	3	2	2
Adaptability	3	3	2	1
Ability to Man-Rate	3	2	2	1-2
Fault Tolerance Capability	4	3	3	1
Ability to Space- Base	3	2	2	2

Qualitative Assessment - 1 (Good) to 5 (Poor)

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STV Advanced Development Cost Benefits - Avionics

Lunar Transportation System

	Advanced Dev	Benefit
Option I Lunar - Manned - 1 HLLV + 1 Shuttle Launches - Crew plus Zero Cargo to Moon - 2 Weeks Stay on Moon - Ground Based	• Dual Fault Tolerance / High Reliability • Health & Status Mgt - Flight Operations	• Crew Safety, $7,248M per Vehicle Saved • Crew Safety, $7,248M per Vehicle Saved
Option II Lunar - Manned - 2 HLLV + 1 Shuttle Launches - Crew plus 4t Cargo to Moon - Autonomous Docking in LEO - LOI/TEI Stage Remains in LLO - Propulsive Return to LEO - 4 Weeks Stay on Moon - Ground Based	• Dual Fault Tolerance / High Reliability • Health & Status Mgt - In-Space Processing - Flight Operations • Autonomous Rendezvous, Berthing & Docking	• Crew Safety, $7,248M per Vehicle Saved • $117M Savings per Flight • Crew Safety, $7,248M per Vehicle Saved • $269M Savings per Flight
Option III Lunar - Manned - 2 HLLV + 1 Shuttle Launches - Crew plus 15t Cargo to Moon - Assembly/Autonomous LEO Dock - Aerobrake Return to LEO - 24 Weeks Stay on Moon - Space Based	• Dual Fault Tolerance / High Reliability • Health & Status Mgt - In-Space Processing - Flight Operations • Autonomous Rendezvous, Berthing & Docking • Advanced Space Power	• Crew Safety, $7,248M per Vehicle Saved • $346M Savings per Flight • Crew Safety, $7,248M per Vehicle Saved • $269M Savings per Flight • $4M Savings per Flight

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STV Technology & Adv. Dev. Assessment - Avionics

Area: Avionics, Power, Software and Vehicle Health Management

Technology/Adv Dev-Requirements

- Man-rated Avionics: Dual Fault Tolerant & High Reliability
- Automated Software Design & Validation: Auto Code Generation & Auto Validation / Verification
- Autonomous Rendezvous, Berthing & Docking: Dual Fault Tolerant; Minimize Cost/Vehicle Weight & Power
- Advanced Space Power: Minimize Vehicle Weight: Fuel Grade Cells; Fault Tolerant Distribution
- Health & Status Management: Dual Fault Tolerant: Redundancy Mgt.; Minimize Cost/Veh Weight & Power

Benefits

	Option One	Option Two	Option Three
• Man-rated Avionics	Required-Crew Safety, $7,248M/Vehicle Saved		
• Automated			
-Code Generation	$565M	$565M	$565M
-Software 'Valid & Verification	$188M	$188M	$188M
• Autonomous Rendezvous, Berthing & Docking		$269M	$269M
• Advanced Space Power			$4M
• Health & Status Management			
-Ground Processing	$42M	$42M	$42M
-In-Space Processing		$117M	$346M
-Flight Operations	Crew Safety, $7,248M/ Vehicle Saved		

Technology Readiness Level

	Current TRL	Yrs to 6/7 TRL
• Man-rated Avionics	3	6/8
• Automated Software Design & Validation	2	8/10
• Autonomous Rendezvous, Berthing & Docking	5	4/6
• Advanced Space Power	3	6/8
• Health & Status Management	3	7/9

Priority & Ranking

	Priority	Ranking
• Man-rated Avionics	Highest	1
• Automated Software Design & Validation	Highest	2
• Autonomous Rendezvous, Berthing & Docking	Higher	4
• Advanced Space Power	Higher	5
• Health & Status Management	Highest	3

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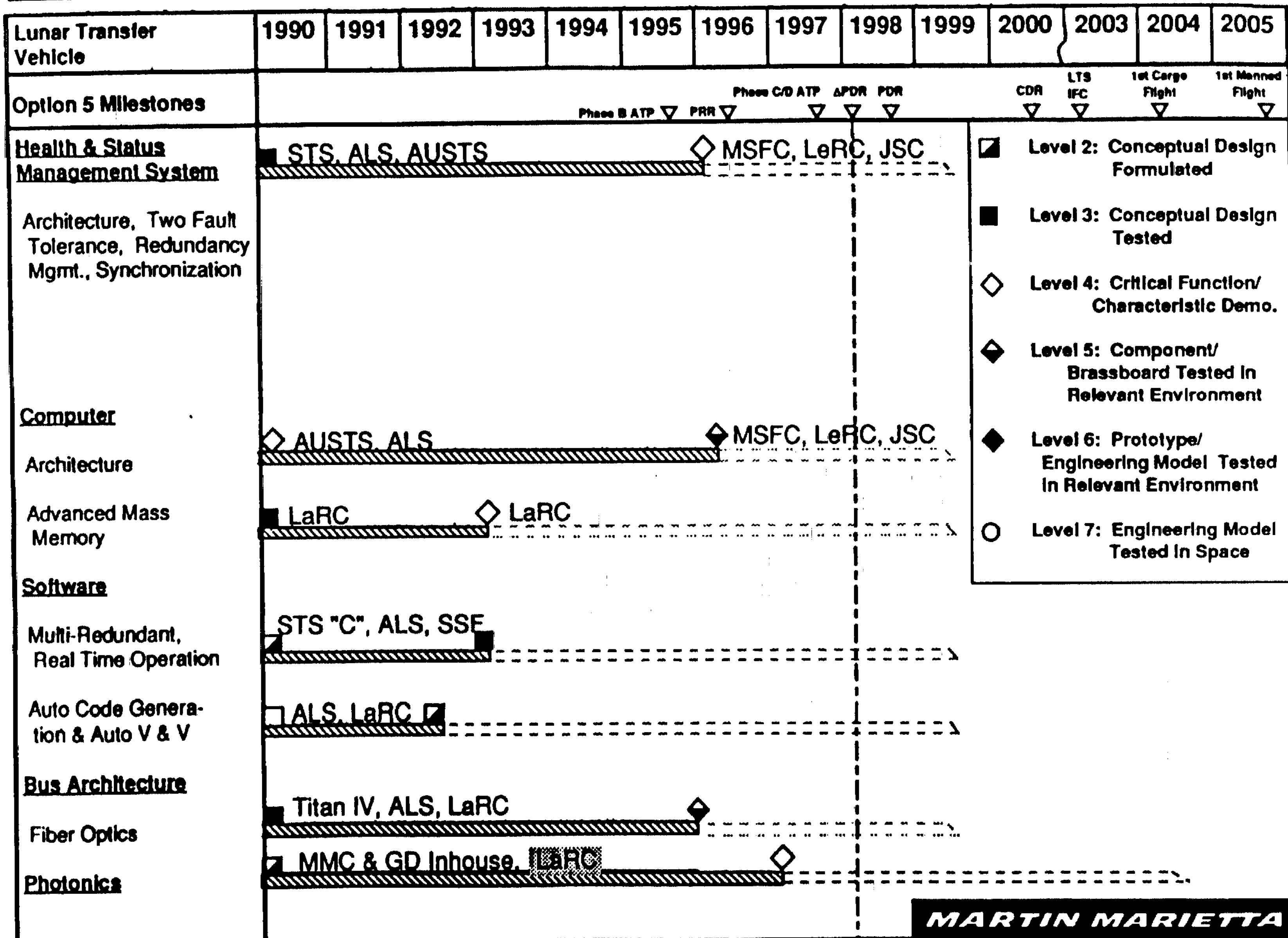
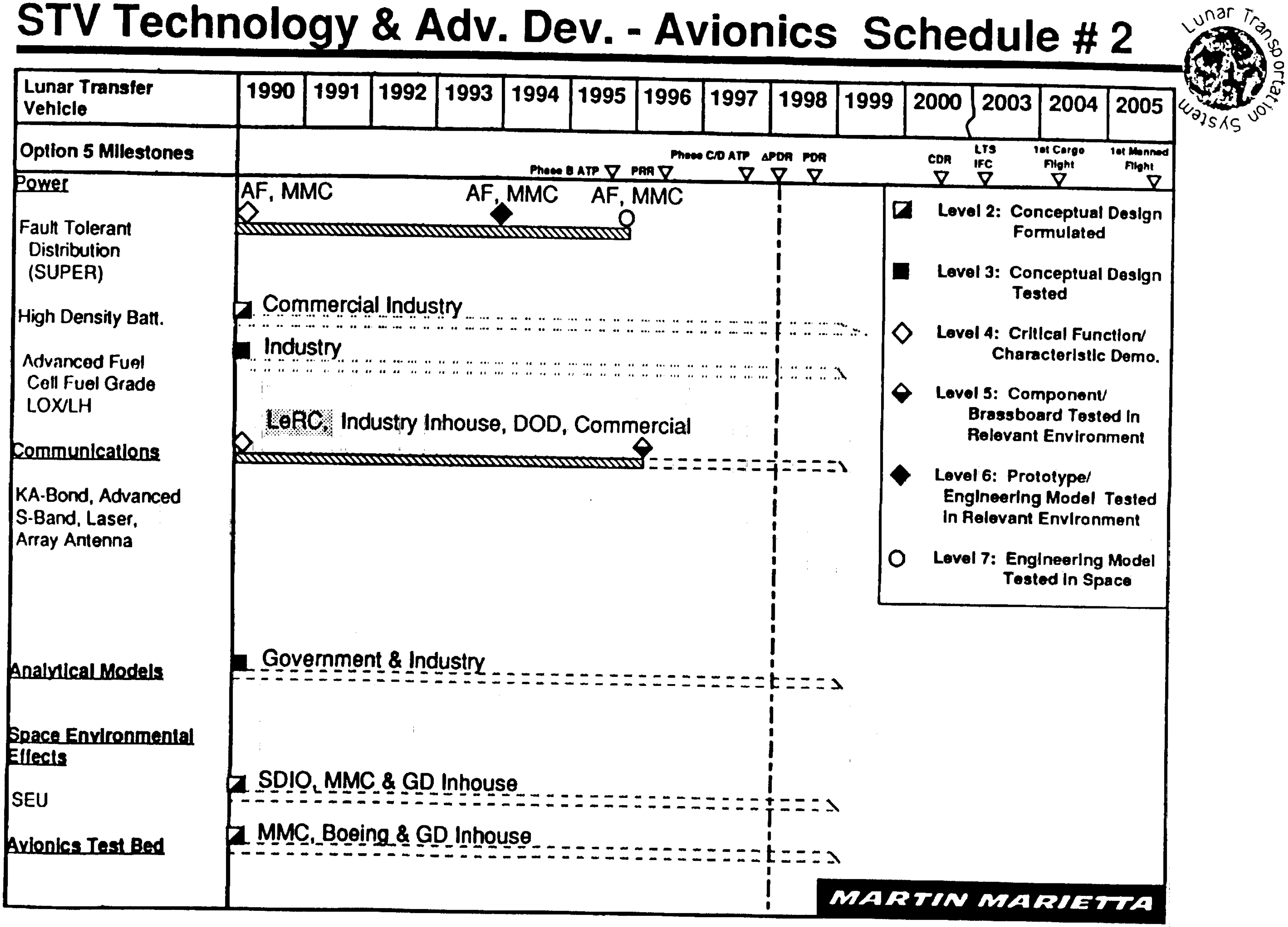
STV Technology & Adv. Dev. - Avionics Schedule # 1



STV Technology & Adv. Dev. - Avionics Schedule # 2



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Avionics Assessment Model

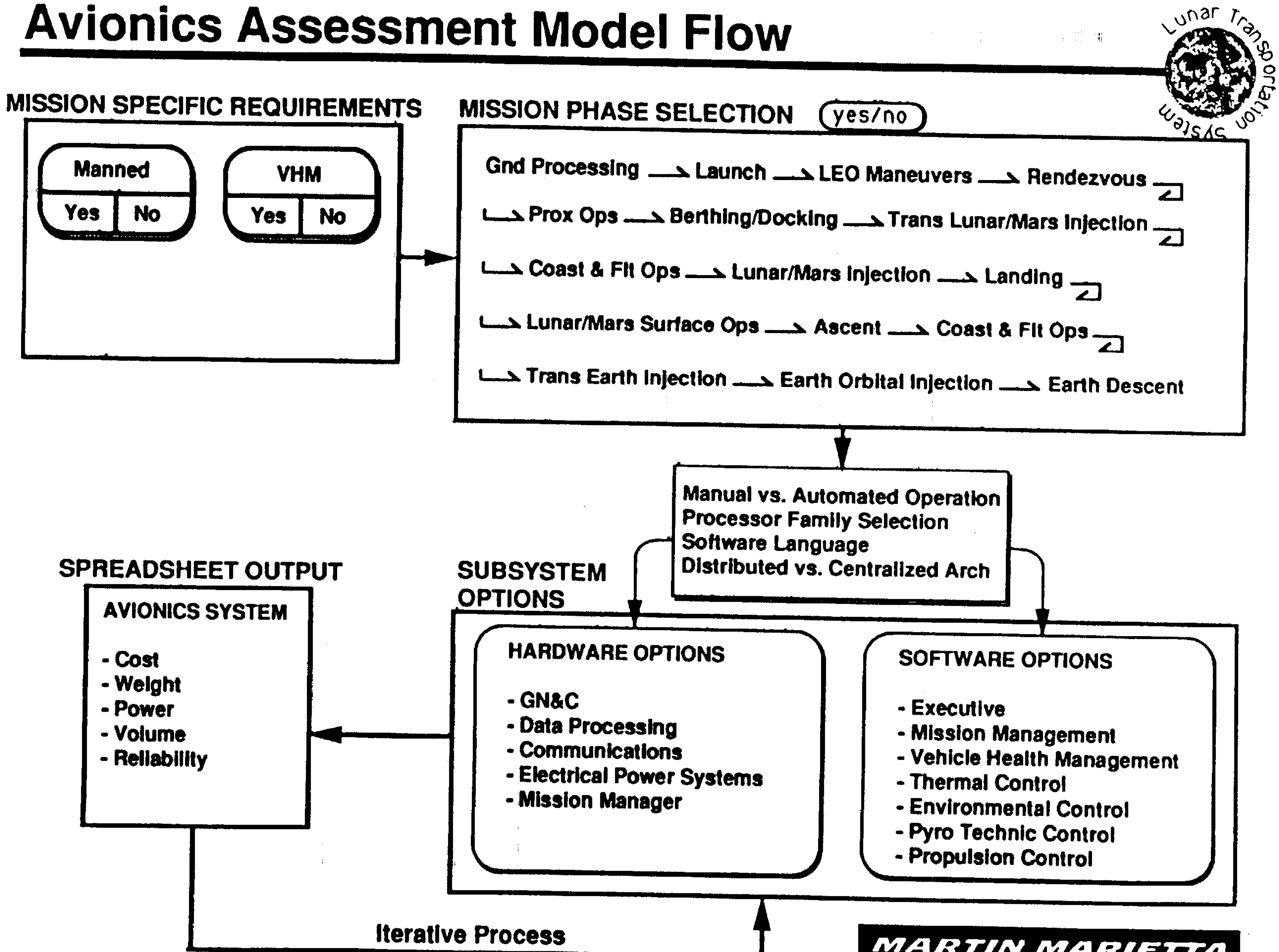
- Model Rapidly Assesses Impact of Mission Specific Requirements and Subsystem Hardware and Software Options have on Avionics System Cost, Performance and Mission Reliability

- Macro-Driven EXCEL Spreadsheet Format

- User Selects Options, Spreadsheet Automatically Accesses Avionics Weight, Power, Volume, Cost and Failure Rate Parametric Database

- Provides Comparison Between Specific Avionics Subsystems, Including GN&C, Databuses, Software, Power, etc., as Well as Complete Avionics System Evaluation

Avionics Assessment Model Flow



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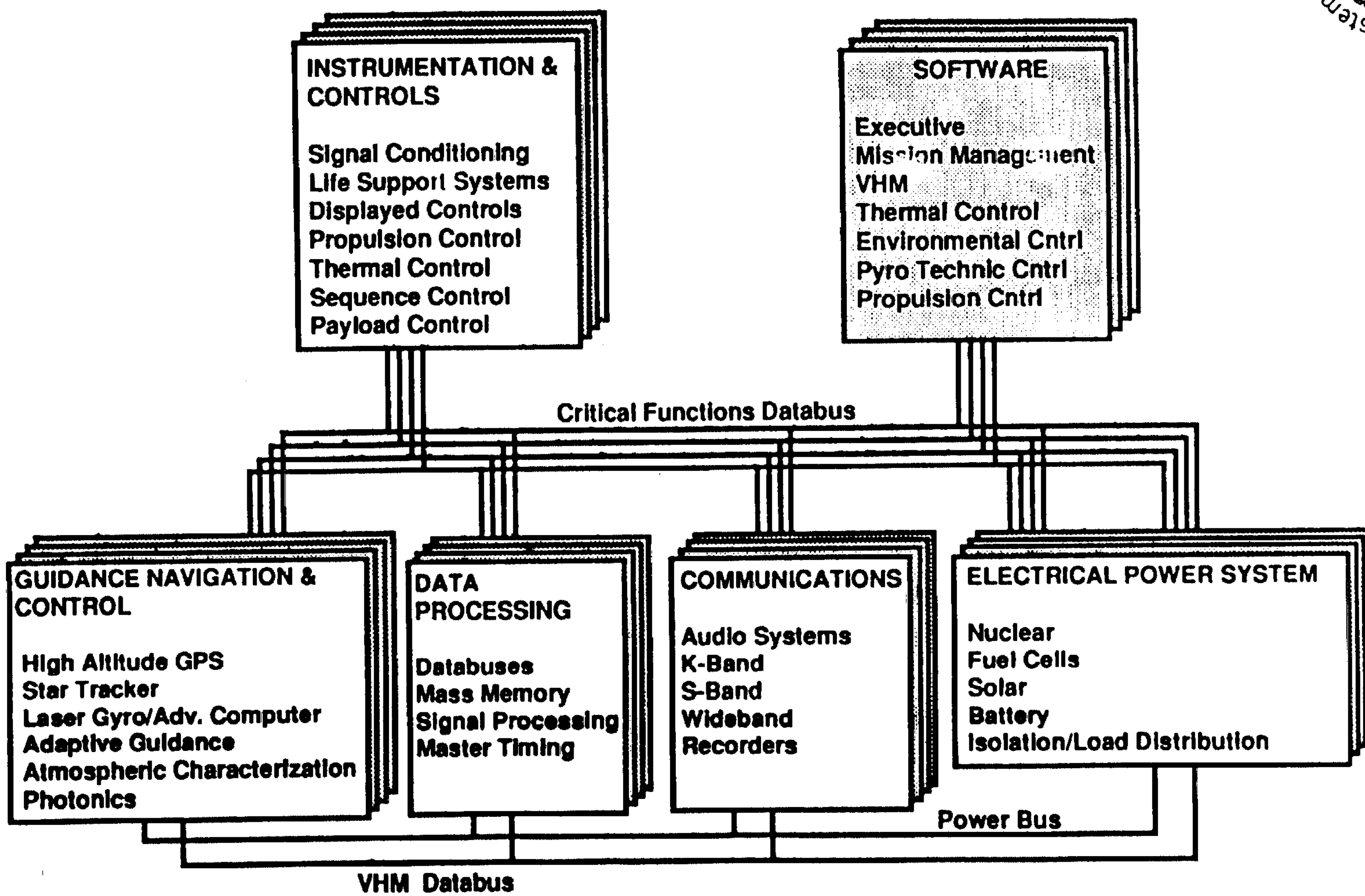
Hardware and Software Avionics Candidates



STV Technology & Adv. Dev. Assessment Summary

Lunar Transportation System

- Completed Detailed Cost and Performance Benefits Assessment, Key Requirements, Technology Readiness Levels, Prioritized and Ranked Technologies and Advanced Development Concepts for:
  - Cryogenic Fluid Management
  - Avionics, Power, Software and Vehicle Health Management
  - Cryogenic Engines and Propulsion
- Completed Initial Cost Benefits Assessment, Key Requirements, Technology Readiness Levels, Prioritized and Ranked Technologies and Advanced Development Concepts for:
  - Vehicle Structure and Tankage
  - Aerobrake
  - Flight Operations
  - Advanced Propulsion
  - Vehicle Assembly, Servicing and Processing
  - Crew Module
  - Environmental Control and Life Support System
- Detailed Avionics Benefits Assessment in Process

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STV Technology and Advanced Development Sensitivity Study

Lunar Transportation System

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JI910813-14

Topics

Task B - Technology and Advanced Development Sensitivity Study

- Approach
- Design of Experiments
- Cryogenic Fluid Management Analysis
- Cryogenic Engines Analysis
- Structures Analysis
- Avionics Analysis
- Vehicle Advanced Technology Sensitivity Spreadsheet
- Summary

Advanced Technology Sensitivities Study

- Objective: Develop a Vehicle Configuration Analysis Tool Which Evaluates the Performance and Cost Benefits of Implementing Advanced Technology on a Vehicle
  - Vehicle Advanced Technology Sensitivity Spreadsheet (VATSS)

- Utilize Taguchi Design of Experiments (DOE) to Minimize the Amount of Analysis Required

- Evaluate Six Technology Areas
  - Cryogenic Fluid Management (CFM)
  - Cryogenic Engines
  - Structures
  - Avionics
  - Aerobrake
  - In Space Operations and Assembly

Groundrules and Assumptions

Lunar Transportation System

- Ground-Based, Autonomous Rendezvous/Docking and Space-Based Vehicle Configuration Options
- Lunar Missions
- Manned Missions Will Include a Crew of Four People
  - Deliver Between 0 and 15 tonnes of Cargo
- Cargo Mission Will Deliver Between 5 and 35 tonnes
- Only Cryogenic Propellant Systems
- HLLV and Crew Module Specifications Will be Provided by MSFC
- Cost Estimates Shall be Reported in Millions of 1991 Dollars
  - Program Phases Include DDT&E, Production, and Integration and Operations
  - A Costed 15% Weight Contingency for Growth will be Included
  - Integration and Assembly Costs for ALL WBS Levels are Included
  - Flight Test Hardware and Operations Included in DDT&E Costs
  - Operations Cost Included for Processing, Vehicle/Payload Integration, LEO Node Operations, Flight Operations, Spares, and ETO Costs

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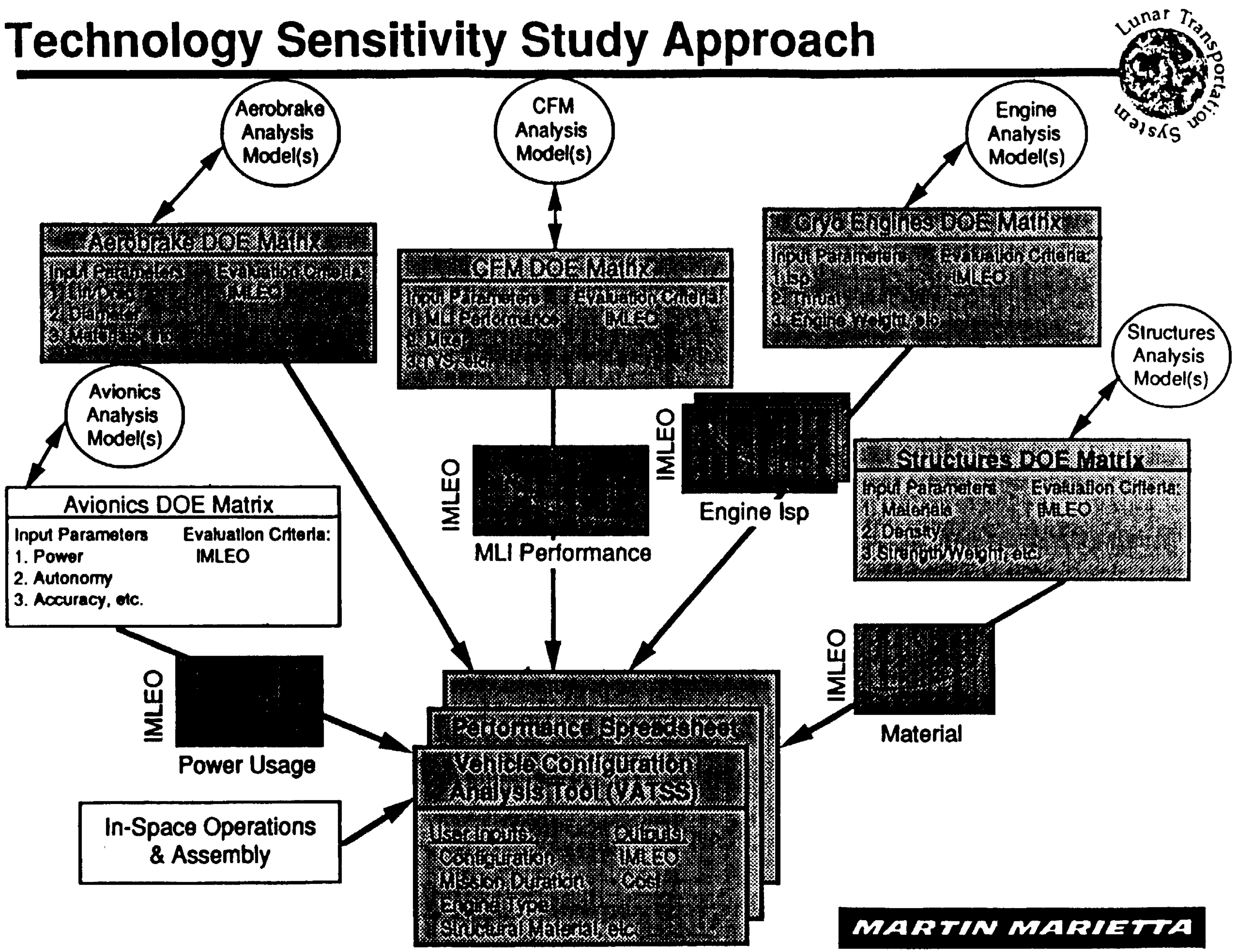
Technology Sensitivity Study Approach

Lunar Transportation System

- Evaluate Individual Technology Areas
  - Identify Parameters Which Describe Technology Area
  - Use DOE to Define Analysis Process
  - Utilize Existing Analysis Tools to Perform Analysis
  - Utilize DOE Statistical Basis to Evaluate Results of Analysis
  - Select Driving Parameters From Technology Area

- Build a Vehicle Configuration Analysis Tool from These Driving Parameters
  - The Spreadsheet Analysis Will be Based on Sensitivity Curves Developed in the DOE Analysis
  - The Spreadsheet Provides a Simplified Analytical Tool Which Precludes Running the Individual Analytical Models for Every Vehicle Analysis

Technology Sensitivity Study Approach



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Design of Experiments Explanation

- Taguchi Design of Experiments (DOE) is a Tool to Evaluate the Importance of Parameters in an Experiment or Analytical Process

- DOE Reduces the Number of Analyses or Experimental Runs by Using Statistical Analysis
  - All Possible Parameters Combinations Could Result in an Excessively Large Analysis Matrix
  - Taguchi Uses Orthogonal Arrays to Define the Minimum Number of Analysis Runs
  - Statistical Analysis Employed to Extract the Important Information from the Analysis Runs

- DOE Helps Organize the Analytical Process
  - Selection of Parameters and Range of Parameters an Important Factor

- Orthogonal Array Matrices Assure Efficient Investigation and Repeatable Results
  - Maximum Information Obtained for Minimum Effort

- DOE Results will Provide Sensitivity Curves Which Can Be Used For Further Analysis

Technology Areas and Related Parameters

Lunar Transportation System

Overall Vehicle Parameters
- Configuration
- Payload Mass
- Mission Duration

Aerobrake
- Lift/Drag
- TPS
- Advanced Materials
- Diameter/Mass

Cryogenic Fluid Management
- Insulation Performance
- Thermodynamic Vent System
- Mixer
- Advanced Tank Materials
- Vapor Cooled Shield

Cryogenic Engines
- Isp
- Thrust
- Engine Weight
- Chamber Pressure
- Engine Cycle

Avionics
- Power Requirements
- Redundancy
- Software Development
- Autonomy
- Accuracy
- System Mass and Volume

In-Space Operations and Assembly
- Autonomy
- Ground vs Space Based

Structures
- Advanced Materials
- Density
- Strength/Weight
- Environmental Factors
- Configuration Approach

■ - Completed Analysis

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CFM DOE Analysis Process

- Investigated 5 Parameters
  - Multilayer Insulation (MLI) Thickness (0.5 to 3.0 inches)
  - Mission Time (70 to 220 days)
  - Mixer - either included or not
  - Thermodynamic Vent System (TVS) - either included or not
  - Vapor Cooled Shield (VCS) - either included or not

- Evaluated These Parameters Using a DOE L16 Matrix for Their Effect on a <u>System Mass</u>
  - Boiloff/Vented Mass - Heat Flux Entering the Fluid (LH2 and LO2)
  - Insulation Mass
  - Additional Tank Mass - Tank Length Added to Contain the Boiloff/Vented Propellant
  - Additional RCS Propellant and Hardware - Required to Settle Tanks Prior to Venting
  - Hardware Mass - Mixer System, VCS, and/or TVS (LH2 Tanks Only)

- Utilized Multiple Analysis Tools
  - Martin Marietta Cryogenic Analysis Program (MMCAP)
  - MLICALC
  - Tank Sizer

- Evaluated the STV Phase I Study Vehicle Configuration Which Is A Space-Based, Reusable Vehicle

CFM DOE L16 Matrix

Parameters and Ranges

Analysis Run	MLI 0.5 3.0	Mission Time 70 220	TVS on off	Mixer on off	VCS on off	System Mass (lbm)
1	0.5	70	on	on	on	16500
2	0.5	70	on	off	off	14296
3	0.5	70	off	on	off	18819
4	0.5	70	off	off	on	16546
5	0.5	220	on	on	off	25447
6	0.5	220	on	off	on	24868
7	0.5	220	off	on	on	27932
8	0.5	220	off	off	off	29806
9	3.0	70	on	on	off	9632
10	3.0	70	on	off	on	10829
11	3.0	70	off	on	on	9667
12	3.0	70	off	off	off	9510
13	3.0	220	on	on	on	15098
14	3.0	220	on	off	off	11837
15	3.0	220	off	on	off	13790
16	3.0	220	off	off	on	14308
Percent Contribution to Variation	[illegible]	30.28	[illegible]	0.22	1.06	

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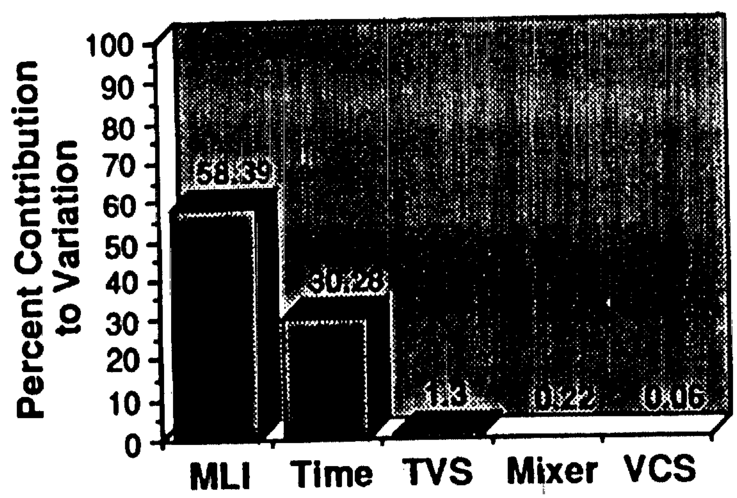
CFM DOE Results

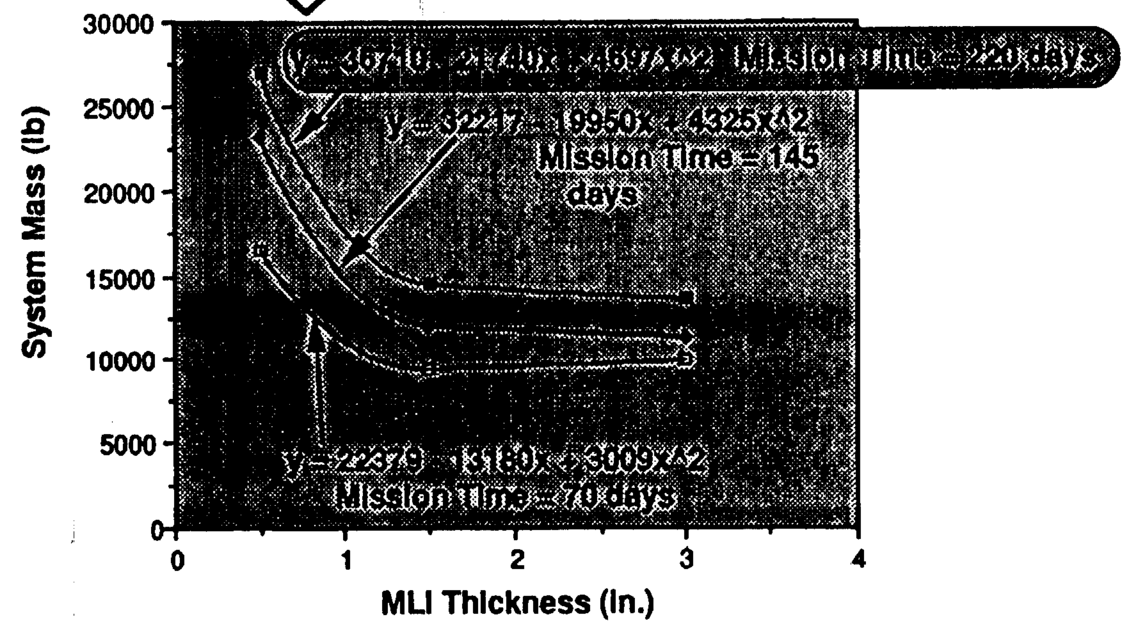
- DOE Reduced the Number of Analysis Runs from 75 to 48
  75 = 5 Parameters at 2 Levels Using 3 Analysis Tools
  48 = 16 Runs Using 3 Analysis Tools
- Percent Contribution to Variation Indicates Which Parameter Is Most Influential on Reduction of System Mass
- MLI Thickness and Mission Time Are Greatest Contributors
  - TVS, Mixer and VCS Do Not Significantly Reduce the System Mass for the Relatively Short Mission



CFM DOE Relationships

- Relationships Between MLI Thickness, Mission Time and System Mass Determined from DOE Results
  - Further Analysis Required to Determine Non-Linearity of Relationship
  - Generated Curve-Fit Equations from DOE Results

- These Curve-Fit Equations Will be Used in the Vehicle Analysis Tool (VATSS)



Cryogenic Engines DOE Analysis Process

- Determine Which Engine Parameters Have the Greatest Influence on Vehicle Performance

- Original Parameters Selected for Evaluation:
  - Isp
  - Thrust/Weight
  - Chamber Pressure
  - Expansion Ratio
  - Mixture Ratio

  But These Parameters Influence Engine Isp and Thrust

- Final DOE Parameters and Ranges
  - Isp - 445, 465, 490 sec
  - Thrust/Weight - 32.5, 44, 57
  - Parameters Span the Range from Current RL-10 Capability to Predicted Advanced Engine Capability

- Evaluated These Parameters Using a DOE L9 Matrix for Their Effect on Vehicle Initial Mass to Low Earth Orbit (IMLEO)
  - Propellant Mass
  - Support Structure
  - Inert Vehicle Mass
  - IMLEO Describes the Mass That Must Be Delivered to LEO in Order to Complete the Cargo Delivery

Cryogenic Engines Analysis Configuration

- Evaluated the STV Phase I Vehicle Configuration
  - Two Payload Capabilities Were Evaluated
    - No Cargo - Delivers Crew to the Lunar Surface
    - 15t Cargo - Delivers Crew and Cargo to Surface
  - Multiple DOE L9 Matrices Required to Evaluate the Different Thrust Levels and Cargo Capabilities
  - Thrust/Weight Varied by Holding Thrust Constant and Changing Engine Weight

SINGLE ENGINE WEIGHT (kg)

Thrust Level	Thrust/Weight		
(N)	32.5	44	57
66 720	209	155	119
92 520	291	215	166
155 680	489	363	279

- STV Phase I Vehicle Performance Model Used for Analysis
  - Mission Time and ΔV Remained Constant
  - Thrust Remained Constant Throughout a Set of Runs
  - 5 Engines on Vehicle
  - Performance Based on Running Only Three Engines (To Meet Two Fault Tolerance)
  - Isp and Thrust/Weight Changes Effect Propellant Load, Support Structure, Propellant Tank Size and Inert Vehicle Mass

Engine DOE L9 Matrices

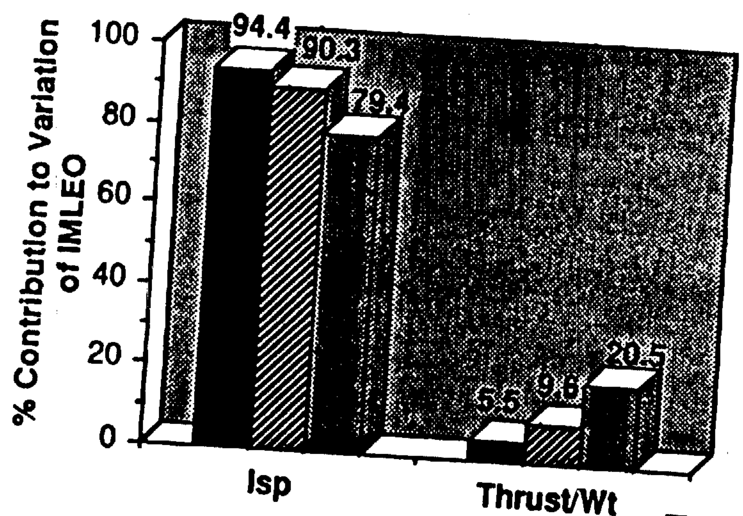
15t Cargo Thrust = 92,520 N

15t Cargo Thrust = 66,720 N Run	Isp 445 465 490	Thrust/Weight 32.5 44 57	IMLEO (kg)
1	445	32.5	276597
2	445	44	269140
3	445	57	264313
4	465	32.5	240218
5	465	44	234584
6	465	57	230892
7	490	32.5	208922
8	490	44	204480
9	490	57	186550
Percent Contribution	94.83	4.21	

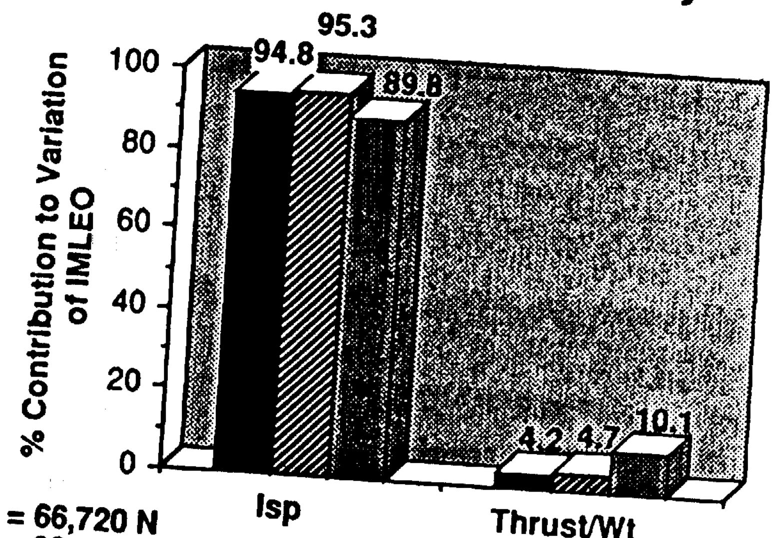
Engine Results

- DOE Reduced the Number of Analysis Runs From 162 to 54
  - 162 = 3 Isp Values at 3 Level x 9 Thr & Wt Values x 2 Cargo
  - 54 = 6 Sets of L9 Runs
- Percent Contribution to Variation Indicates Which Parameter is Most Influential on IMLEO Reduction

No Cargo Capability



15t Cargo Capability



■ Thrust = 66,720 N
■ Thrust = 92,520 N
■ Thrust = 155,680 N

Isp is the Largest Contributor to Changes in IMLEO

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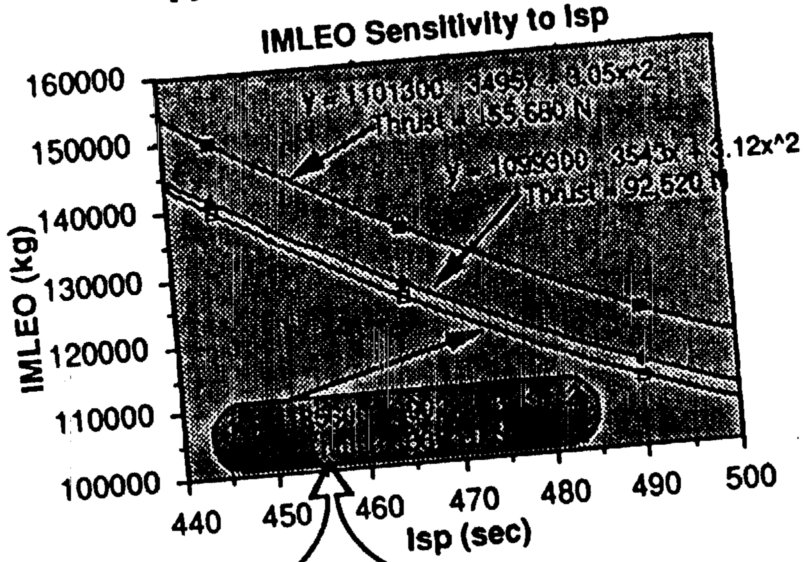
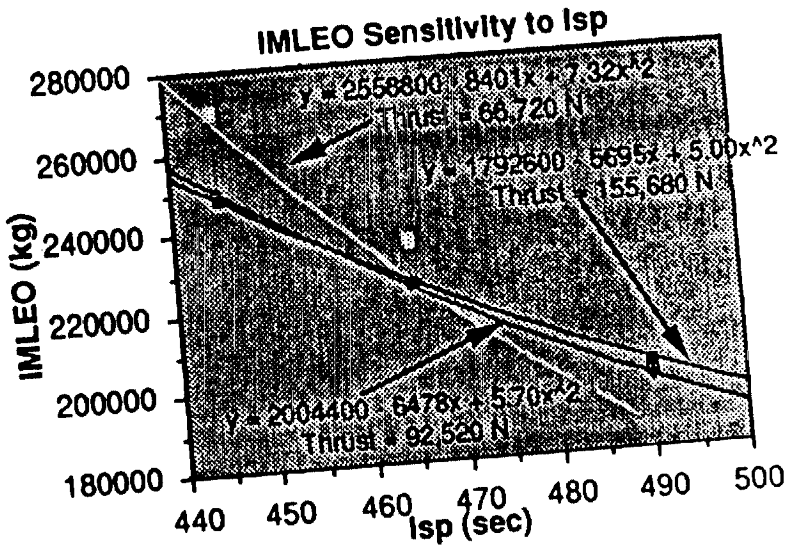
Engine Isp Influence on IMLEO

No Cargo Capability



15t Cargo Capability



These Curve-Fit Equations Will Be Used in the Vehicle Analysis Tool (VATSS)

Low-Thrust, High Isp Engine Provides Lowest IMLEO for the No Cargo Capability Vehicle

High Isp Engine Provides the Lowest IMLEO for a 15t Cargo Capability Vehicle (Thrust/Weight Not As Influential)

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Cryogenic Engines DOE Results

- General Results
  - Isp is the Most Influential Parameter on IMLEO
  - The Influence of Thrust/Weight Increases with Increased Thrust

- No Cargo Capability
  - A Low-Thrust, High-Isp Engine Provides the Lowest IMLEO (Lower Engine Weight for Lower Thrust Engine)
  - IMLEO Decreases as Thrust/Weight Decreases
  - Little to No Interaction between Isp and Thrust/Weight

- 15t Cargo Capability
  - A High-Thrust Engine Provides the Lowest IMLEO (High Thrust Overrides the Engine Mass) (Lower Thrust Engines Require More Propellant)
  - As Isp Increases, Thrust Level Becomes Less Important
  - At Low Isp, A High-Thrust Engine Provides the Lowest IMLEO
  - Little Interaction Between Isp and Thrust/Weight

- Both Engine Isp and Thrust/Weight will be Included in the Overall Vehicle Spreadsheet (VATSS)

Structures DOE Analysis Process

- Evaluated Structural Components of the STV Phase I Configuration
  - Core Structure, Aerobrake, Drop Tanks, Crew Cab, Core Tanks, Lander Legs and Drop Tanks Support Structure

- Evaluated Three Materials
  - Aluminum, Aluminum-Lithium and Composites (Graphite Epoxy)

- Maintained Same Design Configuration for All Materials
  - Did Not Optimize Component Design for Al-Li or Composites
  - Composite Sizing Based on Constant Material Properties, Not Adjusted for Ply Direction or Minimum Ply Thickness

- DOE L27 Matrix Used to Evaluate Combinations of the Seven Structural Components with the Three Materials
  - Response is the Vehicle Dry Mass
  - 15% Growth Factor Included in Dry Mass

- All Pressure Vessels Sized for Burst Pressure

Structural Component Mass Summary

- Structural Component Mass (kg) Based on Material Selection

Component	Aluminum $\rho = 2.85 g/cm^3$	Aluminum-Lithium $\rho = 2.70 g/cm^3$	Composites* $\rho = 1.80 g/cm^3$
Core Structure	6235	5078	4979
Aerobrake	5768	4521	4194
Drop Tanks	3965	[illegible]	[illegible]
Crew Cab	[illegible]	5290	[illegible]
Core Tanks	951	501	458
Lander Legs	239	178	105
Drop Tank Support Structure	7498	6305	6165

- Aluminum-Lithium Structure Reduces Component Dry Mass By 16 to 50%
- Composite Structure Reduces Component Dry Mass By 18 to 56%

* Composite Structure Not Optimized - Greater Mass Reduction Possible If Structure Redesigned

Structures DOE L27 Matrix

Analysis Run	Core	Aero	DT	Crew	C Tank	Legs	DT Str	Vehicle Dry Mass (lbm)
	Al / Al-Li / Comp	Al / Al-Li / Comp	Al / Al-Li / Comp	Al / Al-Li / Comp	Al / Al-Li / Comp	Al / Al-Li / Comp	Al / Al-Li / Comp	
1	Al	Al	Al	Al	Al	Al	Al	82149
2	Al	Al	Al-Li	Al-Li	Al-Li	Al-Li	Al-Li	85755
3	Al	Al	Comp	Comp	Comp	Comp	Comp	84116
4	Al	Al-Li	Al	Al-Li	Al-Li	Comp	Comp	67803
5	Al	Al-Li	Al-Li	Comp	Comp	Al	Al	65110
6	Al	Al-Li	Comp	Al	Al	Al-Li	Al-Li	70898
7	Al	Comp	Al	Comp	Comp	Al-Li	Al-Li	66840
8	Al	Comp	Al-Li	Al	Al	Comp	Comp	70328
9	Al	Comp	Comp	Al-Li	Al-Li	Al	Al	64681
10	Al-Li	Al	Al	Al-Li	Comp	Al-Li	Comp	67935
11	Al-Li	Al	Al-Li	Comp	Al	Comp	Al	66097
12	Al-Li	Al	Comp	Al	Al-Li	Al	Al-Li	70368
13	Al-Li	Al-Li	Al	Comp	Al	Al	Al-Li	65160
14	Al-Li	Al-Li	Al-Li	Al	Al-Li	Al-Li	Comp	67538
15	Al-Li	Al-Li	Comp	Al-Li	Comp	Comp	Al	62461
16	Al-Li	Comp	Al	Al	Al-Li	Comp	Al	74847
17	Al-Li	Comp	Al-Li	Al-Li	Comp	Al	Al-Li	59908
18	Al-Li	Comp	Comp	Comp	Al	Al-Li	Comp	59244
19	Comp	Al	Al	Comp	Al	Comp	Al-Li	67408
20	Comp	Al	Al-Li	Al	Comp	Al	Comp	70239
21	Comp	Al	Comp	Al-Li	Al	Al-Li	Al	66107
22	Comp	Al-Li	Al	Al	Comp	Al-Li	Al	75283
23	Comp	Al-Li	Al-Li	Al-Li	Al	Comp	Al-Li	61202
24	Comp	Al-Li	Comp	Comp	Al-Li	Al	Comp	59023
25	Comp	Comp	Al	Al-Li	Al	Al	Comp	65602
26	Comp	Comp	Al-Li	Comp	Al-Li	[illegible]	Al	61154
27	Comp	Comp	Comp	Al	Comp	Comp	Al-Li	66295
% Contribution	51.16	6.70	31.24	5.25	0.94	0.07	5.73	

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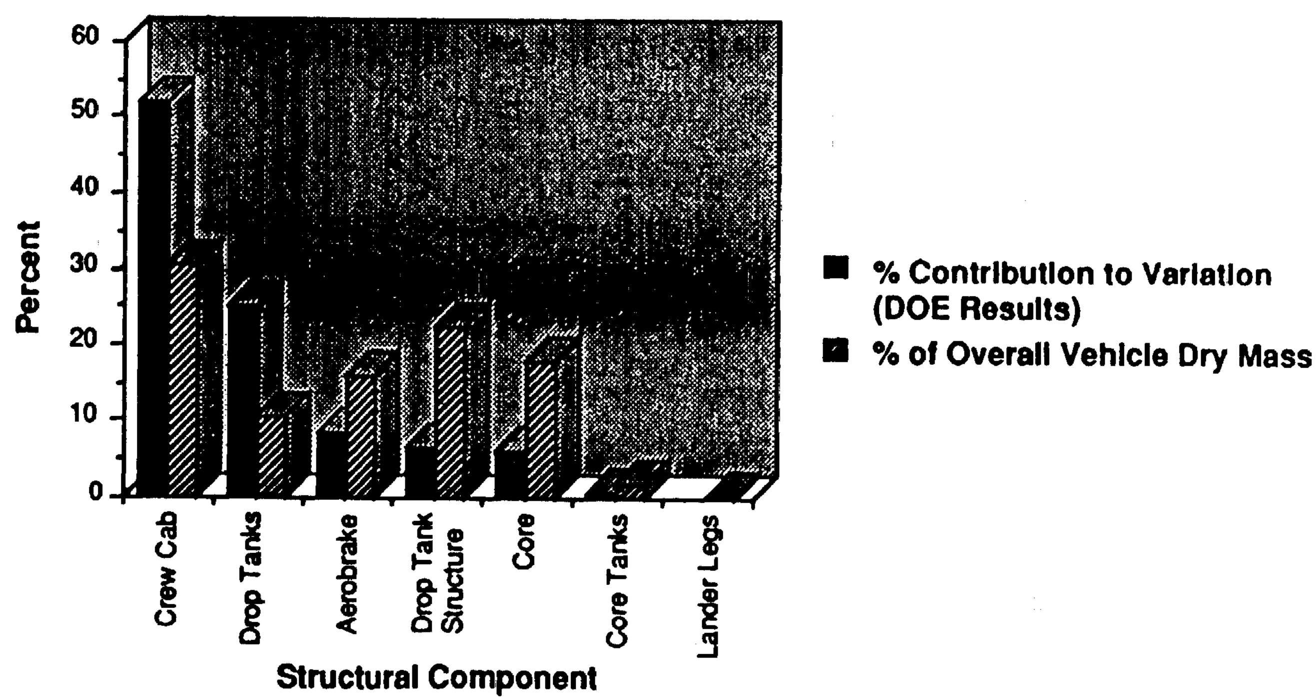
Structures DOE Analysis Results

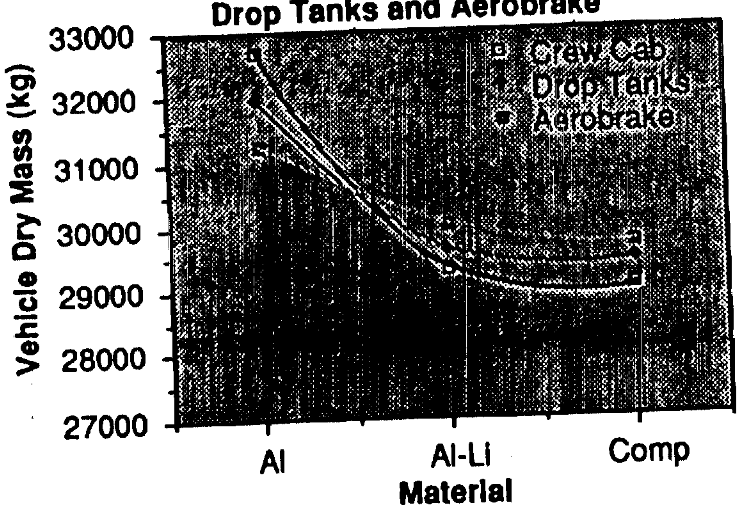
- DOE Reduced Number of Analysis Combinations from 343 to 27
  343 = 7 Components with 3 Combinations

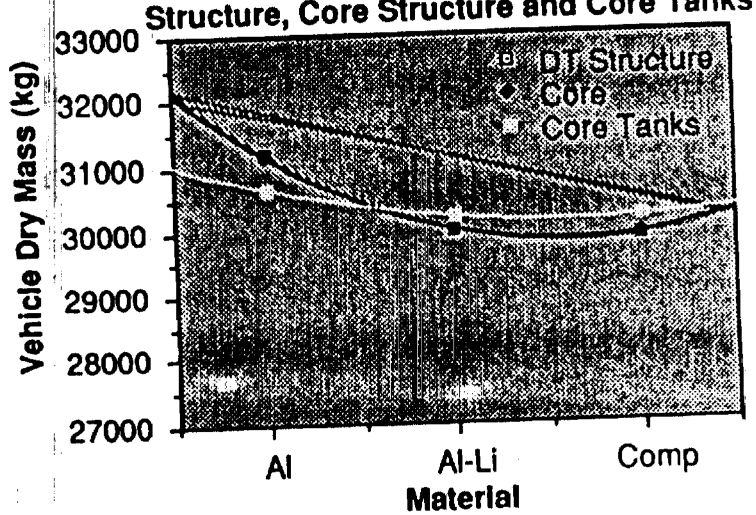
- Comparison of Component DOE Results to the Percent of Overall Vehicle Mass Indicates Which Component Was Influenced Most by Materials Change



Comparison of Structural Material Changes

- Comparison of Materials Change on Vehicle Components
  - Aluminum Structure Is the Heaviest Option
  - Overall Vehicle Dry Mass Reduced Approximately 28% By Using Advanced Structures
  - Vehicle Dry Mass Reduction Trends Illustrated in Graphs





Avionics DOE Analysis Process

Lunar Transportation System

- Avionics Advanced Technology Assessment Will Evaluate Six Parameters Plus Fault Tolerance
  - Vehicle Health Management (VHM)
  - Guidance, Navigation and Control (GN&C)
  - Autonomous Rendezvous, Proximity Operations and Docking
  - Software
  - Power Distribution System
  - Data Management System

- A DOE L16 Matrix Will Be Used to Evaluate the Importance of These Parameters on Vehicle Mass, Cost, Power and Volume Usage
  - DOE Reduces the Number of Runs from 49 (7^2) to 16

- Two Configurations for Each Parameter Will Be Identified
  - State-of-Art Avionics Architecture
  - Advanced Avionics Architecture
  - The Actual Architecture Components Will Be Identified as the Task Progresses

- An Avionics Spreadsheet Is Being Developed to Analyze the Various Architectures and for Future Detailed Analysis

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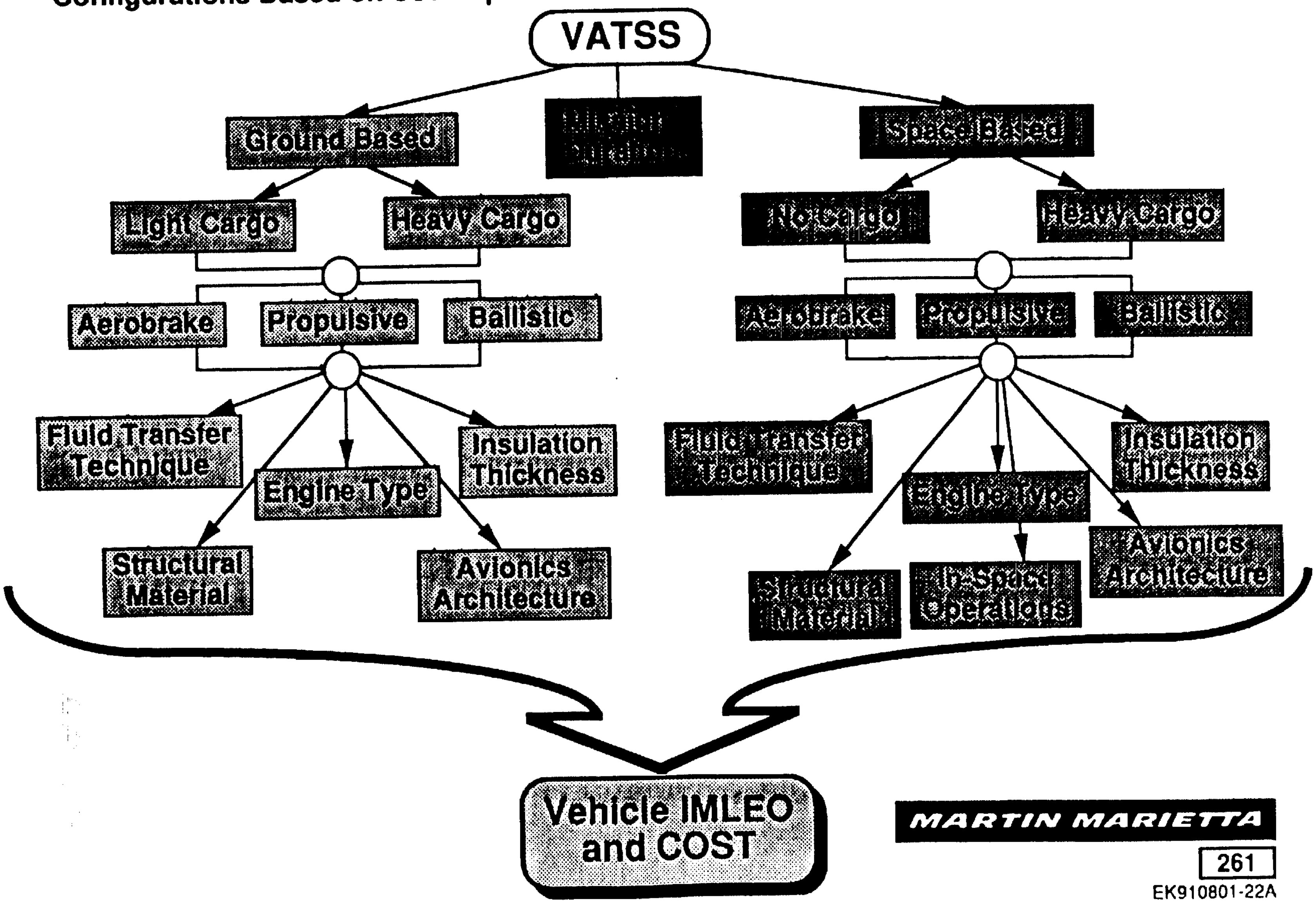
Vehicle Configuration Assessment Tool

- Vehicle Advanced Technology Sensitivity Spreadsheet (VATSS) Evaluates Vehicle Configurations Based on User Inputs and Default Parameters



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Summary

- Completed Several Technology Area Analyses
  - Cryogenic Fluid Management
  - Cryogenic Engines
  - Structures

- Remaining Technology Area Analyses in Progress
  - Avionics
  - Aerobrake
  - In Space Operations and Assembly

- Vehicle Cost Spreadsheet Being Developed

- Vehicle Performance Spreadsheet (VATSS) Being Developed
  - Primary Parameters and Spreadsheet Structure Being Defined
  - First Spreadsheet will be Applicable for Space Based Vehicle
  - Future Activities Can Complete Ground Based Vehicle Spreadsheet(s)

- DOE Has Been Useful in Reducing Workload
  - DOE Has Many Applications in Analysis Processes

- VATSS Will Allow Parametric Analysis of Vehicle Configurations for Impact of Advanced Technology on Cost and Performance

Technical Directive 08

Integrated Modular Engine Feasibility Study

Agenda

• Introduction and General Overview	M. Wakefield
• TLI Stage	
- Selected Design	M. Wakefield
- Logic Behind Selection	J. Greenwood
- Requirements Satisfaction	J. Greenwood
• Lunar Lander	
- Selected Design	M. Wakefield
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• Upper Stage	
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• Reliability Assessment	M. Wakefield R. Welborne
• Technology Plan	M. Wakefield
• Conclusions	M. Wakefield

MW920602-01

IME Program Objectives and Outline

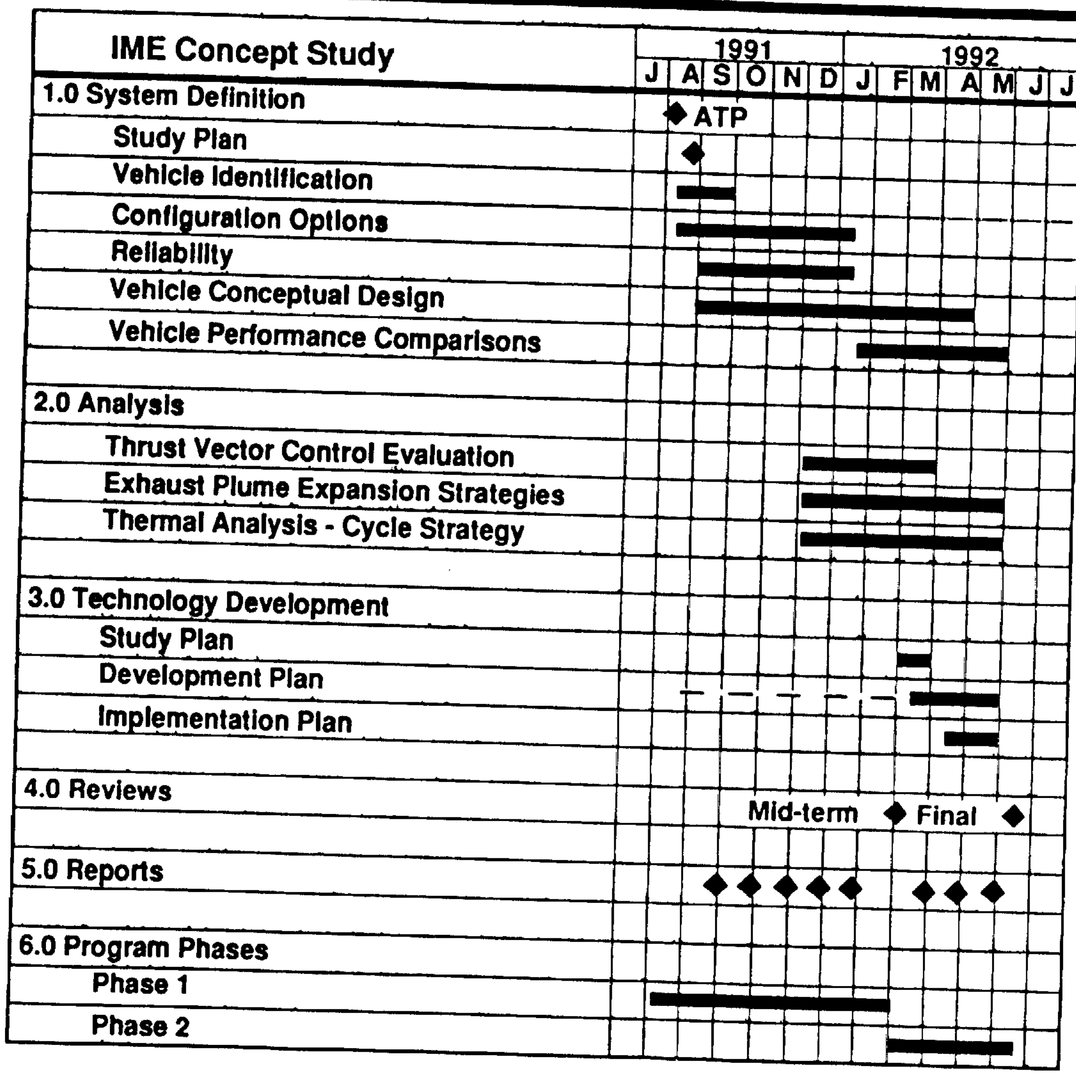
Primary Objectives

- Define Concepts for Space Vehicles Using IME
- Quantify Potential Benefits of the IME Concept
- Identify Issues That Must Be Resolved Prior to Development
- Define Technical and Programmatic Actions Necessary to Allow Development

Program Outline

- System Definition of Upper Stage, Lander, and Transfer Vehicles Using IME
  - Propulsion Options, Operating modes, Interfaces, Operations, Evolution
  - Comparison of Conventional and IME
- Analysis
  - Thrust Vector Control Issues
  - Exhaust Expansion Strategies
  - Thermal Analyses
- Technology Development
  - Technology Requirements and Plans

Program Schedule



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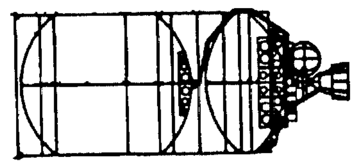
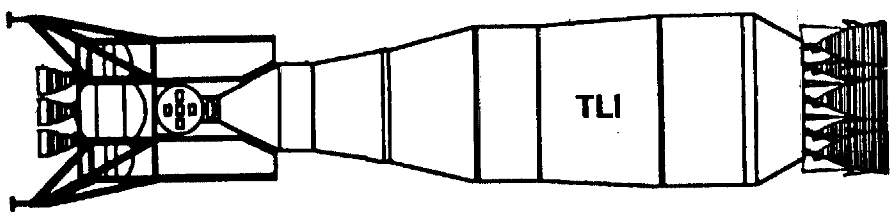
Mission Element Descriptions

	Vehicle	Characteristics	Data
Upper Stage	 Proposed Air Force US for NLS 3 (20Kto LEO)	Stage Engine (1x) Engine Weight Total Thrust	RL10A-4 365 lb 20800 lbf
Lunar Mission	 From TD-07, Rendevous and Docking Arch.	TLI Engine (5x) Engine Weight Total Thrust	RL10C - 1 800 lb 175,000 lbf
		Lander Eng. (5x) Engine Weight Total Thrust	RL10A - 4 365 lb 104,000 lbf

RS920122-01C

IME Matrix

Missions - Mission Characteristics - Requirements/issues		Primary IME Benefit		Additional IME Benefits
• TLI - Dual Fault Tolerant - Fixed IMLEO (how to best utilize) - Gravity Loss Sensitivity - 100 - 200 Klb Thrust - Space Storage - TVC - Number of Burns Sensitivity - Manned vs Unmanned	→	**Meet Fault Tolerance Rqt (w/Improved Reliability)**	+	**Increased P/L to Surface** - Eliminate Gimbal System Cost & Wt - Improved Isp if Use Stage Surface for Expansion - Shorter Interstage Allows More IMLEO for a Given Launch Vehicle
• Lunar Lander - Dual Fault Tolerant - Multiple Burns - Throttling - Fixed P/L - Space Storage - Landing Site Prepared or Unprepared Cargo needs to be Close to Surface Plume Dispersal Dust (or Wind on Mars) - Piloted & Cargo Missions - Thermal Isolation for Cryo - TVC	→	**Meet Fault Tolerance Rqt (w/Improved Reliability)**	+	**Reduced IMLEO Wt (or more cargo) (Reduced Cost)** - Eliminate "Fountain" at Landing - Cargo & Vehicle Closer to Surface - Lower C.G. - Improved Packaging Centerline Thrust Compact Engine - T/W & Isp Allow More Cargo (or lighter vehicle) - Eliminate Gimbal System Wt & Cost
• Upper Stage - Single Engine - 20-40 Klb Thrust Rqt - Gravity Loss Sensitivity - Mission Flexibility, e.g. LEO - Single Burn GEO - Multiple Burns - TVC - Fault Tolerance Issue - Unmanned	→	**Improve Reliability (w/Weight Penalty)**	+	**Reduced Ops Cost** - Elimination of Hydraulics - Elimination of Gimbal System - Increased Component Accessibility - T/W & Isp Allow More Cargo (or lighter vehicle)

TLI Stage

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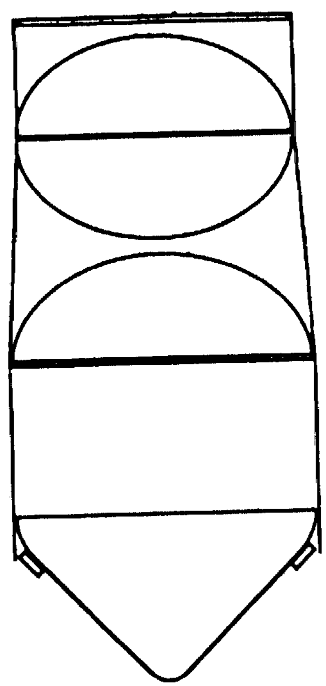
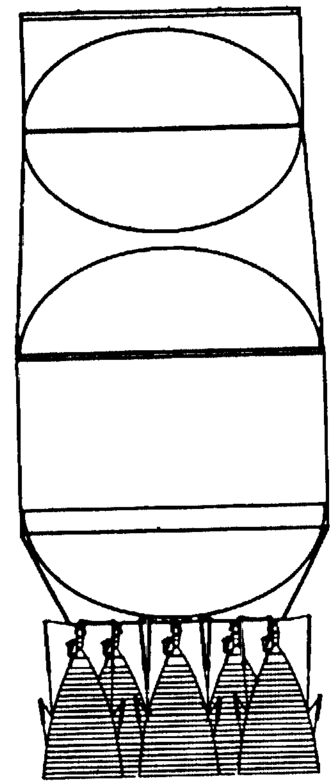
Selected TLI Design vs Reference Configuration





Plug Cluster IME with Conical Plug

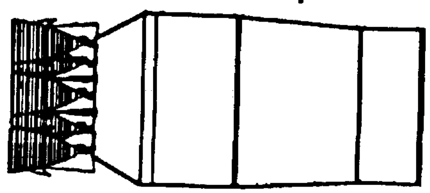
175 Klb Thrust
473 Sec Isp

Reference Configuration RL10-C Engines (5)

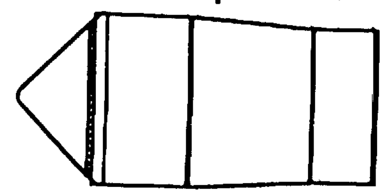
175 Klb Thrust
468 Sec Isp

IME vs Conventional Engine - TLI Stage

Conventional Propulsion



IME Propulsion



Baseline

TLI Stage Propellant Tank Volumes and Flight Profile are Identical for Conventional and IME Engine Configurations. Engine Thrust Has Been Maintained at the Same Level. The Aft Tank (Hydrogen) Shape Has Been Modified to Enhance the IME.

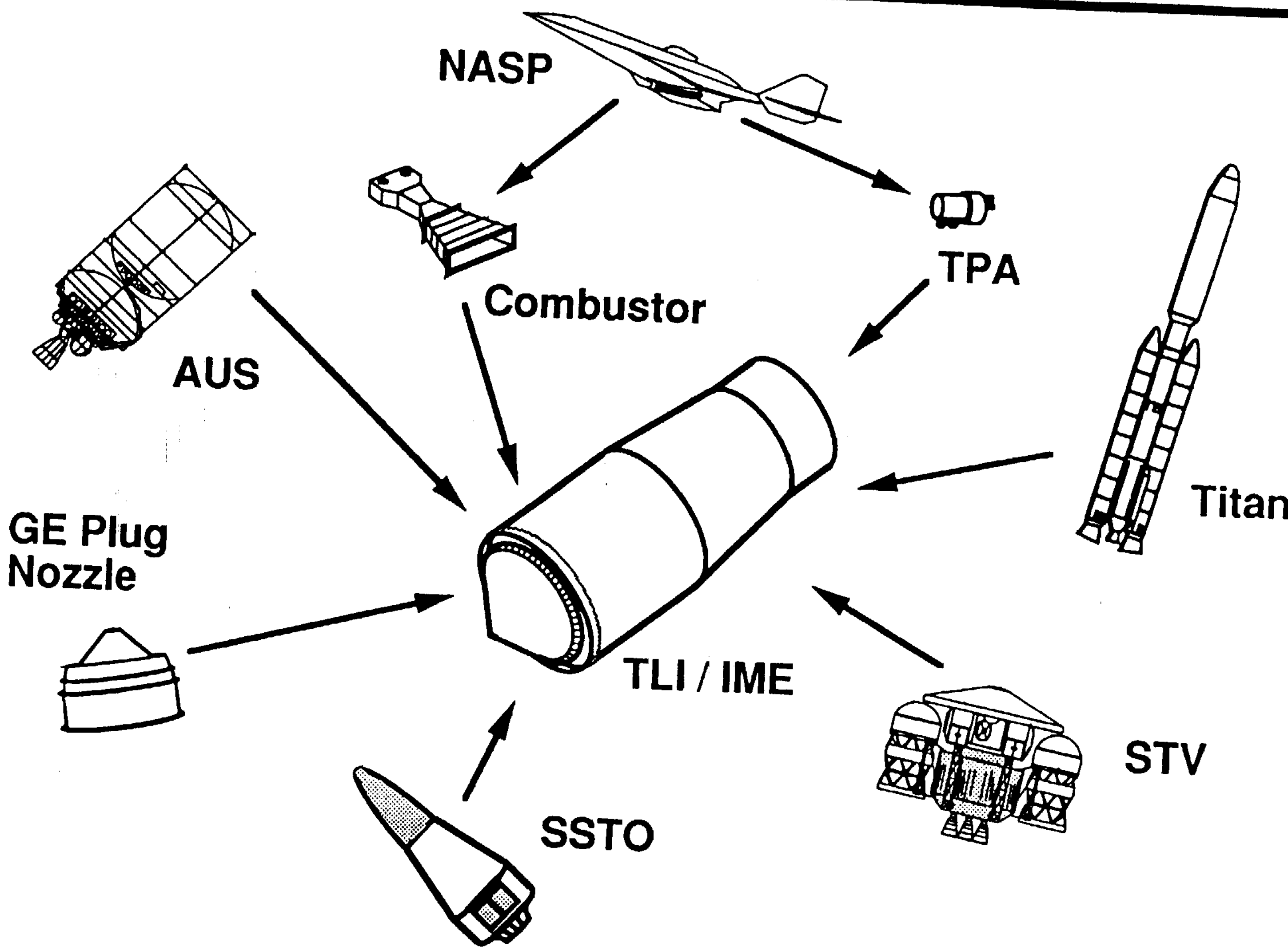
Characteristics	Conventional Propulsion (5 Engines, RL10C)	IME Propulsion (64 Combustors, 4 TPA's)	IME Discriminators
Safety	Dual Fault Tolerance	Dual Fault Tolerance +	Failure Thrust Higher
Reliability (Firings/non-fire)	0.9994 (1667)	0.9988 (833)	Both Excellent, but IME More Recovery Options
Payload, Tonnes	70.2	72.6	+ 2.4 Tonnes
Isp, Seconds	468	473	+ 7 Sec
Health Monitoring	Could be Incorporated	Integral	VHM at Inception
Stage Length, Meters	18.3	16.2	- 2.1 Meters
Thrust Vector Control	Gimbals	Thrust Mod and GG Exh	Eliminate Gimbal System, Replaced with Avionics
Development Cost, $	300M	See P. 211	-
Production Cost, $	4.5X5 =23M	9.0M	$14M Less

Notes: 1. RL10-C Is Conventional Engine Baseline. Engine Is not Developed, so P&W Predicted Performance Characteristics are Assumed.
2. All Costs are ROM

TLI Selected Design

- Plug Nozzle Engine Configuration
  - Same Thrust Level (175Klb) as Reference Configuration
  - 473 sec Isp vs 468 sec for Reference Configuration
  - Pc = 1500
  - 64 Combustors
  - 2734 Pounds Thrust Contribution Per Combustor
  - 30:1 Initial Expansion
- Conical Bottom Tank Dome Serves as Nozzle Expansion Surface
- TPS System Provides Protection for Cryo Tank Structure
  - Carbon/Silicon Carbide Face Sheet with High Temperature Insulation
- Simple Engine/Vehicle Physical Interface
- Fault Tolerance Capabilities
- 2.4 Tonne Net Performance Increase

Integrated Modular Engine Heritage



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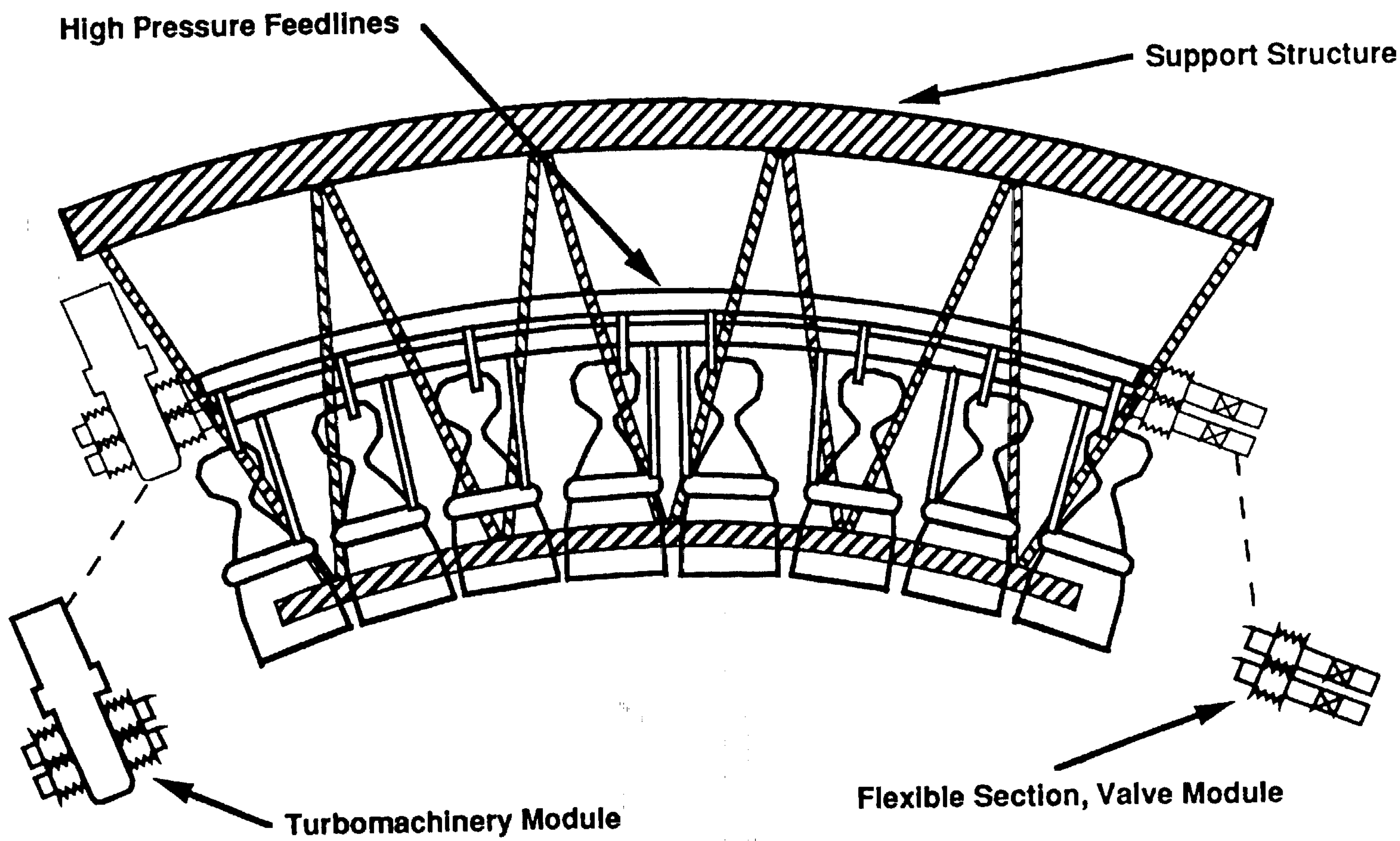
TLI Engine Module Plan View



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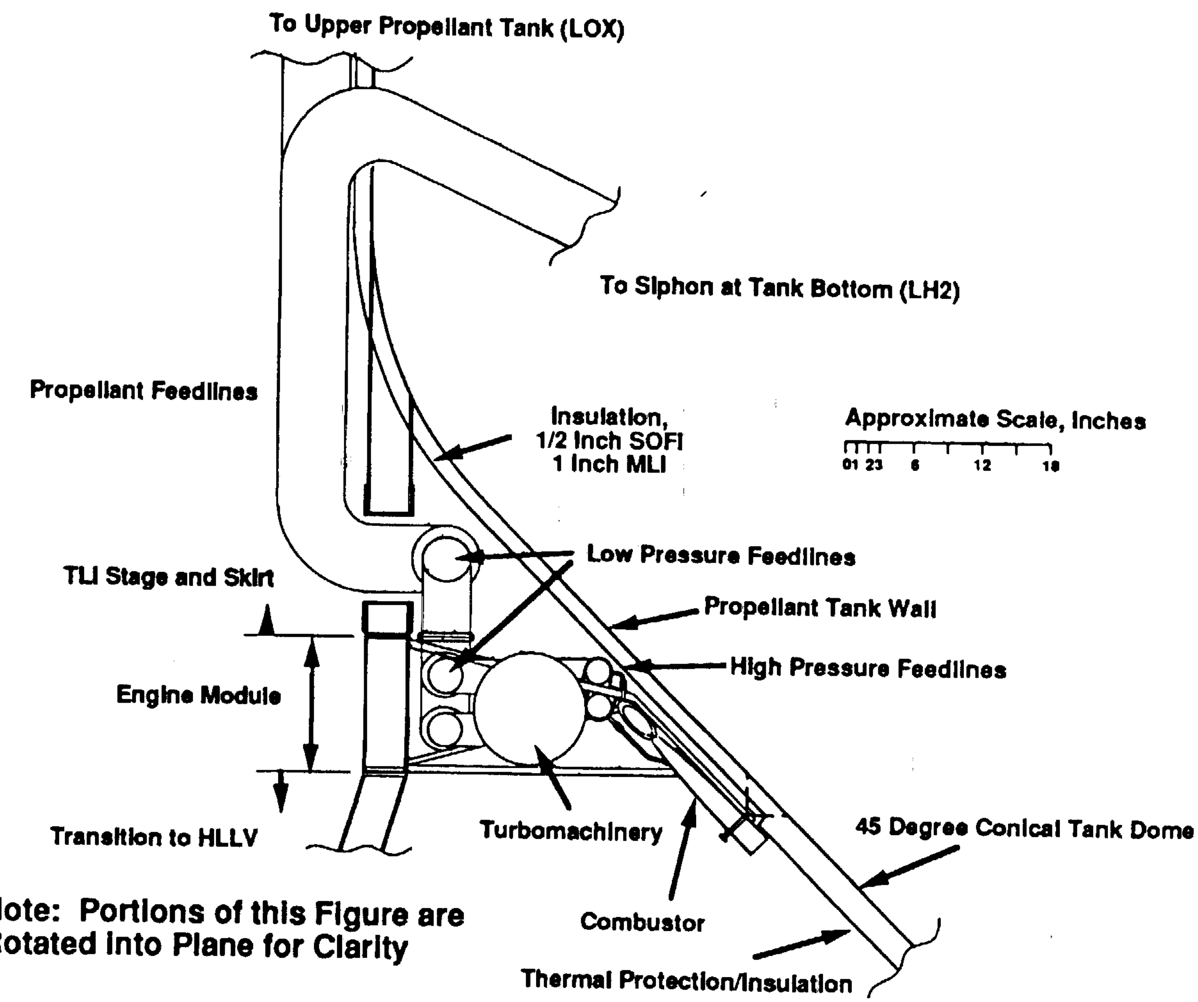
IME Installation on TLI Stage



Note: Portions of this Figure are Rotated Into Plane for Clarity

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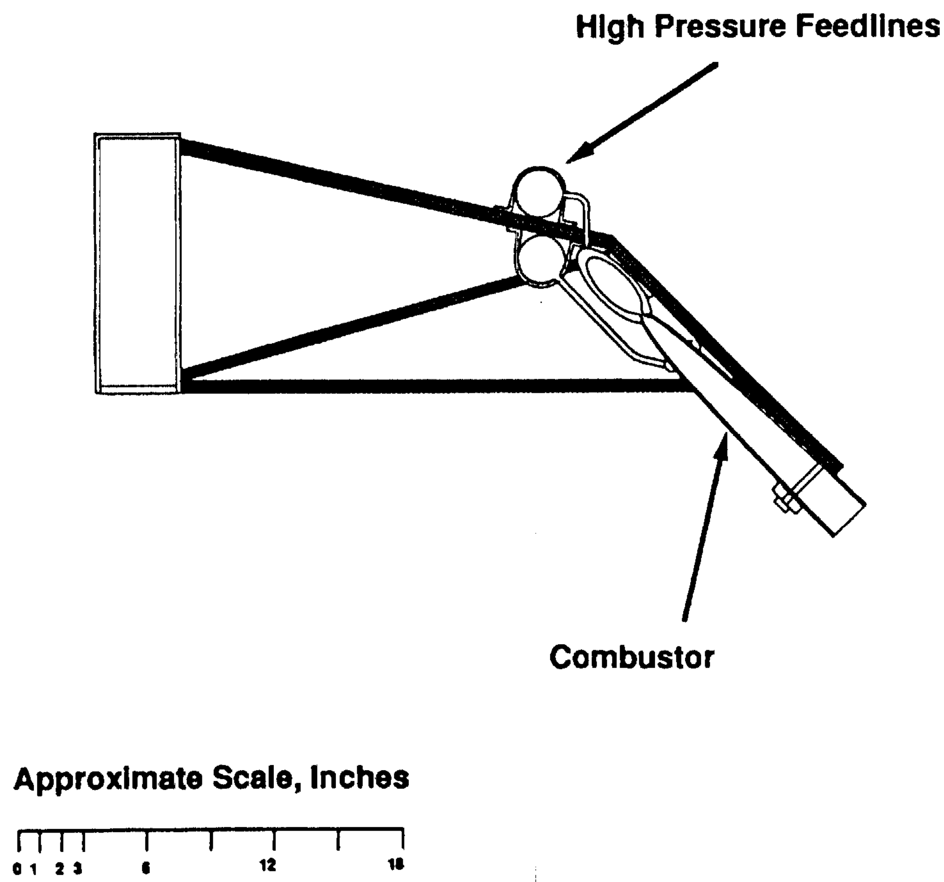
TLI Engine Module



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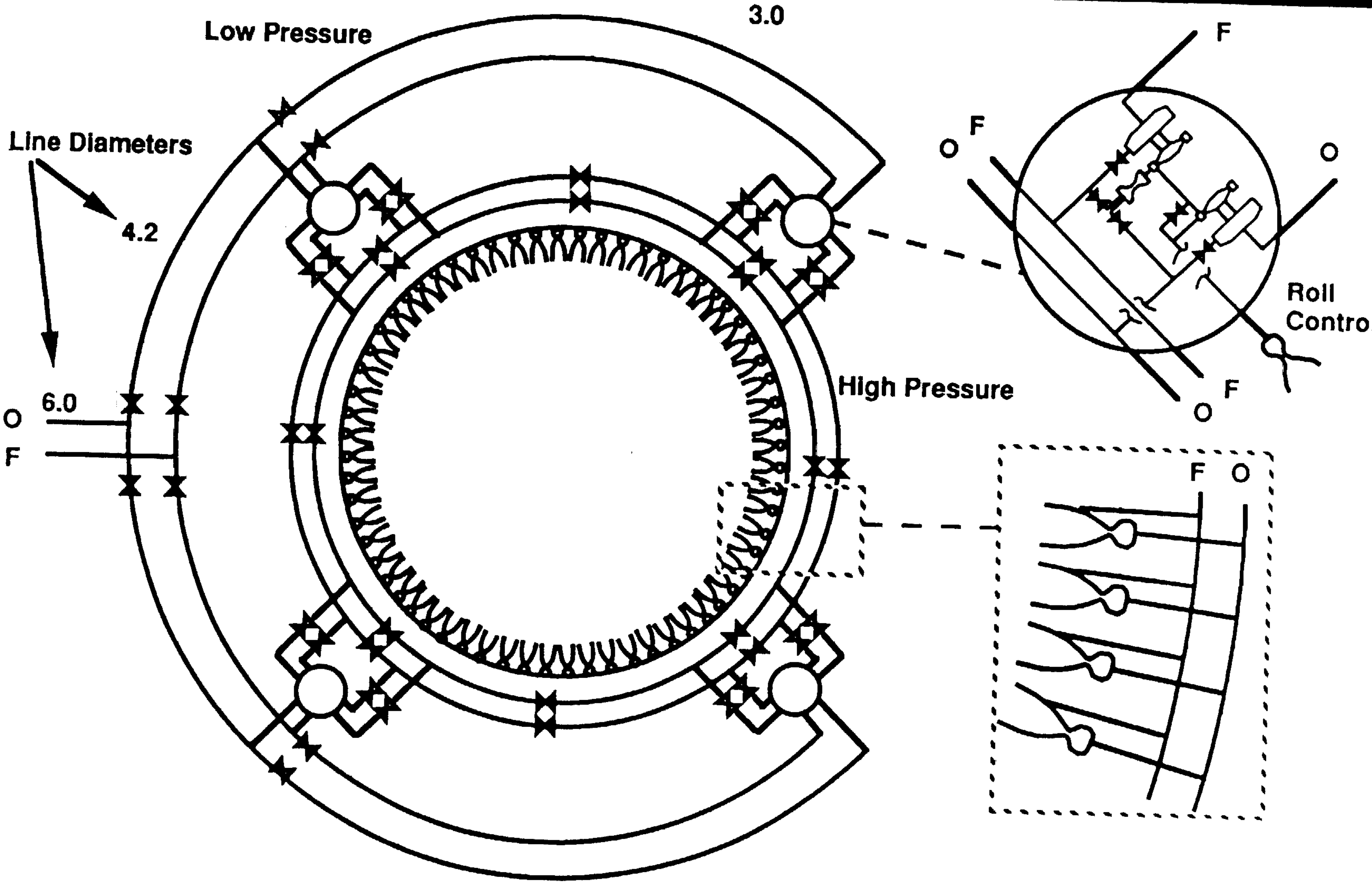
TLI IME Schematic



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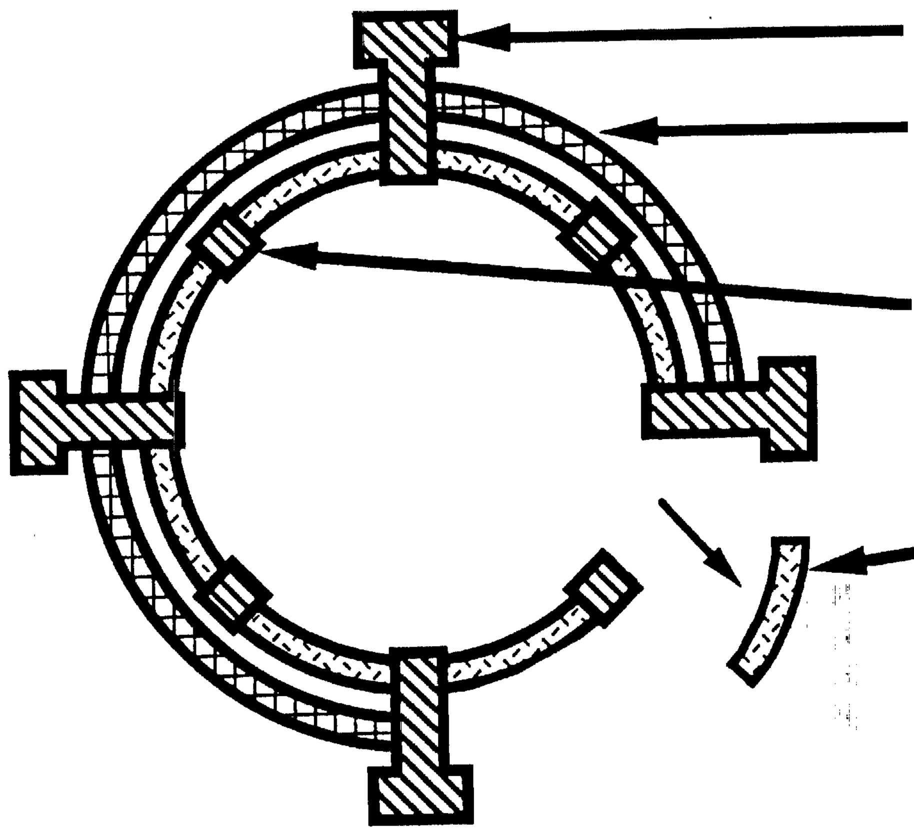
TLI Modular Component Groups



Bottom View of Engine

MW920421-01

Agenda

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- Requirements Satisfaction	J. Greenwood
• Reliability Assessment	R. Welborne
• Technology Plan	M. Wakefield
• Conclusions	M. Wakefield

Logic Behind Selection - TLI Stage

- Summary
- Stage Weight
- Engine Weight
- Engine Cycle
- Manufacturing/Integration
- Thrust Vector Control
- TPS Weight (Tank Bottom/Nozzle Surface)
- TLI Performance vs. Pc
- Net Performance Results

TLI Stage Weight

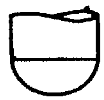
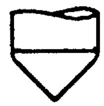
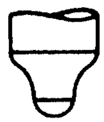
TLI Stage Weight is Considered as a Function of:

- Aft Dome Weight
- Aft Tank Barrel Section Length
- Engine Thrust Structure
- Interstage Length

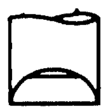
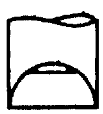
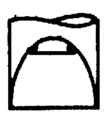
Parametric Evaluations are Presented That Relate These Weights

Plug vs E-D - Dome Structure Weights

Convex Domes

		Calculated Dome Weight	Additional Cylinder Wt for TLI Stage		Net Weight Impact
			(lb)	(%)	
$\sqrt{2}$ Ellipse		1,100 lb	0	0%	0 (Reference)
Cone/Ellipse		2,600 lb	1,000 lb	17%	2,500 lb
Aerospike		2,400 lb	500 lb	8%	1,800 lb

Inverted Domes

		(Isogrid Weights)			
$\sqrt{2}$ Ellipse		4420 lb	5,700 lb	95%	10,120 lb
Hemisphere		3400 lb	6,000 lb	100%	9,400 lb
Hemi/Parabola		6,800 lb	7,000 lb	117%	13,800 lb
Semi-Submerged		5440 lb	1,000 lb	17%	6,440 lb

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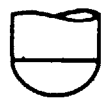
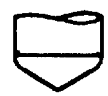
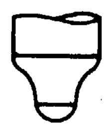
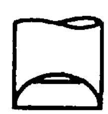
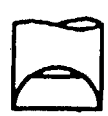
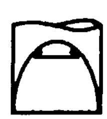
Plug vs E-D - Dome Structure & Engine Nozzle

Convex Domes

		Structure Considerations	Nozzle Considerations
$\sqrt{2}$ Ellipse		- Most Desireable Shape	- Flow Turns Wrong Direction - Complex Flow Analysis - Possible High Local Heat Transfer
Cone/Ellipse		- Moderate Weight Impact - Simple Shape to Manufacture	- Minor Efficiency Loss - Straightforward Analysis - Low Heat Transfer
Aerospike		- Minor Weight Impact - Complex Shape	- Minimal Efficiency Loss - High Heat Transfer - Possible Shock/Severe Heat Transfer

Inverted Domes

$\sqrt{2}$ Ellipse		- "Least Objectionable" Shape - Compression Loads - Buckling - Requires Isogrid or Sandwich Construction Methods - Large Weight Penalty	- Worst Contour for E-D (decreasing radius of curvature) - Severe Heat Transfer (Shocks)
Hemisphere		- All of Root 2 Problems Plus: - Longer Barrel Length - Extra Sensitivity to Buckling (due to constant radius)	- Tolerable Nozzle Contour (constant radius of curvature) - High Heat Transfer - Possible Shock/Severe Heat Transfer
Hemi/Parabola		- All of Root 2 Problems Plus: - Much Longer Barrel Length	- Optimum E-D Contour (increasing radius of curvature) - Medium Heat Transfer - Possible Shock/Severe Heat Transfer

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Plug Cluster vs Expansion-Deflection - Tank Wt

Key Observations:

- All Inverted Dome Concepts Incur Large Weight Penalties

- All Inverted Dome Concepts Have Potential for Severe Heat Transfer

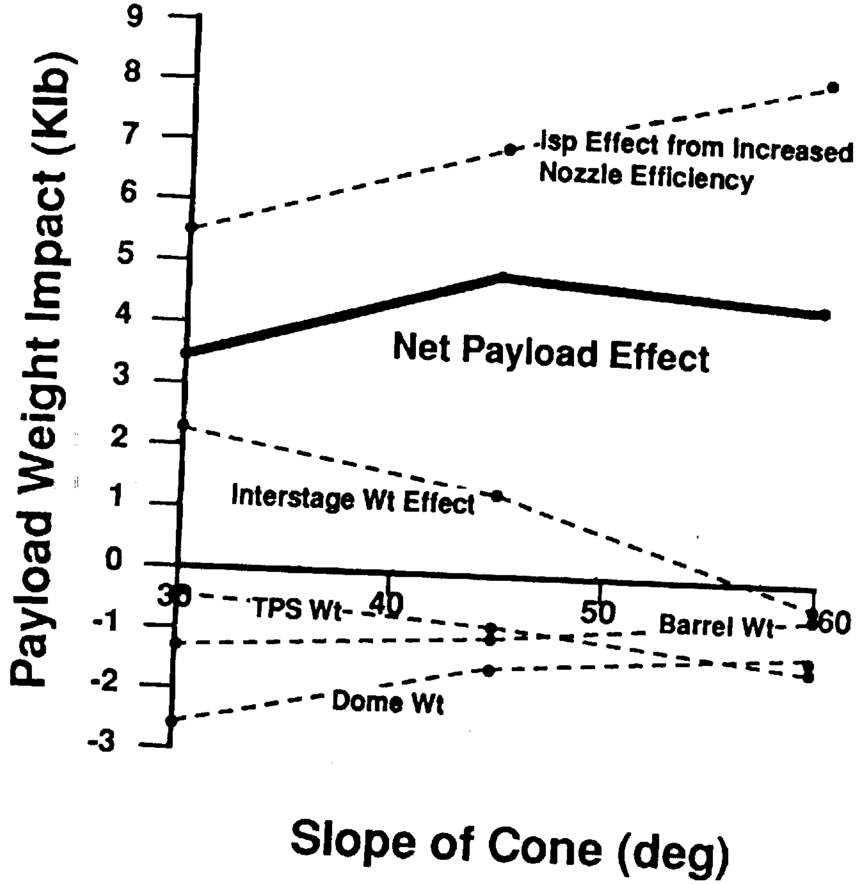
- Aerospike-Shaped Dome:
  - Has Least Weight Impact Relative to Root 2 Dome
  - Has High Complexity
  - Has Possible Severe Heat Transfer Problems

- Cone/Ellipse:
  - Weight Impact is Moderately More Than Aerospike Shape
  - Simple Shape to Manufacture
  - Low Heat Transfer
  - Minimal Efficiency Loss Relative to Ideal Nozzle Shape (with moderate Thruster AR)
  - Straightforward Nozzle Analysis

Conclusion:

- Cone/Ellipse is the Selected Aft Tank Dome Shape

Payload Weight Impact vs Cone Angle



Sensitivities

1 lb Dome Wt = 1 lb Payload Impact

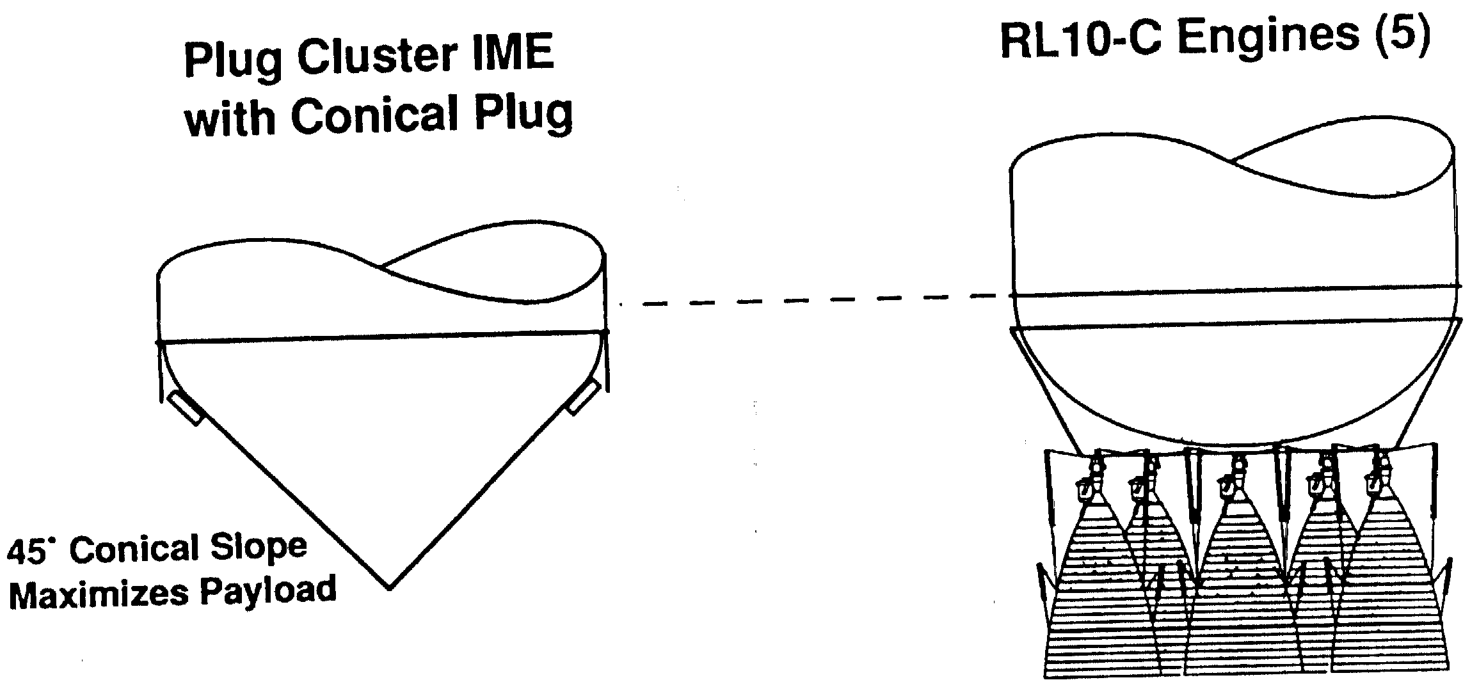
1 lb Barrel Wt = 1 lb Payload Impact

1 lb TPS Wt = 1 lb Payload Impact

1 lb Interstage Wt = 0.39 lb Payload Impact

1 sec Isp = 1000 lb Payload Impact

Final Aft Dome Design Compared to Conventional



Thrust Structure Wt
2000 lb

Thrust Structure Wt
3,600 lb

Net Reduction in Stage Length is 7 Feet. Note that this Considers a Longer Barrel Section, and a Shorter Engine which Includes the Trust Cone.

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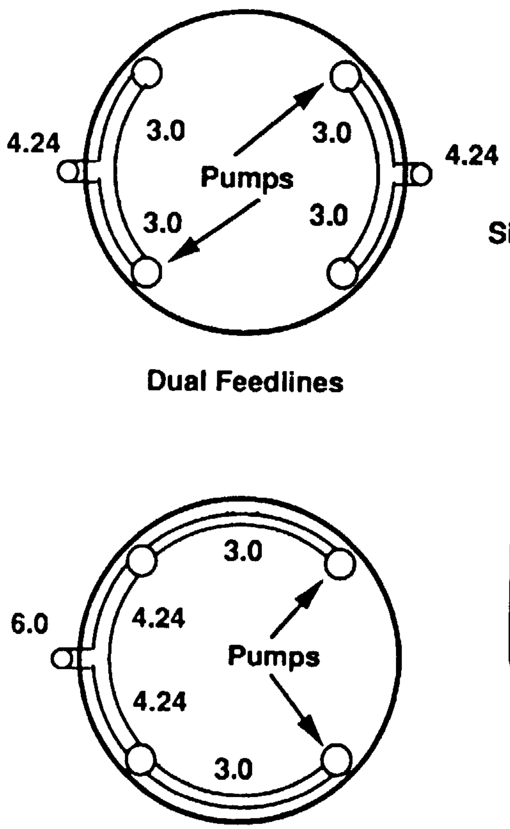
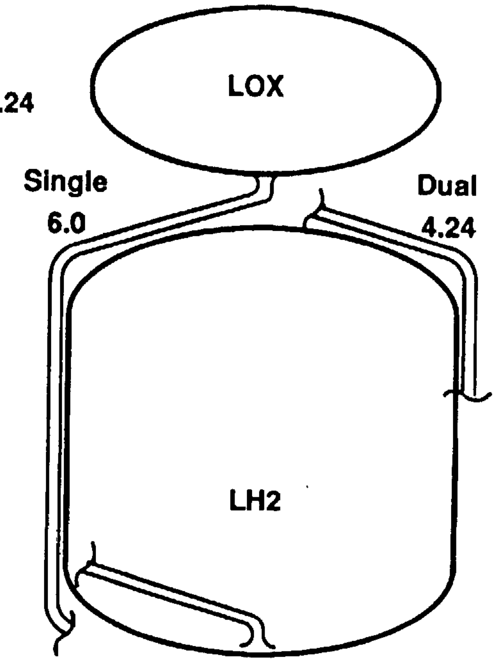
Feedline Weight Trade



Dual Feedlines

Single Feedlines



Feedline Routing at Tanks

Feedline Routing to Pumps

Relative Feedline Weight

(6.0 inch dia line is Relative Weight 1.0 for 1.0 Meter Length)

	LH2	LOX	Total
Single	18.24	24.24	42.48
Dual	17.32	25.79	43.11

Single Feedline System is Somewhat Lighter and is Simpler

Feedline Diameters are Shown in Inches

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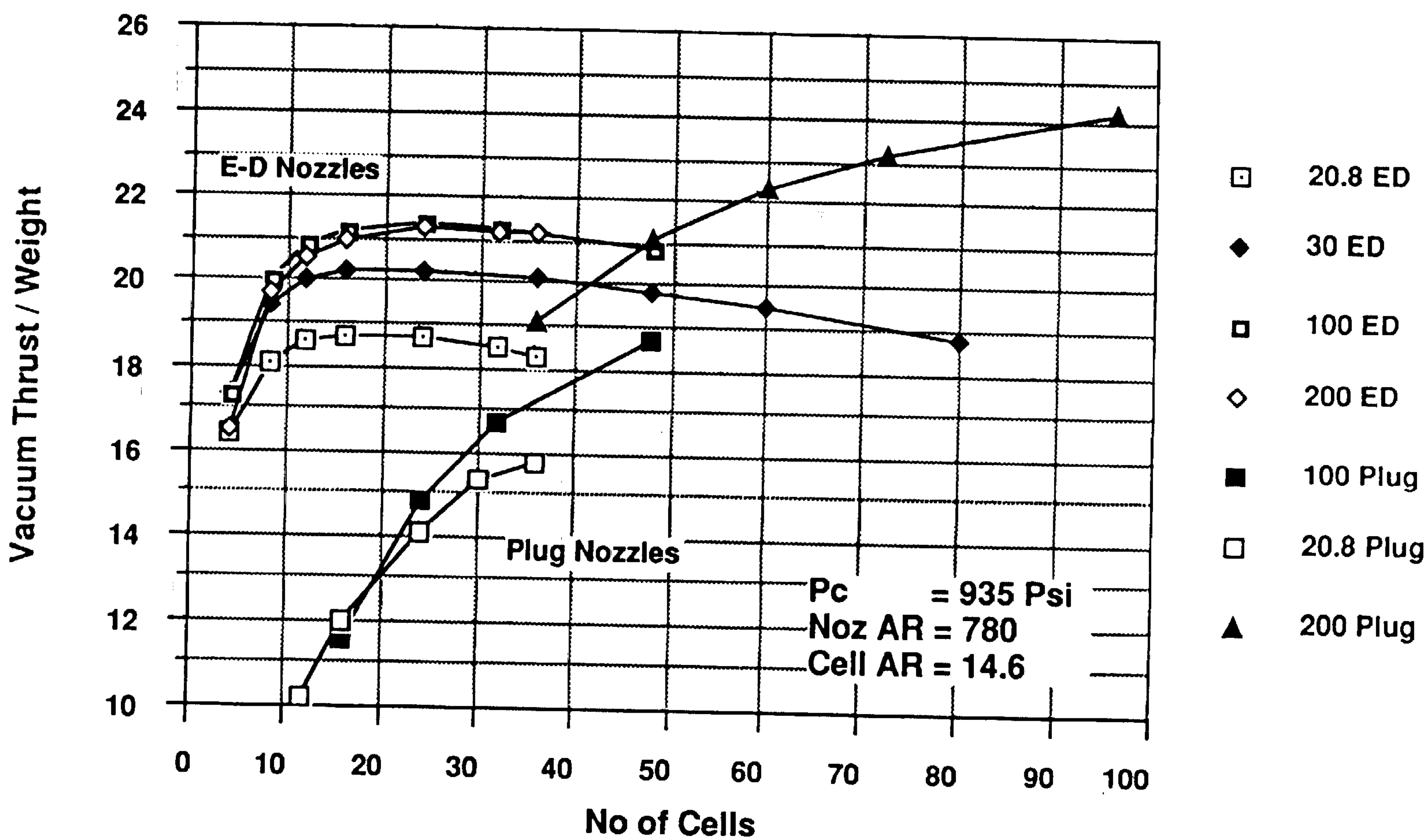
TLI IME Engine Weight

- Performance and Weight Parametrics are Presented for Plug Cluster and E-D Nozzle Engines.
- The TLI IME Application is Discussed

Plug and E-D Nozzle Parametrics for TLI



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Plug Cluster vs Expansion-Deflection - Engine Wt

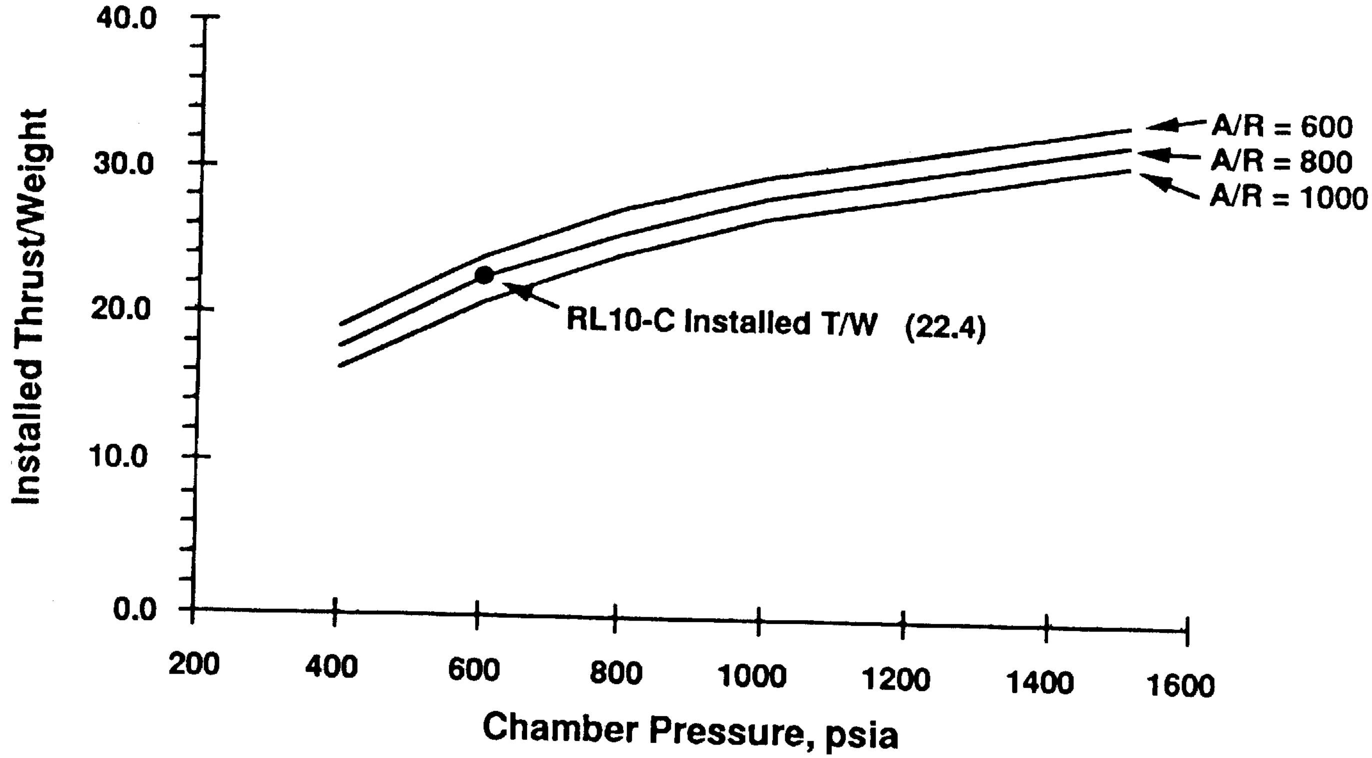
Data

- E-D Engine is Lighter (Higher T/W Ratio) for Smaller Engines
- E-D Engine is Lighter for Smaller Numbers of Thrusters
- Plug Cluster Engine is Equal or Lighter than E-D for Large (200 Klb Class) Engines with Many (50+) Thrusters

Conclusions (Based on Engine Wt Alone)

- E-D is Significantly Better for a Small (30Klb) Upper Stage Engine
- E-D Ranges from Somewhat to Significantly Better than Plug for a 100 Klb Class Lunar Lander Engine, Depending on # of Thrusters
- Plug Cluster ranges from Equal to Somewhat Better than E-D for a Large (175Klb) TLI Engine with 50+ Thrusters

Installed Thrust/Weight for TLI Vehicle



RL10-C

35 Klb Thrust/Engine
5 Engines (175 Klb Thrust)
800 lb Wt/Engine
40 lb Gimbal Sys Wt/Engine
3600 lb Total Thrust Struct Wt
468 sec Isp

IME

175 Klb thrust
1 Engine - 64 Cells
(without Plug Wt)
2000 lb Thrust Struct Wt
750 lb TPS Wt
475 sec Isp (@ A/R=900)

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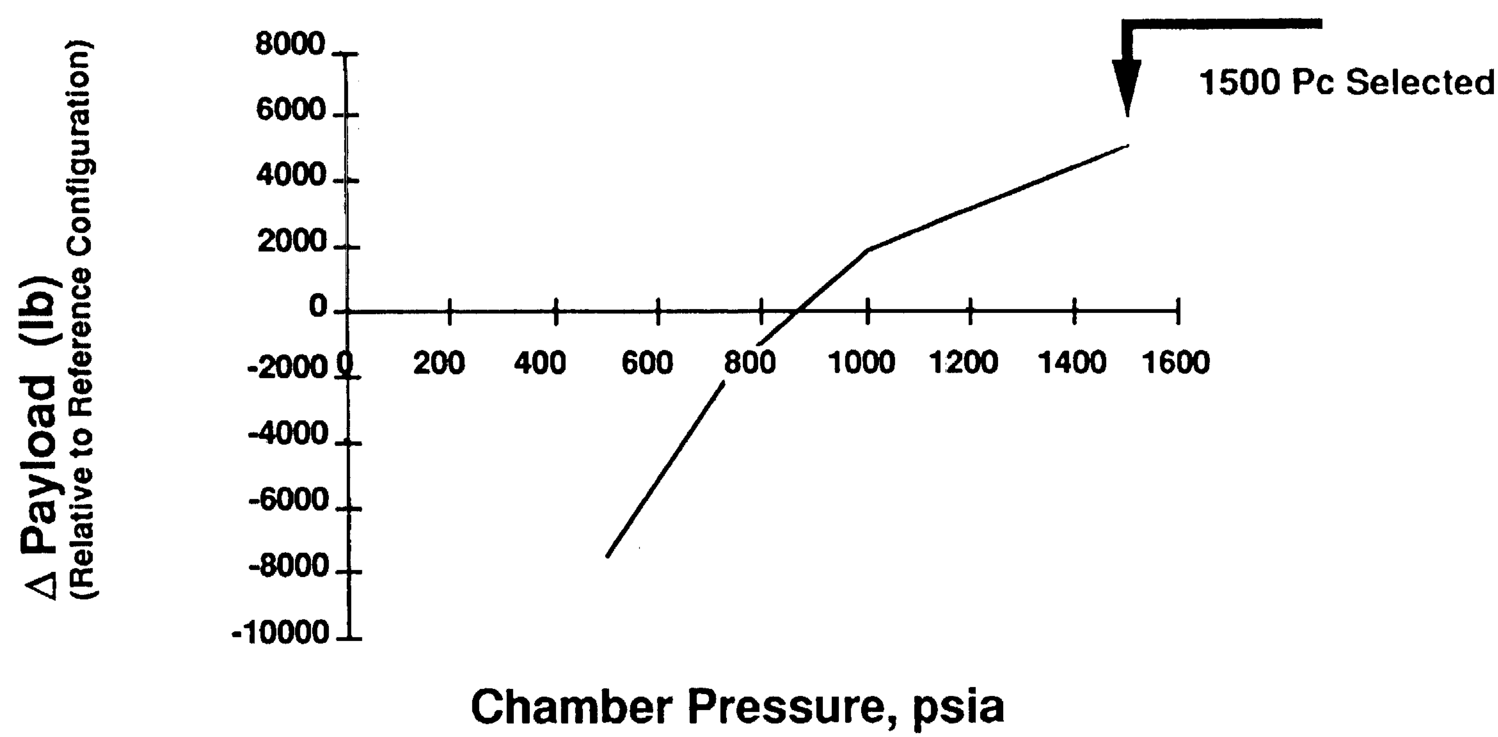
TLI Performance vs Pc: Results



Factors Considered:	
Isp Effects	**Weight Effects**
Isp vs Pc (for fixed Thrust & Total Exit Area) Isp Losses from 45° Conical Plug Isp Losses due to Gas Generator Cycle vs Pc	Engine Wt vs Pc Conical Dome Weight TPS for Dome/Nozzle Surface Elimination of TVC Weight Shortened Interstage

Engine Selection for TLI Stage - Plug vs E-D

Selection: Plug Cluster Engine with Cone/Ellipse Aft Dome

Rationale:

- **Engine Weight**
  175 Klb Thrust, 64 Thrusters
  - Plug Engine Has Slightly Better Thrust-to-Weight Ratio than an E-D (Plug T/W = 22 vs E-D T//W = 21)
- **Tank Weight**
  Dome & Barrel Section Wt
  - E-D Weight Penalty is approx. 10,000 lb more that Plug Weight Penalty
- **Aft Dome**
  Structure & Nozzle Considerations
  - Minimal Nozzle Efficiency Loss
  - Low Heat Transfer
  - Simple Manufacturing

TLI Engine Cycle Selection

- The Pros and Cons of the Gas Generator Cycle, Expander Cycle, and Staged Combustion Cycle are Discussed for TLI IME Application.

Gas Generator Cycle - Pros/Cons Relative to IME

Pro

Components are Relatively Independent of Each Other - No Complex Interactions

- Reduced Development Costs Because Components Can be Tested Separately (smaller individual components also reduce test facility costs)
- Engine Performance Relatively Unaffected by Changes in Numbers of Components

Simple High Pressure Plumbing

- Simplest Possible Arrangement, One Line from Pump to Chamber for Each Propellant

Simple, Positive Start

Throttleable

Con

Multiple Combustion Devices (GG's) Causes Ignition System Complexity

High Turbine Thermal Stresses at Startup

Expander Cycle - Pros/Cons Relative to IME

Pro

Combustion in Thrust Chambers Only - Reduced Ignition Requirement
Gentle Thermal Transient & Running Conditions on Turbine
Demonstrated Throttle Capability

Con

Low Start Margin

Component Interactions - Performance of Each Component Dependant on Function of Other Components
Affects Ability of Engine to Accommodate Component Failures

Multiple Lines Between Turbopump & Chamber (2X or 4X Lines)

Split Expander
Same Basic Pro's & Con's - Multiple Line "Con" is Slightly Worse Than Simple Expander

Dual Propellant Expander
Same Basic Pro's & Con's - Multiple Line "Con" is Worse Than Simple Expander

Dual Augmented Expander

Pro - More Positive Start, Component Interdependence is Reduced Somewhat
Con - Many Combustion Devices, Very Many Lines of High Pressure Plumbing

Staged Combustion Cycle - Pros/Cons Relative to IME

Pro

Positive Start

Demonstrated Throttling

Con

Very High Level of Component Interdependence & Complex Component Interactions

- All Tests Must be Engine-Level, Component-Level Tests not possible Without Very Expensive Ground Facilities to Simulate Inlet Conditions, Transients, & Feedback Effects
- Staged Combustion Modular Engine Would be Very Sensitive to Condition Changes Caused by Failed Components

Multiple Combustion Devices - Complex Ignition System Requirements

Severe Thermal Stress on Turbines at Engine Start-Up

Engine Cycle Selection

Cycle Selected: Gas Generator

Rationale:

- Simplest Plumbing
- Positive Start
- GG Exhaust Available for Roll Control Use

Observation:

A Simple Expander Cycle, While Not the Absolute Optimum, Is Still a Reasonable Choice (Benign Thermal Environment). Use of AETB Expander Cycle Hardware Components Adapted to GG Cycle Could Significantly Reduce Development Costs of an IME.

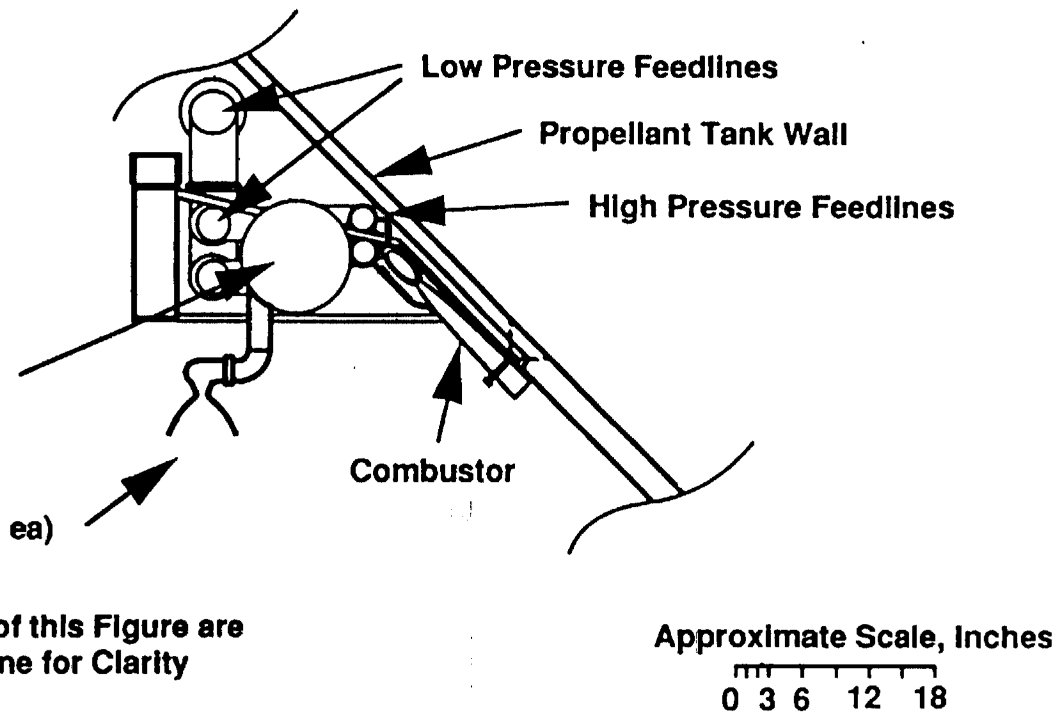
TLI Stage IME Thrust Vector Control

- Thrust Vector Control for the TLI IME Engine is Discussed. Subjects Include:

  - Implementation of Pitch, Yaw, and Roll Control
  - Throttle Response Required as a Function of Propellant Slosh
  - Available Control Authority

- Conclusions are Made For the TLI IME Application

TLI Roll Control



Note: Portions of this Figure are Rotated into Plane for Clarity

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Thrust Vector Control - Roll Control

- **Approach**: Swiveled Nozzle on Gas Generator Exhaust

  - GG Exhaust Provides 0.76% of Total Engine Thrust
    (4 Roll Thrusters @ 330 lb ea. for a 175 Klb Thrust Engine)

  - 4 Nozzles Swiveled to Opposing 90° Positions Provide 17,160 ft-lb of Roll Torque
    (4 x 330 lb x 13 ft Moment Arm = 17,160 ft-lb)

  - Titan Stage 2 Roll Torque:
    525 lb x(35° limit due to hydraulic actuator) x 3 ft Moment Arm = 903 ft-lb Roll Torque

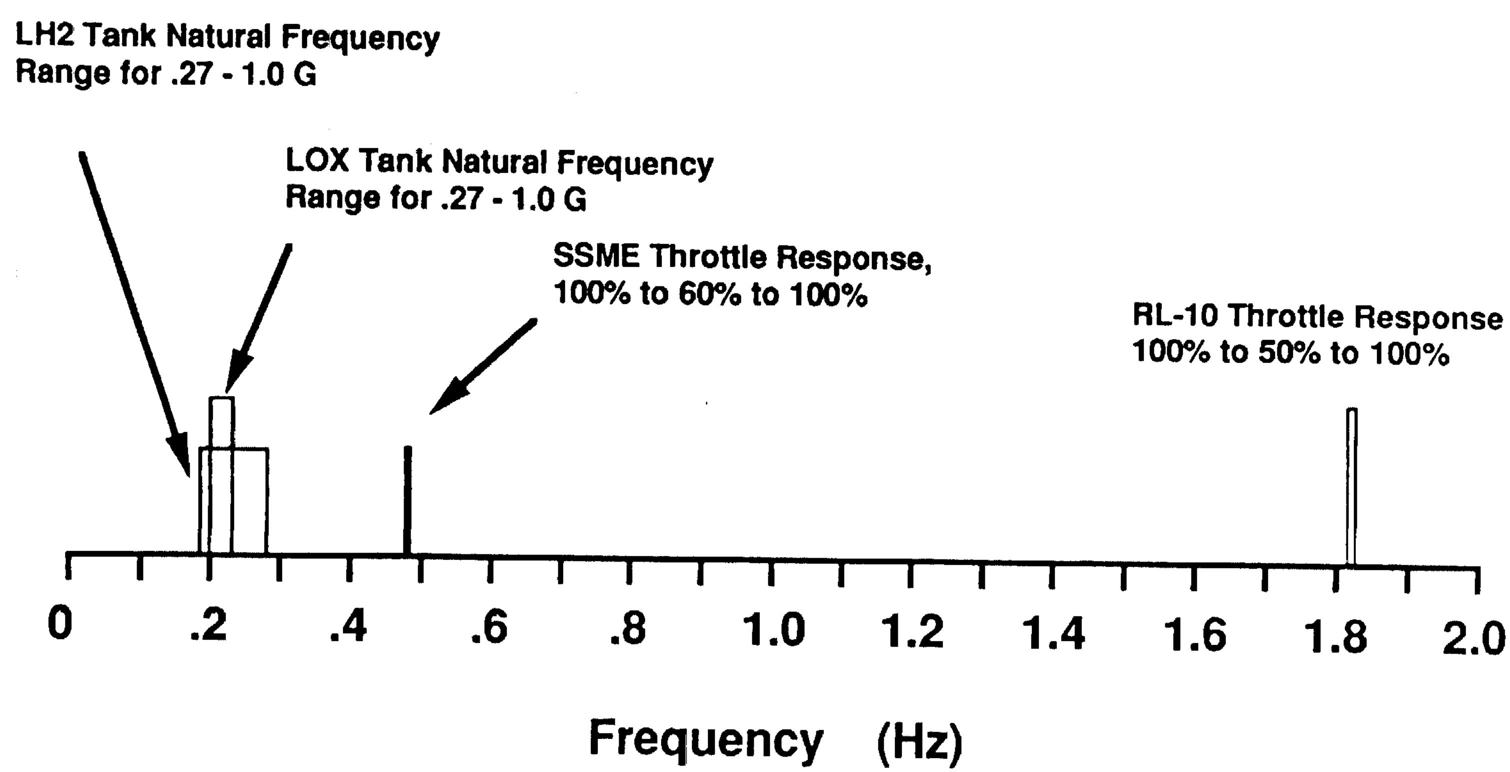
  - Comparison of Roll Authority:
    - Titan IV Stage 2 (w/ Centaur & Payload)
      Authority = 903 ft-lb/150,000 lb Stack Wt = 6 ft-lb/1000 lb Stack Weight
    - IME-Powered TLI Vehicle (w/Lunar Payload)
      Authority = 17,160 ft-lb/660,000 lb Stack Wt = 26 ft-lb/1000 lb Stack Wt

  - Provides 4.3X the Roll Torque per Vehicle lb Compared to Titan Stage 2, Which Uses a Similar Approach. Vehicle Diameter Differences, Reductions Due to GG-out, and TBD Roll Rate Requirements Need to be Considered.

  - 2 Nozzles Swiveled to Same 90° Positions Provide 0.21° of Pitch or Yaw TVC
    (Titan flight Data Indicates Most TVC Deflections are Less Than 0.2°)

Thrust Vector Control Pitch and Yaw Response - TLI

Throttle Response vs Propellant Slosh Frequencies



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TLI Thrust Vector Control Conclusions

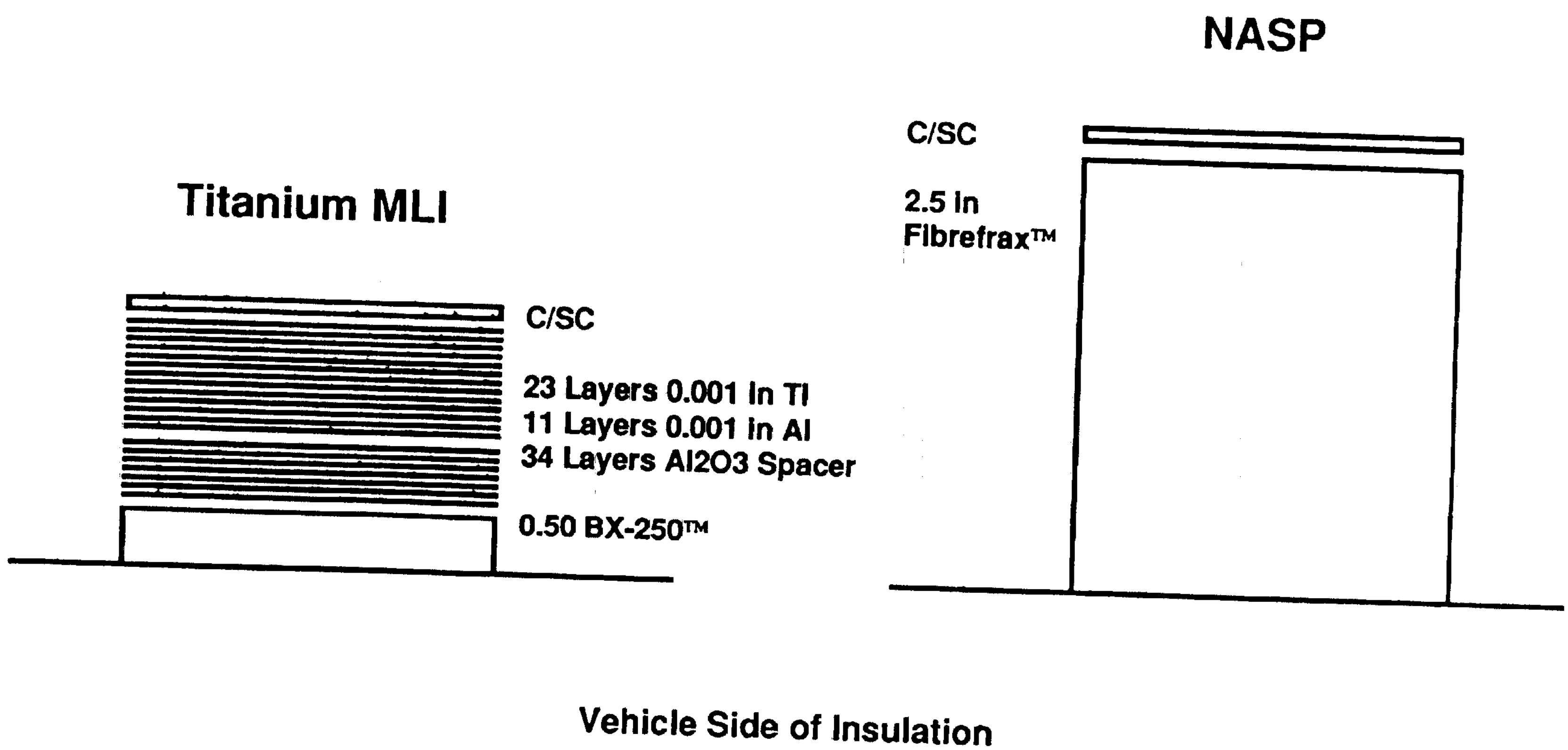
- Thrust Vector Control - Requirements
  - Pitch and Yaw Axes — Less Than 2 Degrees
  - Response — $\geq$ 0.3 HZ
- Thrust Vector Control - Capabilities
  - Pitch and Yaw Axes — Approximately 2 Degrees
  - Roll Control — On the Order of Titan
  - Response — Approximately 1.8 HZ
- Conclusion:
  - IME May Meet TLI Stage TVC Requirements Without Hydraulic Gimbal System

Note: TVC Requirements and Authority Are Addressed in Mid-term Briefing of 3-5-92.

Insulation Requirement

- All of the Insulation on the Lower TLI Stage Tank (Assumed to be LH2 as the Worst Case) Must Meet Requirements for Two Thermal Conditions.
    - Condition 1: It Must Prevent Condensables from Forming during Ground Hold, Assuming a Gaseous Nitrogen Purge.
    - Condition 2: It Must Hold Total Propellant Loss to 2% per Month or Less, or Approximately 5% for the LH2 Tank during the Two Month LEO Mission Specified for the TLI Stage.
- In Addition, the Insulation of the Engine Plug Nozzle Must Meet the Following:
    - Condition 3: It Must Withstand the Temperature of the Engine Exhaust Gasses. Protection from Approximately 2300 °F is Required for About the First 2 Feet from the Combustor Outlets, and from Approximately 1000 °F for the Balance of the Plug.

Insulation Configuration Candidates



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Insulation Solution 1 - "MLI-Type" System

- Insulation Configuration and Performance Prediction and Verification has Been Identified as a Technology Which Requires Development. Data Shown Here Represent Quick-look Solutions to Complex Problems.

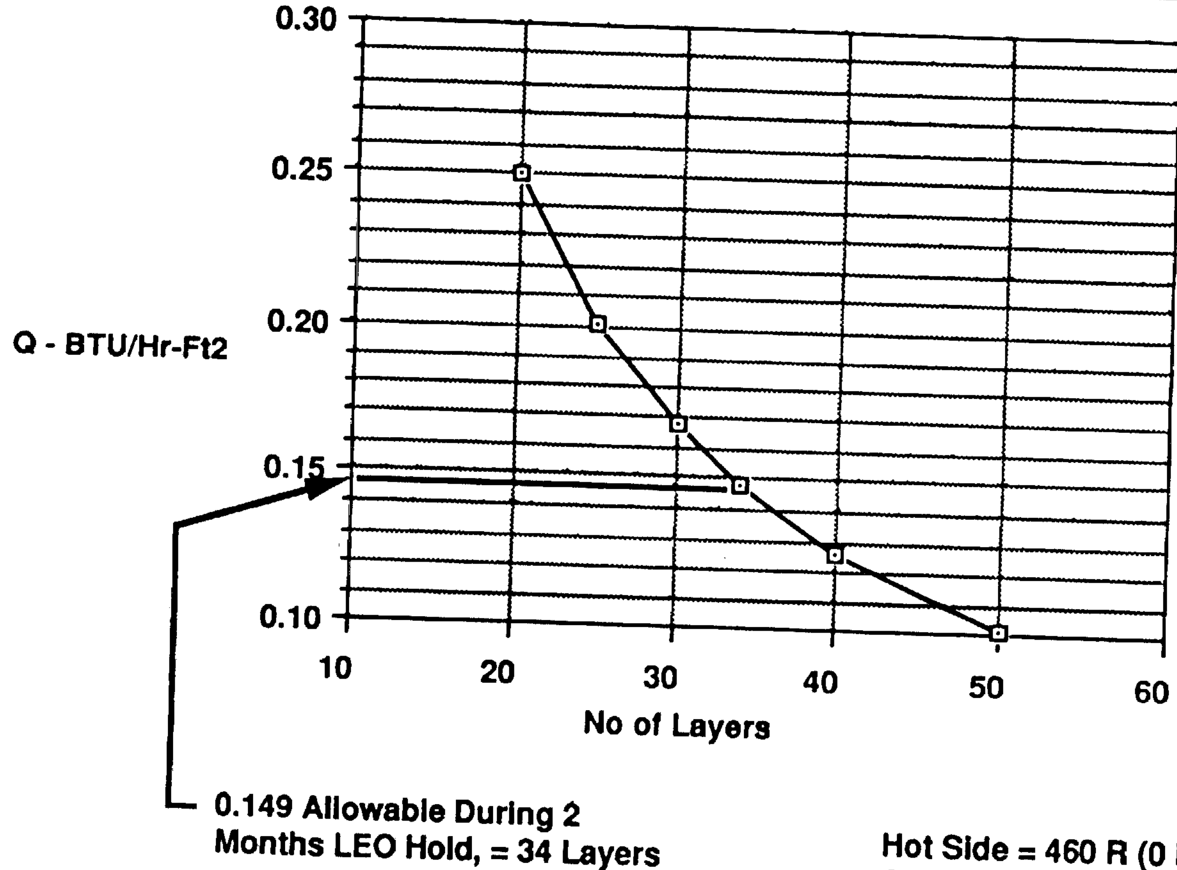
- Two Insulation Systems are Subjected to Top Level Evaluation. Insulation Solution 1 Uses a Conventional MLI Approach, but Made from Materials that Will Withstand the Expected Temperatures. Insulation Solution 2 Uses a High Temperature System Developed for NASP. This Chart Deals with Solution 1.

- The Conventional Solution Requires use of a 1/2 Inch Thickness of a Closed Cell Foam Insulation (SOFI - Sprayed on Foam Insulation) Applied Directly to the Tank Surface to Satisfy Condition 1, and Requires Approximately 1.0 Inch of Multilayer Insulation (MLI) to Satisfy Condition 2. The SOFI has Negligible Insulating Characteristics for Condition 2, and the MLI has Negligible Insulating Characteristics for Condition 1.

- If the Requirement for LEO Storage is Assumed to Size the Number of Layers and Thickness of a Multilayer Insulation System, the Following Logic Assesses How That Solution Handles the Engine Operation Represented by Engine Operation of Condition 3.

- The Conventional MLI System Requires Approximately 34 Layers of Radiation Barrier to Reduce LEO Heat Leak to 0.149 BTU/Hr-Ft2. (Shown on Subsequent Chart)

Multi Layer Insulation Requirement for Low Δ Temp



Plot Covers Insulation Density from 20 Thru 80 Layers per Inch

Hot Side = 460 R (0 F)
Cold Side = 37 R (-423 F)
Radiation Degradation Factor = 2
Emissivity = .042
Conduction Constant = 2.18×10^{-9}
Radiation Constant = 1.25×10^{-11}
(Based on Lockheed Correlation)

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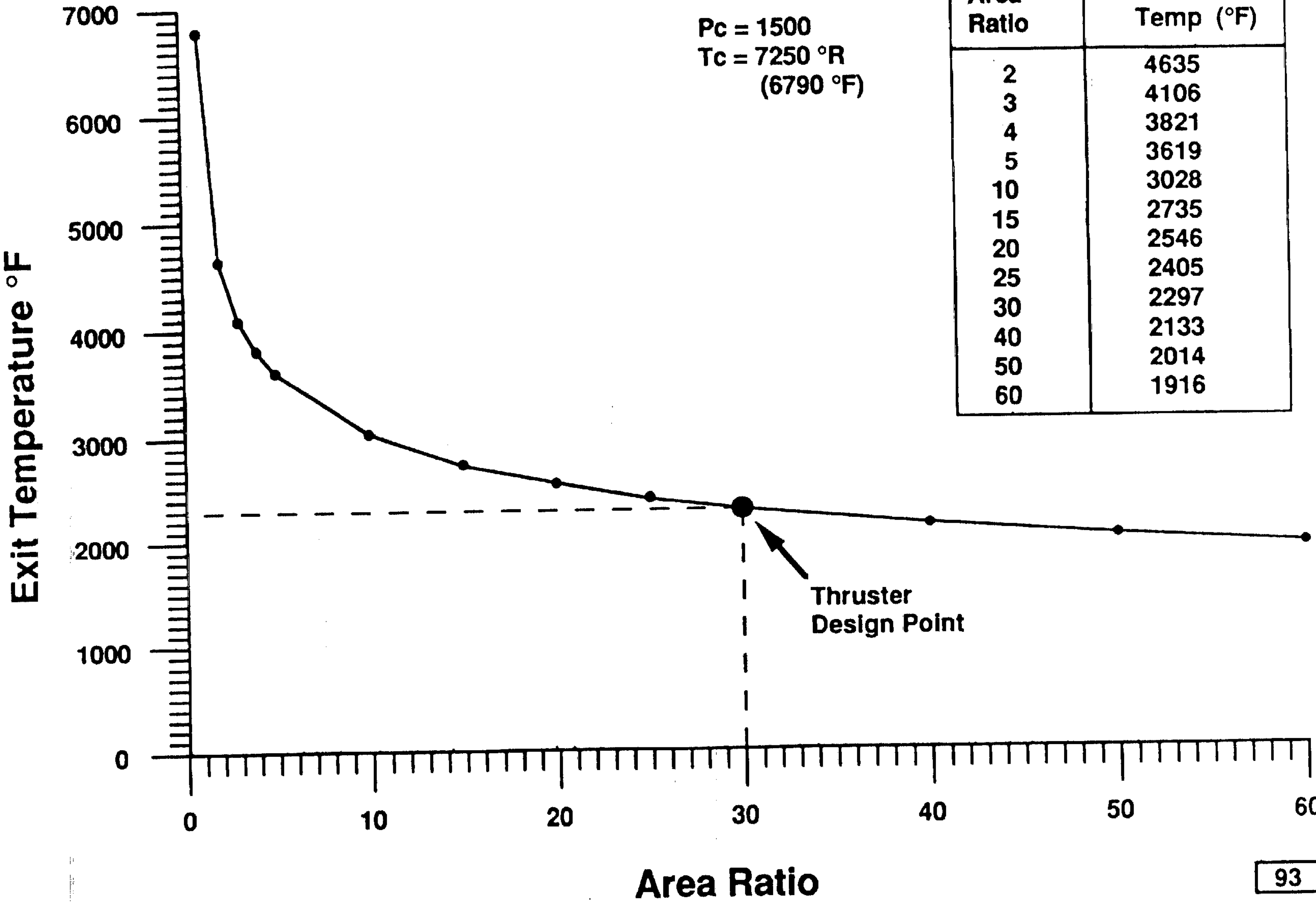
Engine Exit Temperature vs Area Ratio



Area Ratio	Exit Temp (°F)
2	4635
3	4106
4	3821
5	3619
10	3028
15	2735
20	2546
25	2405
30	2297
40	2133
50	2014
60	1916

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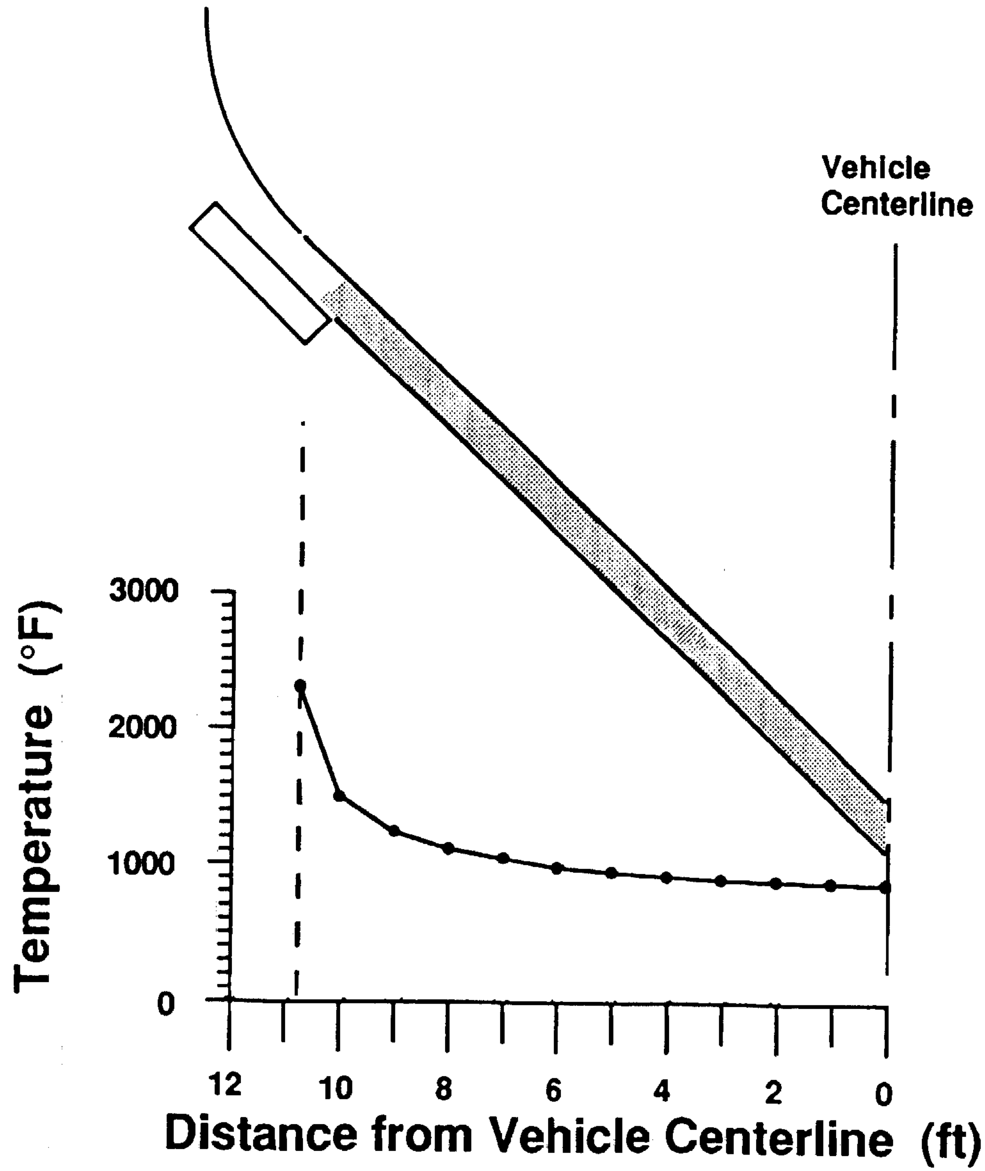
Gas Temperature vs Distance from Vehicle Centerline



Radius (ft)	Effective Area Ratio	Temperature (°F)
10.75	30	2297
10	154	1497
9	305	1235
8	440	1112
7	559	1040
6	662	987
5	750	949
4	822	923
3	877	904
2	919	891
1	940	884
0	948	882

Pc = 2500 Psia
Throat Area = 0.414 ft2

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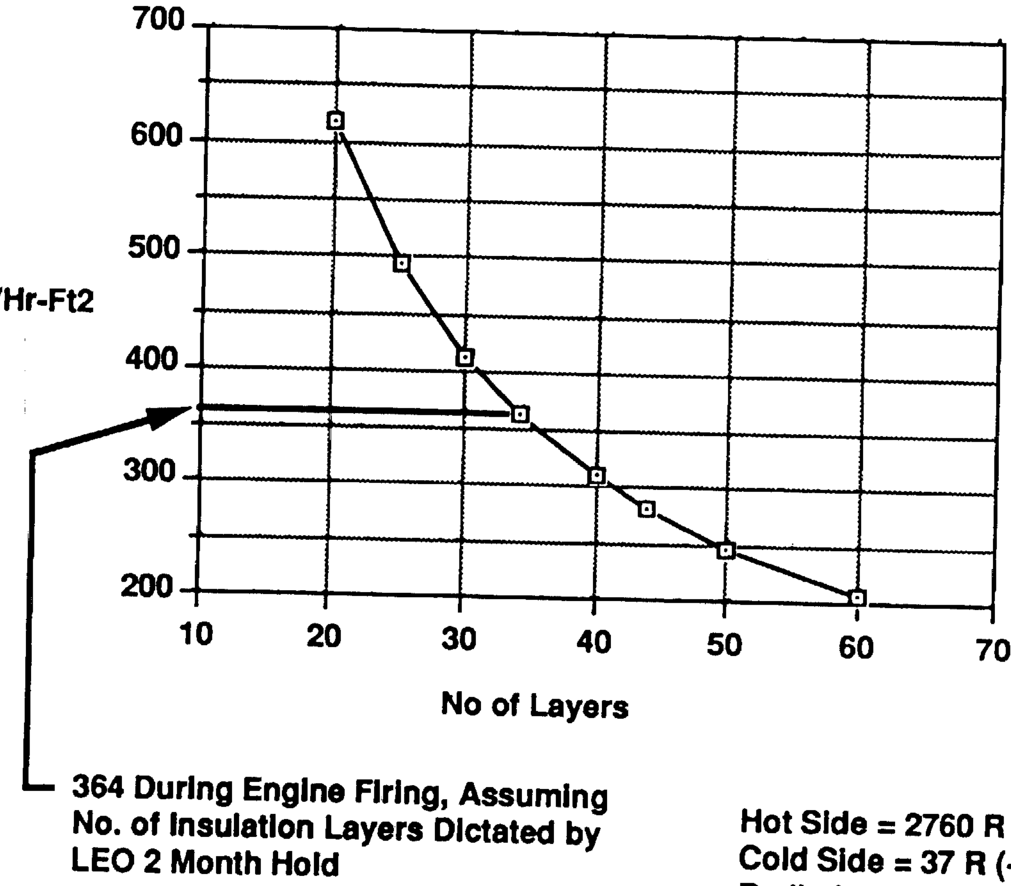
Insulation Solution 1 - "MLI -Type" System (Continued)

- If the Assumption is Made that a 34 Layer System is Capable of Withstanding the Exhaust Environment, and that Spacer Materials are Used Which Do Not Significantly Drive up the Conduction Constant in the High Temperature Areas, the Q Allowed by Such a System is 364 BTU/Hr-Ft2 in Those Areas Immediately Downstream of the Combustor Exhaust (Refer to the Following chart). This is Not Totally Conservative, Due to Extrapolation of the MLI Relationship to High Temperatures, but Will be Compared with the NASP System.

- Similarly, the Q Allowed by Such a System is 19 BTU/Hr-Ft2 for the Remainder of the Conical Nozzle.

Multi Layer Insulation Requirement for High Δ Temp



Plot Covers Insulation Density from 20 Thru 80 Layers per Inch

Hot Side = 2760 R (2300 F)
Cold Side = 37 R (-423 F)
Radiation Degradation Factor = 2
Emissivity = .042
Conduction Constant = 2.18 X 10^{-9}
Radiation Constant = 1.25 X 10^{-11}
(Based on Lockheed Correlation)

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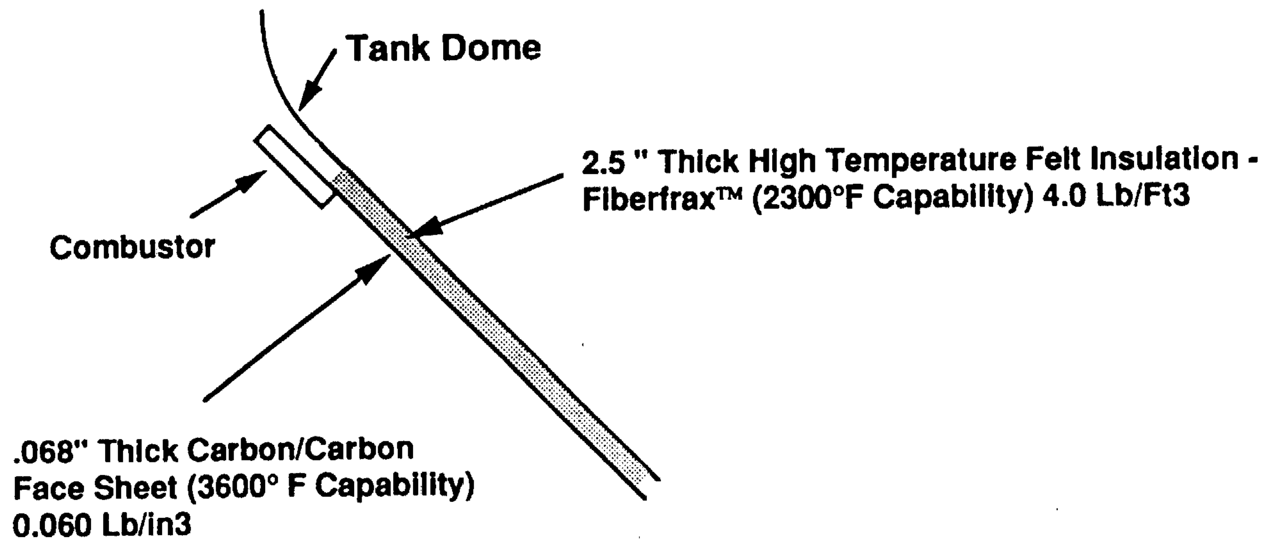
Insulation Solution 2 - NASP System

- A Proposed NASP Insulation System Was Evaluated for This Application. It is Capable of Withstanding Very High Temperatures, but is Not Designed for Long Term Cryogenic Storage. It Also Must Deal with Atmospheric Presence, and Cannot Make Full Use of the Radiation Predominant Heat Transfer Capabilities of Insulations for Space Use.



- LEO Heat Leak for This Insulation System is Calculated to be 245 Btu/Hr Ft2
- Heat Leak for This Insulation System at 2300 °F Rocket Exhaust is Calculated to be 1579 Btu/Hr Ft2

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Insulation Summary

- Insulation System Characteristics

	Ti MLI System		NASP System	
	Thickness (In)	Weight/Ft2 (Lb)	Thickness (In)	Weight/Ft2 (Lb)
Carbon/Carbon	0.068	0.578	0.068	0.578
Ti (23 ea)	0.023	0.563		
Al (11 ea)	0.011	0.158		
Al_2O_3 spacer (34ea)	0.578	0.097		
Fibrefrax™			2.500	0.833
BX-250™ SOFI	0.500	0.083		
	1.180	1.488	2.568	1.420

	Ti MLI	NASP
• Thermal Conductivity		
Engine Operation: 2300 to -423 °FDT (Btu/Hr Ft2)	364	1579
Engine Operation: 1000 to -423 °FDT (Btu/Hr Ft2)	19.2	825
LEO Storage: 0 to -423 °F DT (Btu/Hr Ft2)	0.149	* 245
• Engine Heat Input		
Assuming 2300°F for 2 Ft, and 1000°F for the Bal (Btu/Hr)	67,700	533,000
• Allowable Engine Heat		
Assuming All Heat Enters Engine Inlet Stream, and 5 Psig Vapor Press Increase is Acceptable (Btu/Hr)	4,590,000 (68:1 Margin)	4,590,000 (9:1 Margin)
• LEO Storage Heat Input		
Assuming 5% Total LH2 Loss In Two Months (Btu/Hr Ft2)	0.149 (Design Point)	245 (1640:1 Negative Margin)

* Extrapolated, Since Data Not Available: May be Better In Vacuum

Note: All Calculations Assume Steady State Thermal Conditions.

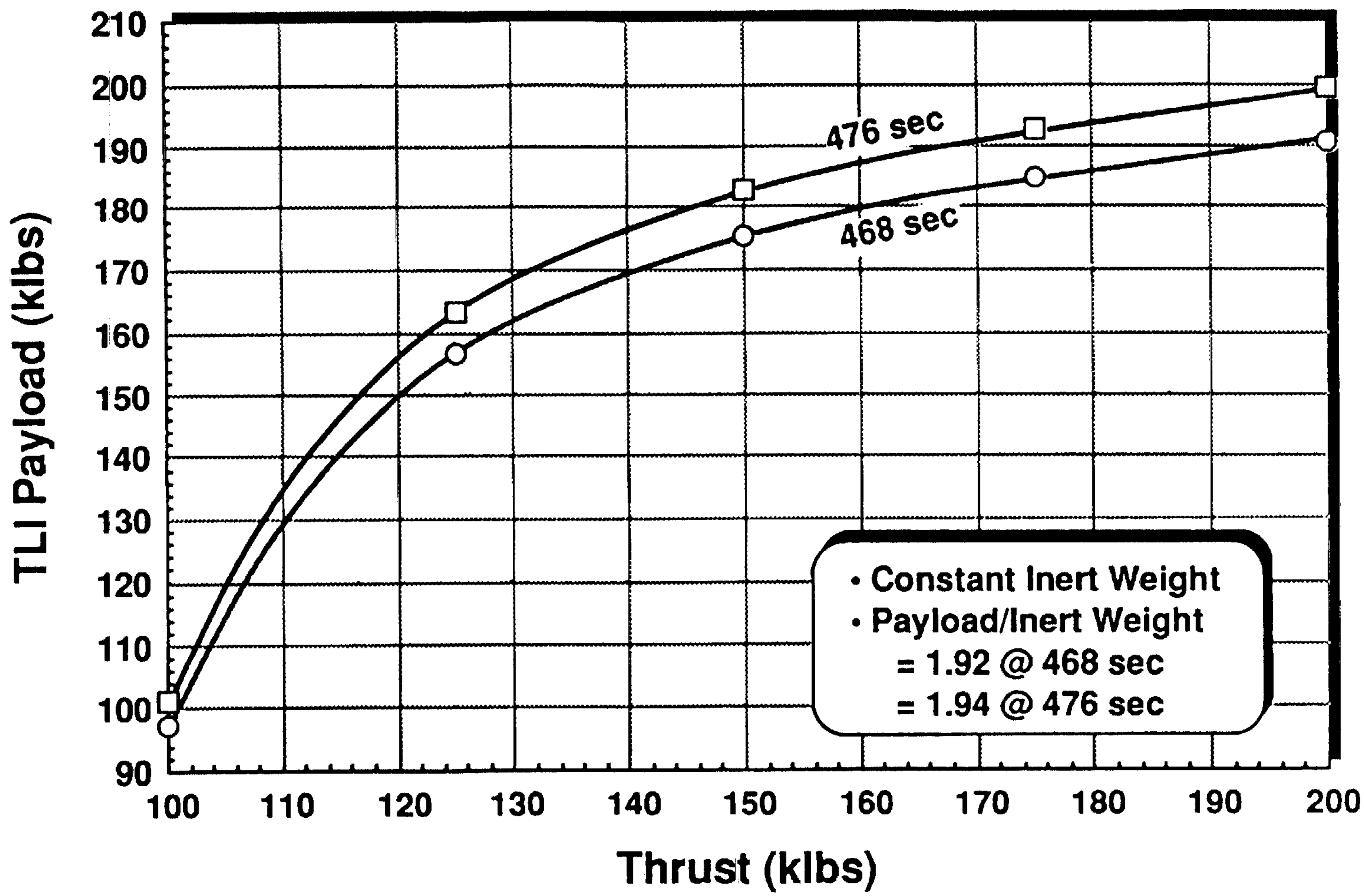
Insulation Conclusions

- The Insulation Requirement for LEO Hold is Probably the Driving Heat Transfer Requirement for a Vehicle Cryogenic Tank Expansion Surface that Must be Exposed to IME Exhaust.
- Either of the Two Insulation Systems Considered Show Promise of Satisfying the Requirements, Although the Data are Preliminary.
- Using the Surface of a Cryogenic Tank for Rocket Exhaust Expansion, While an Engineering Challenge, Appears to be a Practical thing to Consider.

TLI Net Performance

- Performance Gain Realized as a Trade of Increased Isp and Increased Engine Weight

TLI Payload Capabilities



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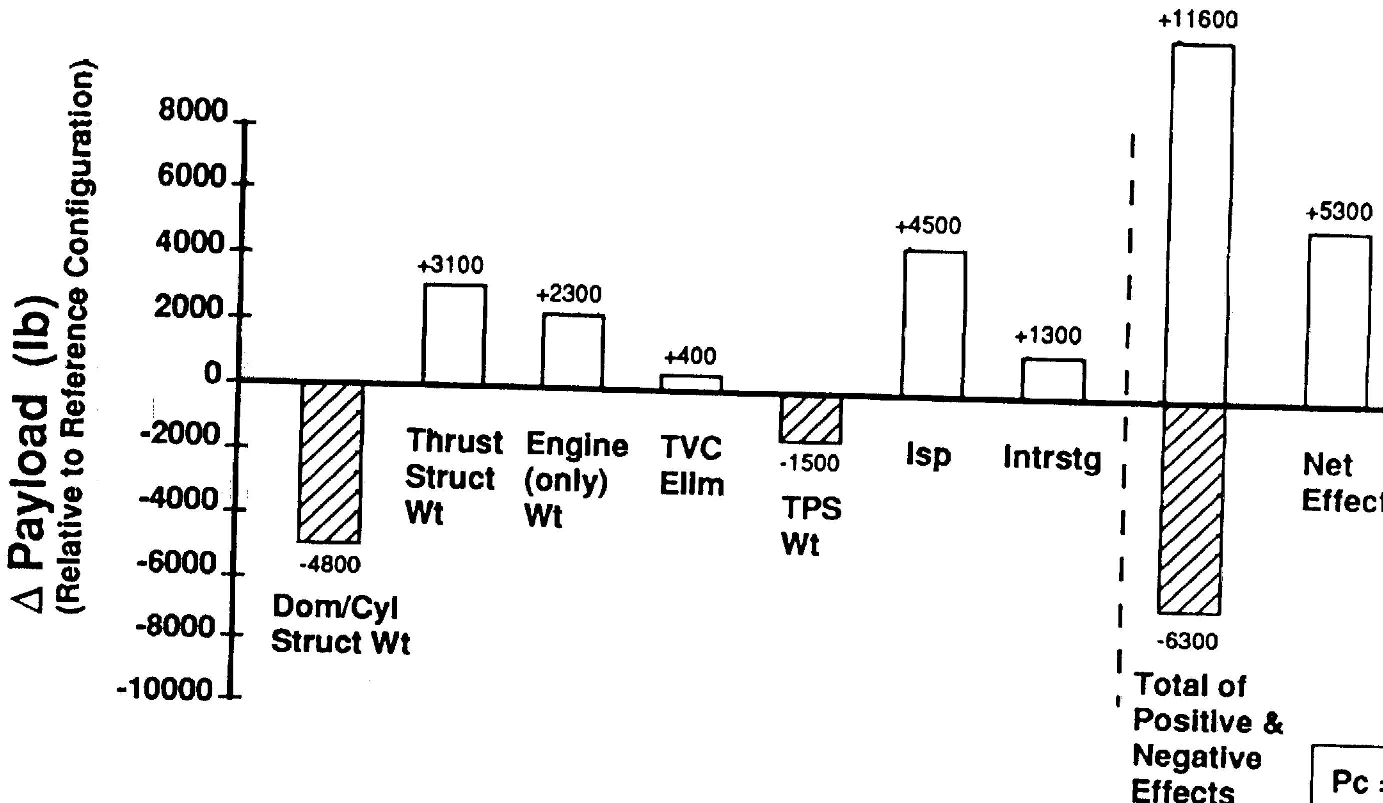
TLI Payload Effects - Details



Relative Strength of Effect (% of Total of Effects)	Dom/Cyl Struct Wt	Thrust Struct Wt	Engine (only) Wt	TVC Elim	TPS Wt	Isp	Intrstg	Total of Positive & Negative Effects
	27%	17%	13%	2%	9%	25%	7%	100% (17,900 lb)

Pc = 1500
Plug Engine
45° Conical plug

Sensitivities: +1 lb Δ TLI Stage Dry Wt = -1.94 lb Δ Payload
+1 sec Δ Isp = +1000 lb Δ Payload

JG920512-01B

Agenda

• Introduction and General Overview	M. Wakefield
• TLI Stage	
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- Logic Behind Selection	J. Greenwood
- Requirements Satisfaction	J. Greenwood
• Lunar Lander	
- Selected Design	M. Wakefield
- Logic Behind Selection	J. Greenwood
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• Reliability Assessment	
• Technology Plan	M. Wakefield
• Conclusions	M. Wakefield

MW920602-05

IME Matrix

Missions - Mission Characteristics - Requirements/issues		Primary IME Benefit		Additional IME Benefits
• **TLI** - Dual Fault Tolerant - Fixed IMLEO (how to best utilize) - Gravity Loss Sensitivity - 100 - 200 Klb Thrust - Space Storage - TVC - Number of Burns Sensitivity - Manned vs Unmanned	→	**Meet Fault Tolerance Rqt (w/Improved Reliability)**	+	**Increased P/L to Surface** - Eliminate Gimbal System Cost & Wt - Improved Isp if Use Stage Surface for Expansion - Shorter Interstage Allows More IMLEO for a Given Launch Vehicle

JG920310-01C

TLI - Requirements, Mission Characteristics, & Issues

- Dual Fault Tolerant	- Key Requirement
- Fixed IMLEO (how to best utilize)	- Optimization Issue
- Gravity Loss Sensitivity	- Mission Characteristic
- 100 - 200 Klb Thrust	- Requirement
- Space Storage	- Requirement
- TVC	- Requirement
- Number of Burns Sensitivity	- Mission Characteristic
- Manned vs Unmanned	- Issue

JG920413-21A

Fault Tolerance - Thrust Comparison

Number of Failed Components	IME			Conventional			
	3 Pump 6 Seg	4 Pump 8 Seg	6 Pump 12 Seg	3 Eng	4 Eng	5 Eng	6 Eng
1 Pump	100	100	100	83	94	100	100
2 Pumps							
A. Opposite	N/A	75	100	N/A	63	75	83
B. Adjacent	41	75	67	41	63	75	$\boxed{83}$
C. Intermediate	N/A	N/A	67	N/A	N/A	75	$\boxed{83}$
1 Segment	83	100	100	83	94	100	100
2 Segment							
A. Same Pump	41	63	83	$\boxed{83}$	94	100	$\boxed{100}$
B. Diff Pumps							
- Opp Pump/Opp Seg	N/A	94	100	N/A	63	75	83
- Op Pump/Non-Op Seg	N/A	76	94	N/A	63	75	83
- Adj Pump/Adj Seg	41	63	94	41	63	75	83
- Adj Pump/Opp Seg	83	63	94	41	63	75	83
1 Pump/Adj Seg	83	75	91	41	63	75	83
1 Pump/Opp Seg	83	75	91	41	63	75	83

IME Assumptions:
1. Pump Normal Op Is at 2/3 Capacity
2. Max Op Pc Is 25% Above Normal

Conventional Engine Assumptions:
1. Max Engine Operation Is 25% Above Normal

All Numbers Are Percent of Normal Thrust

Boxed Numbers Indicate IME Benefit Over Conventional Engines Having Same Number of Pumps

Boxed Numbers Indicate Conventional Engine Benefit Over IME Having Same Number of Pumps

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MW920407-01

C-4

Fault Tolerant - TVC Comparison

Number of Failed Components	IME						Conv.
	3 Pump 3 Seg	3 Pump 6 Seg	4 Pump 4 Seg	4 Pump 8 Seg	6 Pump 6 Seg	6 Pump 12 Seg	3, 4, 6 Eng
1 Pump	Y	Y	Y	Y	Y	Y	Y
2 Pumps							
A. Opposite	N	N/A	Y	Y	Y	Y	Y
B. Adjacent	N	N	Y	Y	Y	Y	Y
C. Intermediate	N/A	N/A	N/A	N/A	Y	Y	Y
1 Segment	Y	Y	Y	Y	Y	Y	Y
2 Segment							
A. Same Pump	N/A	Y?	N/A	Y	N/A	Y	Y
B. Diff Pumps							
- Opp Pump/Opp Seg	N	Y	Y?	Y	Y	Y	Y
- Op Pump/Non-Op Seg	N	Y?	Y?	Y	Y	Y	Y
- Adj Pump/Adj Seg	N	Y?	N	Y	Y	Y	Y
- Adj Pump/Opp Seg	N	Y?	N	Y	Y	Y	Y
1 Pump/Adj Seg	Y	Y	Y	Y	Y	Y	Y
1 Pump/Opp Seg	N	Y	Y?	Y	Y	Y	Y

IME Assumptions:
1. Pump Normal Op Is at 2/3 Capacity
2. Max Op Pc Is 25% Above Normal
3. Segments are Interconnected

Conventional Engine Assumptions:
1. All Engines Gimbal Sufficiently to Maintain TVC.

Y Means The System Is Capable of Maintaining TVC

Boxed Numbers Indicate Reduced Capability

MW920423-04

TLI Requirements Satisfaction

- Dual Fault Tolerance
  - Comparisons have been Made to Show that IME Architecture Can Easily Meet Dual Fault Requirements.
  - A Detailed Reliability Analysis is Presented in the Reliability Section of this Report.
- Fixed IMLEO
  - Fixed IMLEO Allows the Payload to Increase with Increased Engine Performance on a TLI Stage.
- Gravity Loss Sensitivity, 100-200 Klb Thrust
  - Isp and Thrust Level Relate Directly to Payload Capability. Payload Increase is Approximately 5,000 Pounds for the Predicted Isp Increase of the IME Engine, or the Thrust Could have been Approximately 15,000 Pounds Less for the Same Payload.
- Space Storage
  - Ability to Use an Engine System in Space after a Significant Storage Duration is Related to Materials Selected, VHM, and Operating Margins, Especially Starting Margins. Any New Engine System May Apply the Principals Necessary to Achieve this Requirement, which is a Significant Benefit for a New System. Existing Systems May Meet the Requirements, but Full Evaluation is Required, and Some Compromise May be Necessary.

TLI Requirements Satisfaction (Continued)

- **Thrust Vector Control (TVC)**
  - Basic Capability of an IME System to Provide Thrust Vector Control by Thrust Modulation has Been Shown. Significant Analysis, Combined with Subscale Testing, Will Be Required to Further Quantify the TVC Capabilities and Subtleties Associated with Combustor-Out and/or Throttling Requirements and Capabilities.

- **Number of Burns Sensitivity**
  - Number of Burns for an Event Like TLI Is Not Very Sensitive to Propulsion System Characteristics at the High Thrust Levels Contemplated Here.
  - Number of Engine Starts Must Be Addressed from the System Standpoint. A New Design has an Advantage in Addressing Requirements Like Multiple Engine Starts without Resulting in Compromise.

- **Manned vs. Unmanned**
  - For High Value Cargo, There are Few, if any, Discriminators between Manned and Unmanned Missions. Reliability, Redundancy and Fault Tolerance, Vehicle Health Monitoring - All Play a Key Role in Mission Success and its Corollary - Safety.

Lunar Lander

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• Reliability Assessment	R. Welborne
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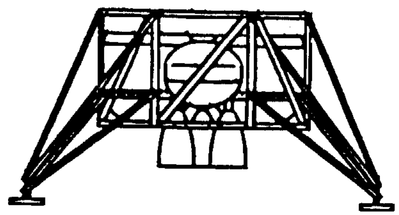
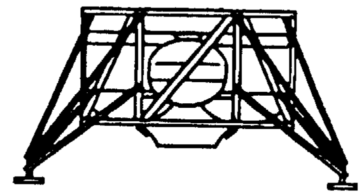
• Conclusions	M. Wakefield

MW920602-06

IME vs Conventional Engine - Lunar Lander

Baseline

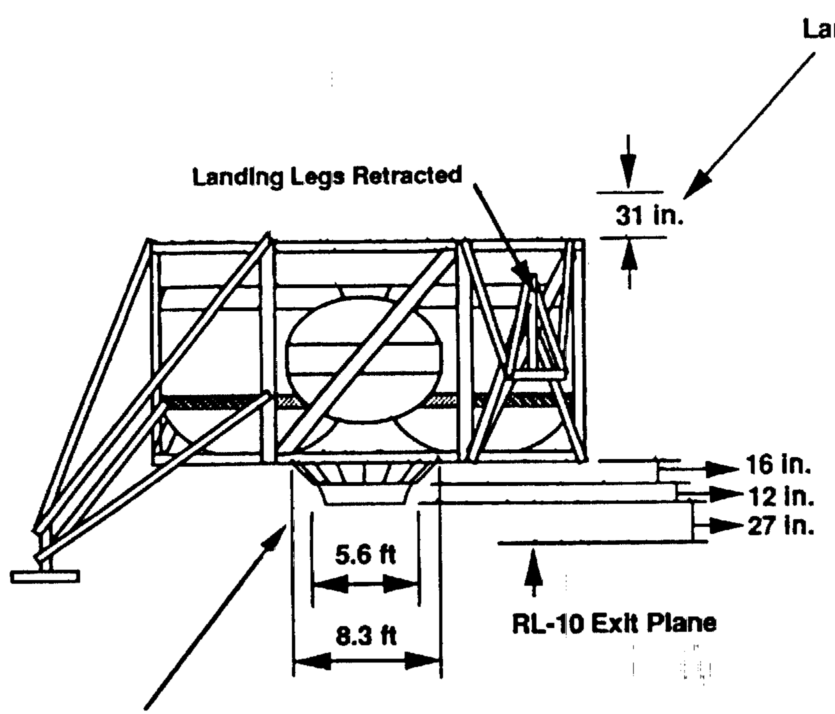
Propellant Tank Diameters & Body Diameter Stays Constant. Landing Legs Shortened 2.3 ft (Vertically) Due to Shorter Engine, Lander Body Shortened 0.3 ft Due to 1000 lb Reduction in Propellant Wt.

Common Characteristics	Conventional Propulsion (5 Engines, RL10A-4)	IME Propulsion (4TPA, 32 Compustors)	Discriminators
Safety	Dual Fault Tolerance	Dual Fault Tolerance	Failure Thrust Higher
Reliability	0.9994	0.9995	Both Excellent, but IME More Recovery Options
Payload, Tonnes	23.1	23.6	+ 0.5 Tonne
Isp, Seconds	450	466	+ 16 Sec
Health Monitoring	Could be Incorporated	Integral	Benefit of New Des
Throttling Range	17:1 (Assumes Development)	5:1	5:1 vs 17:1
Space Storage	Needs EMA's, Throttling	Integral	Benefit of New Des
Manned vs Unmanned		2.6 ft Closer to Surface	2.6 ft Closer to Surface
Plume Impingement Pressure, psi	2.4	Approximately 0.5	Reduce Debris
Thrust Vector Control	Gimbals	Thrust Mod and GG Exh	Eliminate System
Development Cost, ROM $	150 M	See P 211	-
Production Cost, ROM $	3.2X5=16M*	3.0M	13M Less
Landed Height	9.02 M	8.0 M	- 2.6 Ft
Landing Gear Span			- 3.3 Ft

Note: *Not Throttleable

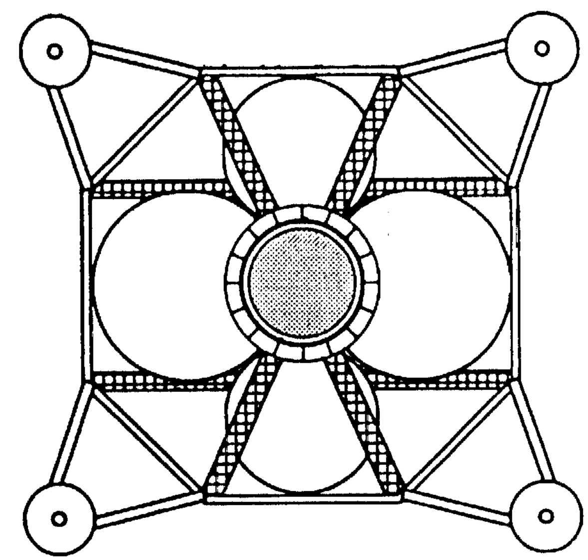
MW920514-02

Lunar Lander: Plug Cluster Engine



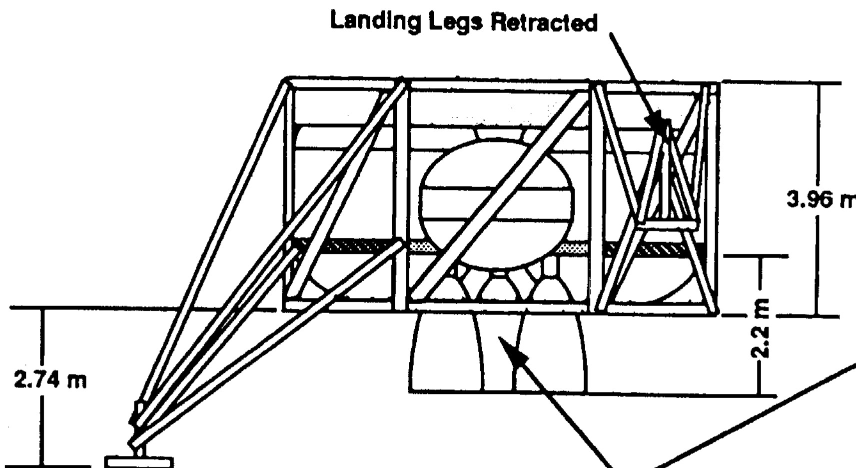
33 Klb Plug IME (Plug Truncated at 10%)
- 16 Modules,
- Thrust = 2063 lb ea.
- Pc = 1500
- Isp = 466 sec
- Weight = 1852 lb
- Thruster Nozzle: 1.5 in. x 13.3 in.
  - 34 Nozzle Thicknesses Above Lunar Surface

Side View



Bottom View

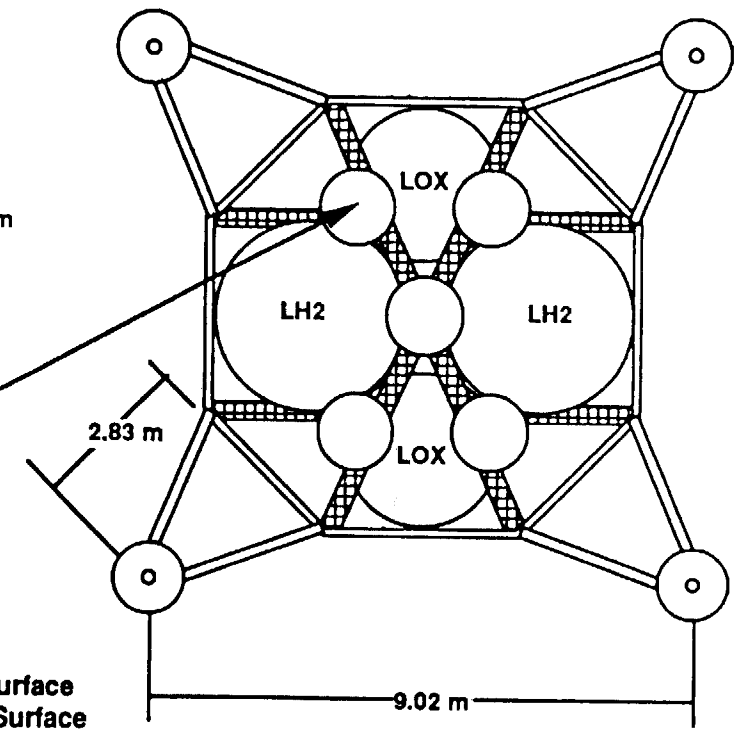
Lunar Lander: RL-10 Engines



Landing Legs Compress Slightly Under Load to Provide 1 Meter Clearance Between Nozzles & Lunar Surface

RL10A-4 Engines (5x)
- 20.8 Klb Thrust ea.
- Isp = 449
- Pc = 578
- Nozzle Dia = 46 In.
- 39 In. from Nozzle to Lunar Surface
- .85 Nozzle Diameters Above Surface

Side View



* Diagonal Length 12.7 m

Bottom View

JG920423-03A

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MW920602-07

Issues Considered for Lunar Lander

- Engine Thrust Level Selection
- Engine Pc Selection
  - Payload Effects
  - Engine Size & Weight
- Landing Plume Impingement
- Thrust Vector Control

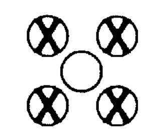
Lunar Lander IME Thrust Level Selection

Approach: Match Thrust Levels for 2-Fault Condition (vs. Nominal)

Rationale: Vehicle Sizing is Based on Worst Case not Nominal

Benefit: Faults Affect the Thrust Level of an IME Much Less than a Conventional Engine Configuration for a Lander.

- IME with 2 Faults = 37% Thrust Reduction (worst case)
- Conventional with 2 Faults = 80 % (worst case)



Results: Worst case Conventional = 0.20 x 104 Klb = 20.8 Klb

Nominal IME Thrust = 20.8 Klb/(1-.37) = 33 Klb

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Thrust/Weight Comparisons - Lunar Lander Vehicle

- **Apollo Lunar Module:**

Orbit Insertion Wt:	36,252 lb	→ T/W = .27
Thrust:	9,802 lb	

- **STV Configuration (5 RL-10's, Two Faults - Only One Engine Running)**

Orbit Insertion Wt:	110,400 lb	→ T/W = .19
Thrust (1 eng.):	20,800 lb	
2 engines:		→ T/W = .38
3 engines:		→ T/W = .57
4 engines:		→ T/W = .76
5 engines:		→ T/W = .95

- **STV/IME Configuration (33 Klb Nominal Thrust)**

Orbit Insertion Wt:	110,400 lb	→ T/W = .30
Nominal Thrust:	33,000 lb	
2-Fault Thrust:	20,800 lb	→ T/W = .19

- **Conclusion:**

IME Engine: → Good Match with Apollo Lander T/W Experience

Conventional Engine: → Overpowered with all 5 engines Running

→ Underpowered with 1 Engine Running (2 faults)

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Engine Throttling for Lunar Lander Vehicle

	Descent	Weight	Thrust/ Weight	Thrust	Thrust/ Engine (5 RL10's)	% of Full Thrust	Throttle Ratio	IME @ 33 Klb %	IME @ 33 Klb Ratio
Cargo	Start Descent	48.4 mt (106,480 lb)	.27 (Earth G's)	28,750 lb	5,750 lb	28%	4	87%	1
Cargo	Touchdown	29.2 mt (64,240 lb)	0.8 (Lunar G's)	8,565 lb	1,713 lb	8%	12	26%	4
Piloted	Start Descent	43.1 mt (94,820 lb)	.27 (Earth G's)	25,600 lb	5,120 lb	25%	4	78%	1
Piloted	Touchdown	21.0 mt (46,200 lb)	0.8 (Lunar G's)	6,160 lb	1232 lb	5.9%	17	19%	5

Ascent (Piloted)	Weight	Thrust/ Weight	Thrust	Thrust/ Engine (5 RL10's)	% of Full Thrust	Throttle Ratio	IME @ 33 Klb %	IME @ 33 Klb Ratio
Liftoff	14.0 mt * (30,800 lb)	3 Lunar G's	15,400 lb	3,080 lb	15%	7	46%	2
Burnout	8.56 mt (18,830 lb)	3 Lunar G's	9,415 lb	1,883 lb	9%	11	29%	3

Weights:	Lander Dry Wt	Descent Prop Wt	Ascent Prop Wt	Crew Cab Wt	Cargo	Initial Descent Wt	Touchdown Wt
Cargo:	6.1 mt	19.2 mt	N/A	N/A	23.1 mt	48.4 mt	29.2 mt
Piloted:	6.1 mt	13.5 mt	7.5 mt	2.5 mt	5.0 mt	43.1 mt	21.1 mt

Throttle Ratio
1-5 "Easy"
5-10 "Hard"
>10 "Very Hard"

* includes boiloff during Lunar stay

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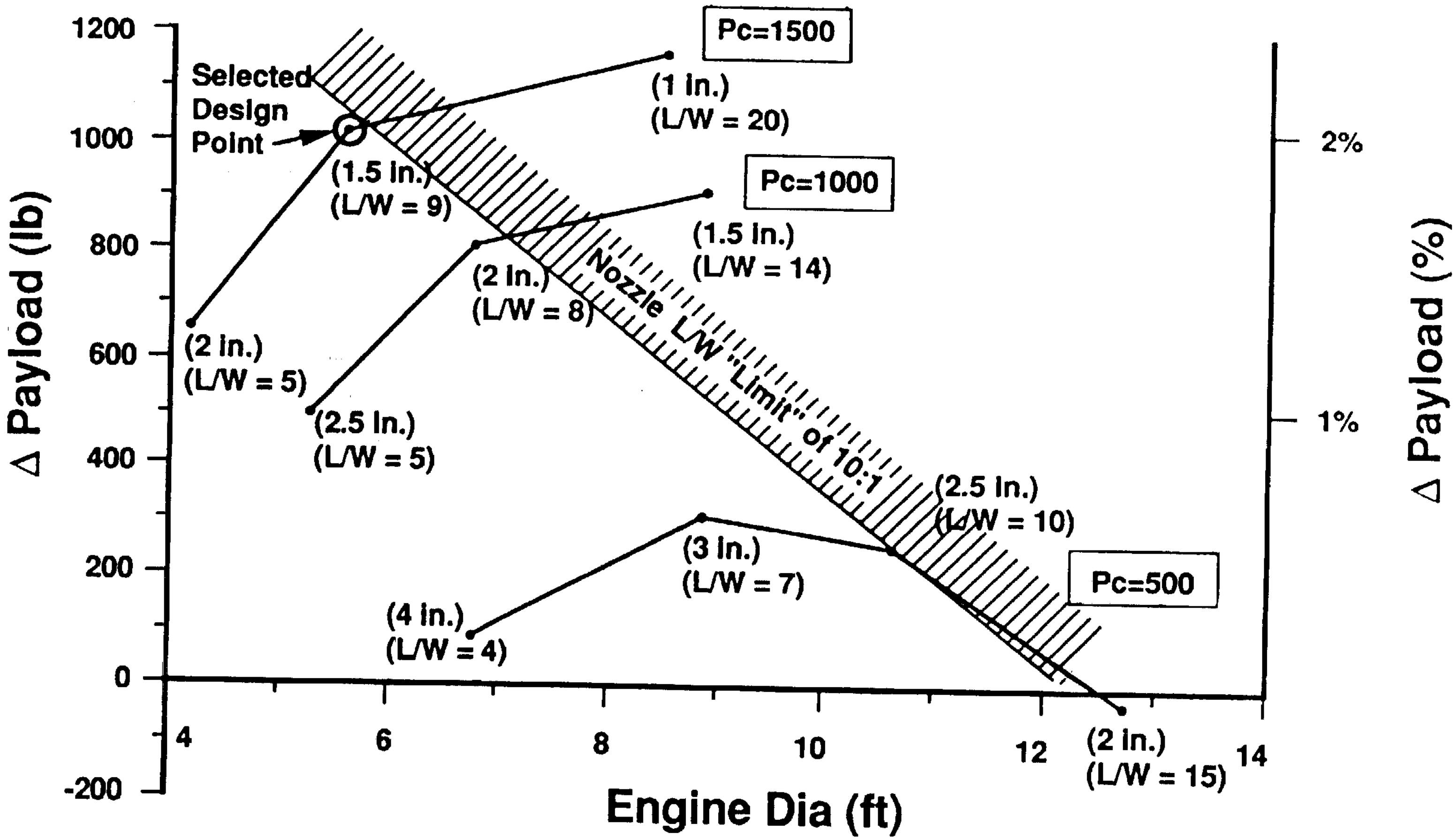
Δ Payload vs Pc for Lunar Lander

Reference Payload = 50,800 lb (23.1 mt)



Notes: (X in.) - Thickness (or width) of rectangular nozzle exit
(L/W = 15) - Length-to-Width ratio of the rectangular opening
Each engine is comprised of 16 thrust cells, each with an expansion ratio of 30

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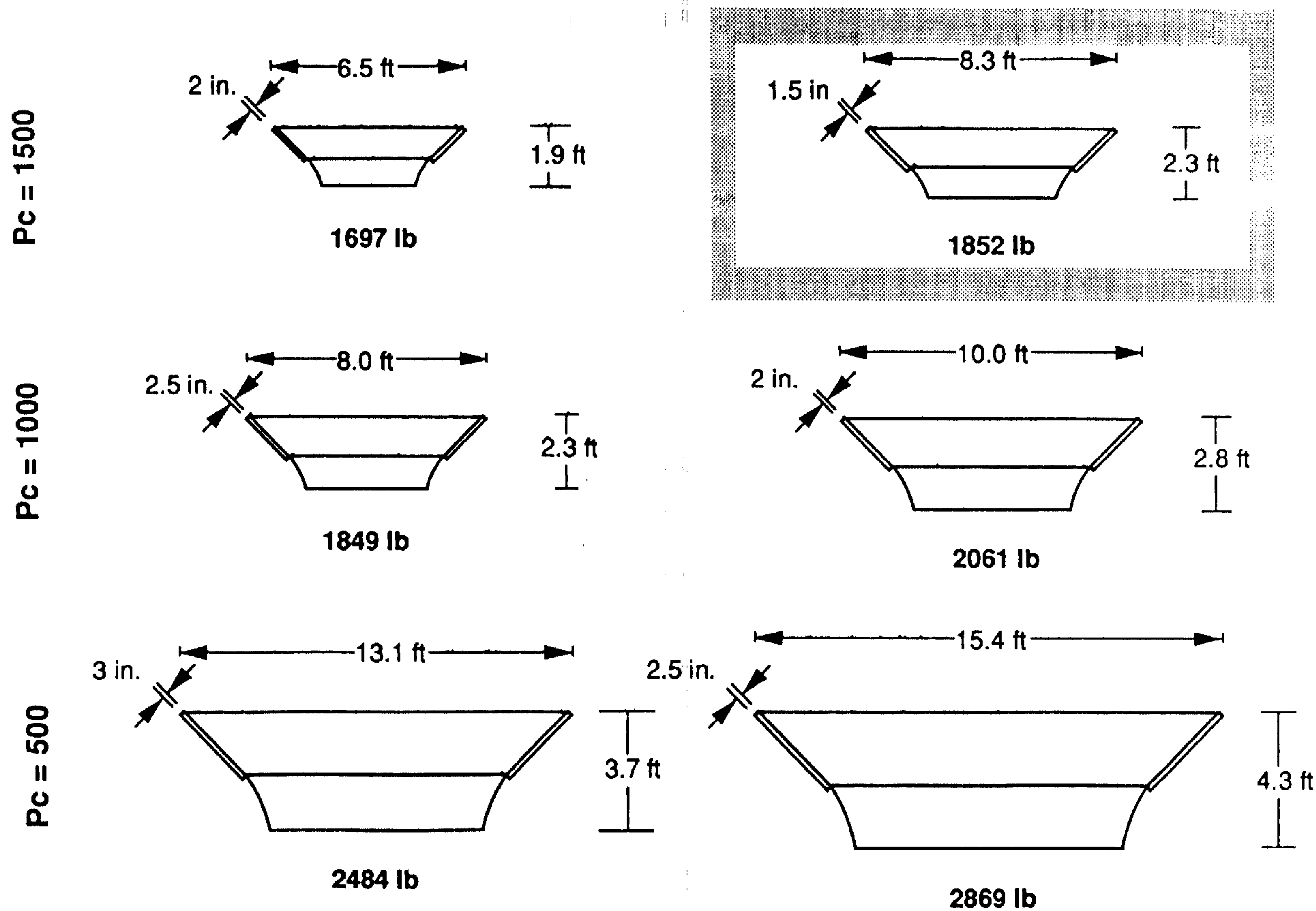
Lunar Engine Size & Weight vs Pc



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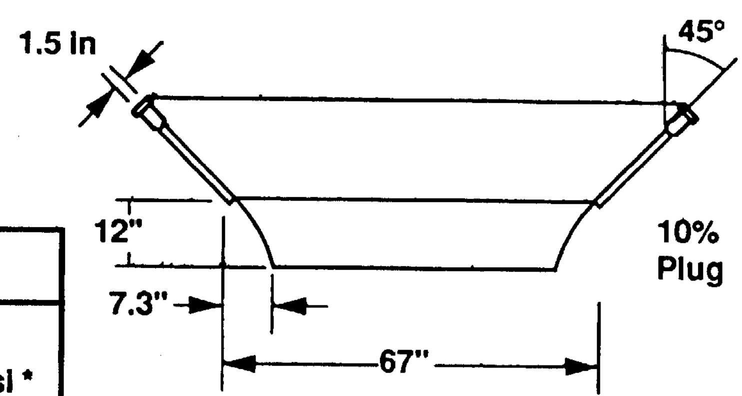
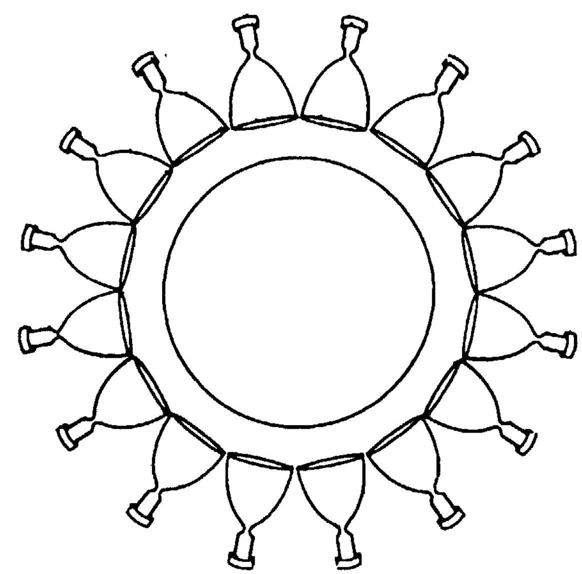
Selected Lunar Engine Configuration

Description
Max Pc = 1500 psi Pc at Touchdown = 487 psi *
Throat Area 0.67 in2 per module 10.7 in2 total (for 16 modules)
Module Area Ratio = 30 Module Nozzle Exit: (rectangular) - 13.3 in wide - 1.5 in thick



Side View
(Cross Section)



Bottom View

(nozzle exits are
essentially rectangular)

* Thrust = 10,700 lb = Lunar Wt of heaviest cargo configuration (T/W =1 at Touchdown)
Isp = 466 sec
Mass Flow = 23.0 lb sec

JG920610-01B

Lunar Landing Debris Concern

Concern - Debris Generated by Engine Exhaust Impingement on the Lunar Surface may Have a Number of Undesirable Effects

- Obscuring of Landing Site, Making Hazard Avoidance and Navigation More Difficult
- Throwing of Large Size Debris Significant Distances (Hundreds or Even Thousands of Feet). Concern over Major Damage from Debris, or Problems from Dust Contamination of Optics, Windows, or Mechanisms on Existing Lunar Installations.
- Causing Damage to Critical Portions of a Lander, which May Interfere with a Safe Landing, or Preclude Reuse as an Ascent Vehicle.
- Debris May Be Generated at a Rate Capable of Creating a Significant Crater Beneath the Landing Vehicle.

Basis for Concern

- Analysis
  - Calculated Engine Exhaust Plume Impingement Pressures on the Lunar Surface for:

"Conventional" 4 Engine Lunar Lander	- 1.3 Psi	Numbers are for Landing, Takeoff Values are 3 Times Larger
Single Plug Nozzle Engine	- 0.3 Psi	
Apollo (Approximately)	- 1.0 Psi	

  - Analysis Shows Reverse (Upward) Flow in Center of 4 Engine Configuration While Engine Bell is as High as 50 Nozzle Diameters above the Lunar Surface

Lunar Landing Debris Concern (Continued)

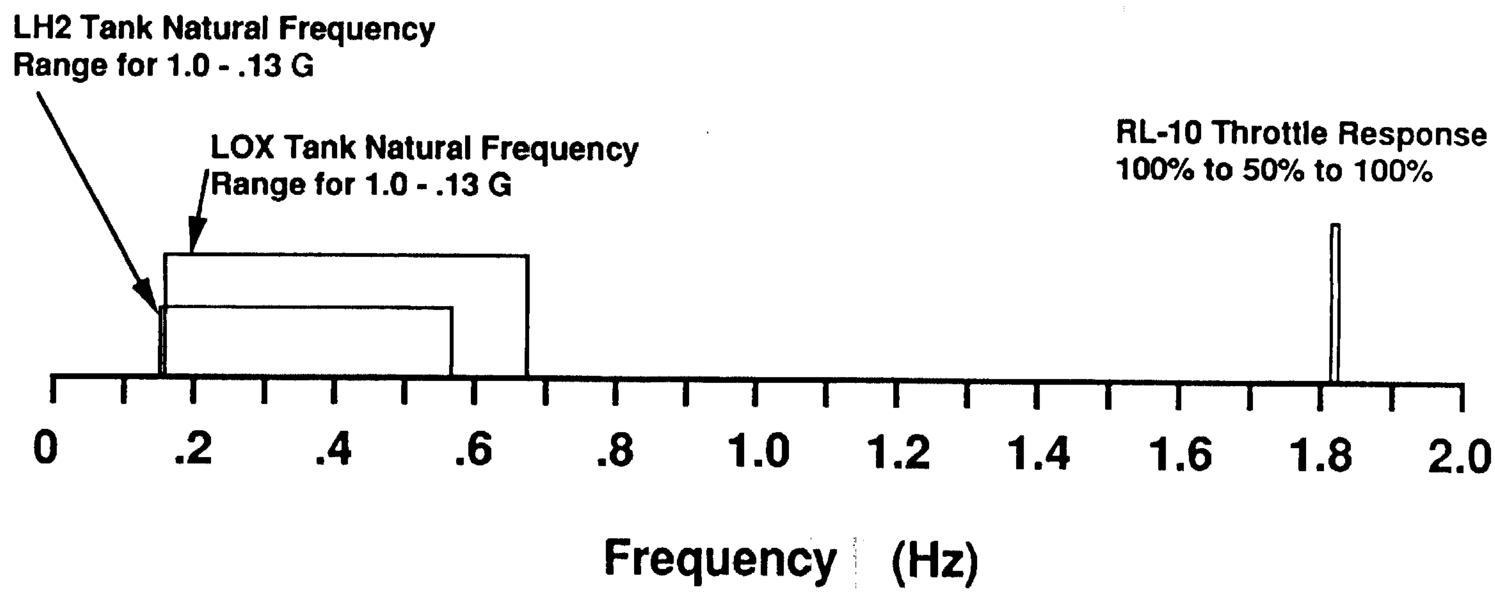
- Analysis (continued)
  - Impingement Pressure is a Function of Thrust, so Landers Larger than Apollo Will Have Higher Impingement Pressure, and Will Cause More Volume of Debris to be Displaced
- Apollo Experience
  - Surface Erosion Began as High as 80 Feet above the Lunar Surface (Apollo 15 - also the Heaviest Mission)
  - A Rock Measuring Approximately 4 by 5 Inches Was Thrown on Apollo 11

Methods to Alleviate Concern

- Prepare a Hard Surfaced Landing Site as Soon as Possible
  - Eventually, Routine Flight Operations Should Use Prepared Sites
- Place Engines in Close Proximity
  - This Will Help Minimize the "Fountain" Effect, Which May Blast Lunar Debris at the Lower Side of the Lander
- Keep Lunar Surface Flight Operations Far from Ground Operations, Until an Assured Prepared Landing Site is Available
- Use Engine Configurations that Provide Minimum Impingement Pressure (IME Engines)
  - IME Configurations Appear to Have a Significant Advantage in Reducing Lunar Debris, Due to their Much Lower Engine Exhaust Plume Impingement Pressure

Thrust Vector Control - Lunar Lander

Throttle Response vs Propellant Slosh Frequencies



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Thrust Vector Control - Lunar Lander

TVC Authority - Addressed in Mid-term Briefing

- TVC Capability of 2°
- Adequacy of 2° TVC for Space Vehicle Applications
- May Need Assistance from RCS During Landing Phase

Conclusion:

IME Can Meet Lunar Lander TVC Requirements Without Hydraulic Gimbal System

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IME Matrix - Lunar Requirements

Missions

- Mission Characteristics - Requirements/issues	Primary IME Benefit		Additional IME Benefits
• Lunar Lander →	**Meet Fault Tolerance Rqt (w/Improved Reliability)**	**+**	**Reduced IMLEO Wt (or more cargo) (Reduced Cost)**
- Dual Fault Tolerant - Multiple Burns - Throttling - Fixed P/L - Space Storage - Landing Site Prepared or Unprepared Cargo needs to be Close to Surface Plume Dispersal Dust (or Wind on Mars) - Piloted & Cargo Missions - Thermal Isolation for Cryo - TVC			- Eliminate "Fountain" at Landing - Cargo & Vehicle Closer to Surface - Lower C.G. - Improved Packaging Centerline Thrust Compact Engine - T/W & Isp Allow More Cargo (or lighter vehicle) - Eliminate Gimbal System Wt & Cost

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Requirements Satisfaction

Requirement	Satisfaction of Requirement
Dual Fault Tolerant	IME is Dual Fault Tolerant
Multiple Burns	IME Designed for Multiple Burns
Throttling	IME meets Lunar Landing Needs with 5:1 vs. 17:1 for Conventional
Fixed P/L	IME Performance Allows 2% more Payload or 2% Reduction in IMLEO
Space Storage	Designed for Space Storage
Landing Site Prepared or Unprepared Plume Dispersal Dust	IME has much Lower Plume Impingement pressures (<25% of Conv.) (0.3 psi vs 1.3 psi for conventional engines, & 1 psi for Apollo). Much Less Cratering & Debris Ejected by Exhaust Plume.
Cargo Close to Surface	IME Allows Payload to be Over 2 ft. Closer to Lunar Surface
Piloted & Cargo Missions	IME Thrust Level & Throttle Ratios More Appropriate for both Piloted & Cargo Missions than Conventional Engines
Thermal Isolation for Cryo	No Discriminator Between IME & Conventional Engines
TVC	IME Meets TVC Requirements without Gimbal System

Additional IME Benefits

IME Benefit	IME vs Conventional Engine
• Eliminate "Fountain" at Landing •	• IME - No Fountain, Low Plume Impingement Press Conventional - Significant Fountain Potential - Impingement Pressures 4 times IME
• Improved Packaging (Centerline Thrust)	• IME - Centerline Thrust Regardless of Failures Conventional - 4 of 5 Engines Not On Centerline - Most Failures Require Shutdown of Healthy Engines (opposing engine) to Maintain Centerline Thrust.
(Compact Engine)	• Pc of 1500 Allows Compact Engine (2.3 ft x 8.3 ft) IME Volume is 47% of the Volume of 5 RL10-A4's (IME = 8.3 ft dia x 2.3 ft Cylinder = 124.5 ft3) (5 RL10-A4's = 5 x 3 ft dia x 7.5 ft cylinder = 265 ft3)

JG920615-03A

Upper Stage

Agenda

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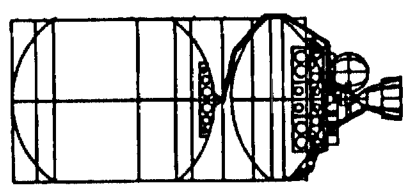
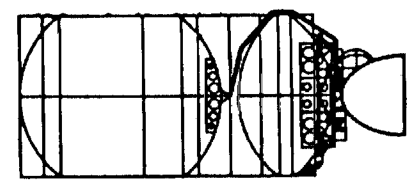
MW920602-09

IME vs. Conventional Engine Summary- Upper Stage

Conventional Propulsion



IME Propulsion



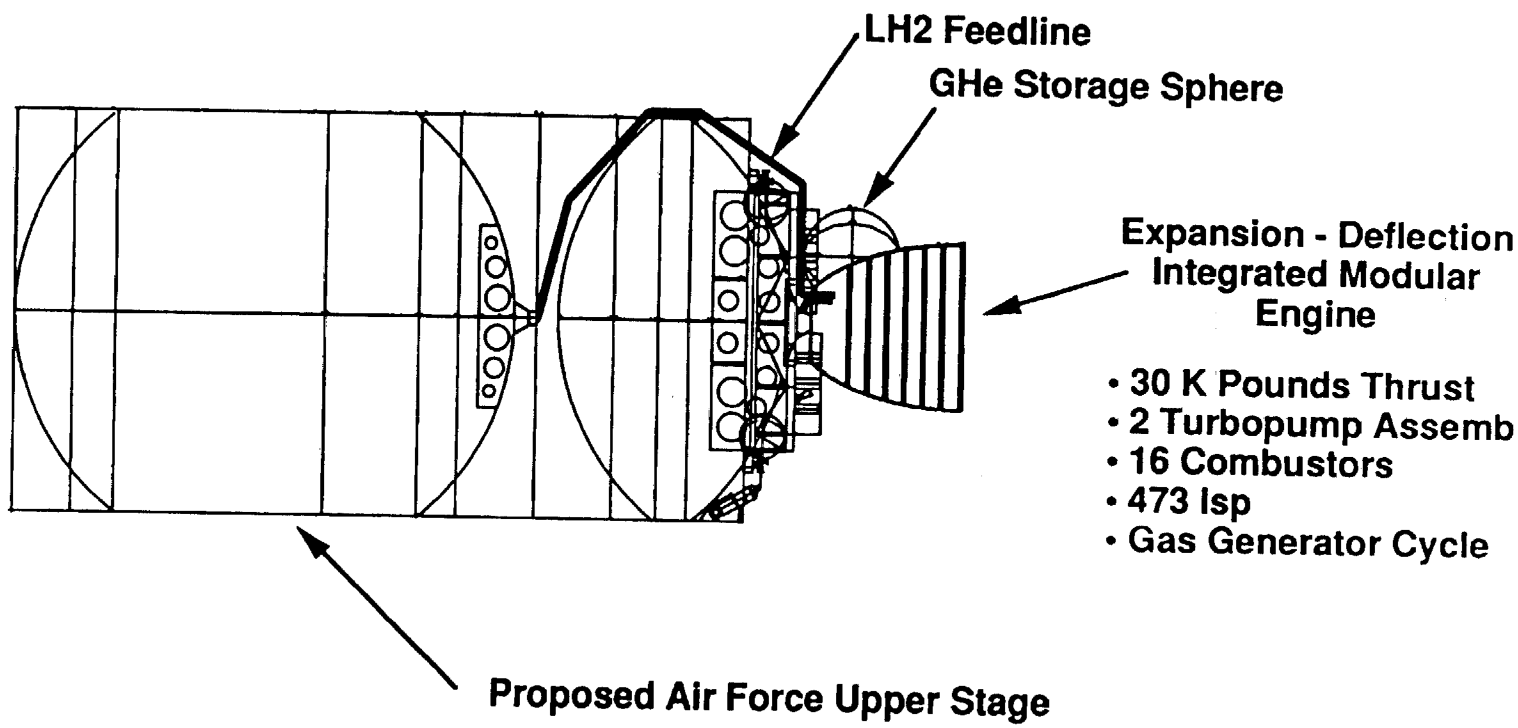
Baseline

Upper Stage Tankage, Structure, Propellant Load, and Flight Profile is the Same for Both Engine Configurations. The Only Change is to the Engine System. The Air Force Designation for the Engine is the OIME, (Operational Integrated Modular Engine).

Characteristics	Conventional Propulsion (Single Engine, RL10A-4)	IME Propulsion (E-D, 16 Combustors, 2 TPA)	Discriminators
Safety	Zero Fault Tolerance	Single Fault Tolerance	Addtl Single Fault Tolerant
Reliability (Firings/Failure)	0.9987 (769)	0.9996 (2500)	325% Improvement
Payload, Pounds to LEO	~~11,000~~ 16,000	21,000	5 ~~10~~,000 Pound Increase
Health Monitoring	Could be Incorporated	Integral	Minimal Improvement, Since Mission is Short.
Isp, seconds	449	473	24 Sec Increase
Thrust Vector Control	Gimbals	Thrust Mod and GG Exh	Eliminates Gimbal System, Replaces with Avionics.
Development Cost, ROM$	-0-	See P. 211	-
Production Cost, ROM$	3.2 M	1.5M	$1.7M Less

The IME Engine Selection for the Proposed Air Force Upper Stage was Made by Aerojet Propulsion Division under Air Force Contract FO4701-91-C-0073. Reevaluation, Organization, and Critique of this Selection Were Made Under TD-08 of NAS8-37856.

Upper Stage with Integrated Modular Engine



- 30 K Pounds Thrust
- 2 Turbopump Assemblies
- 16 Combustors
- 473 Isp
- Gas Generator Cycle

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30K E-D Engine for Air Force Upper Stage

GenCorp
AEROJET
Propulsion Division

MW 20528

ORIGINAL
OF POOR QUALITY

30K Turbomachinery Layout

GENCORP
AEROJET
Propulsion Division

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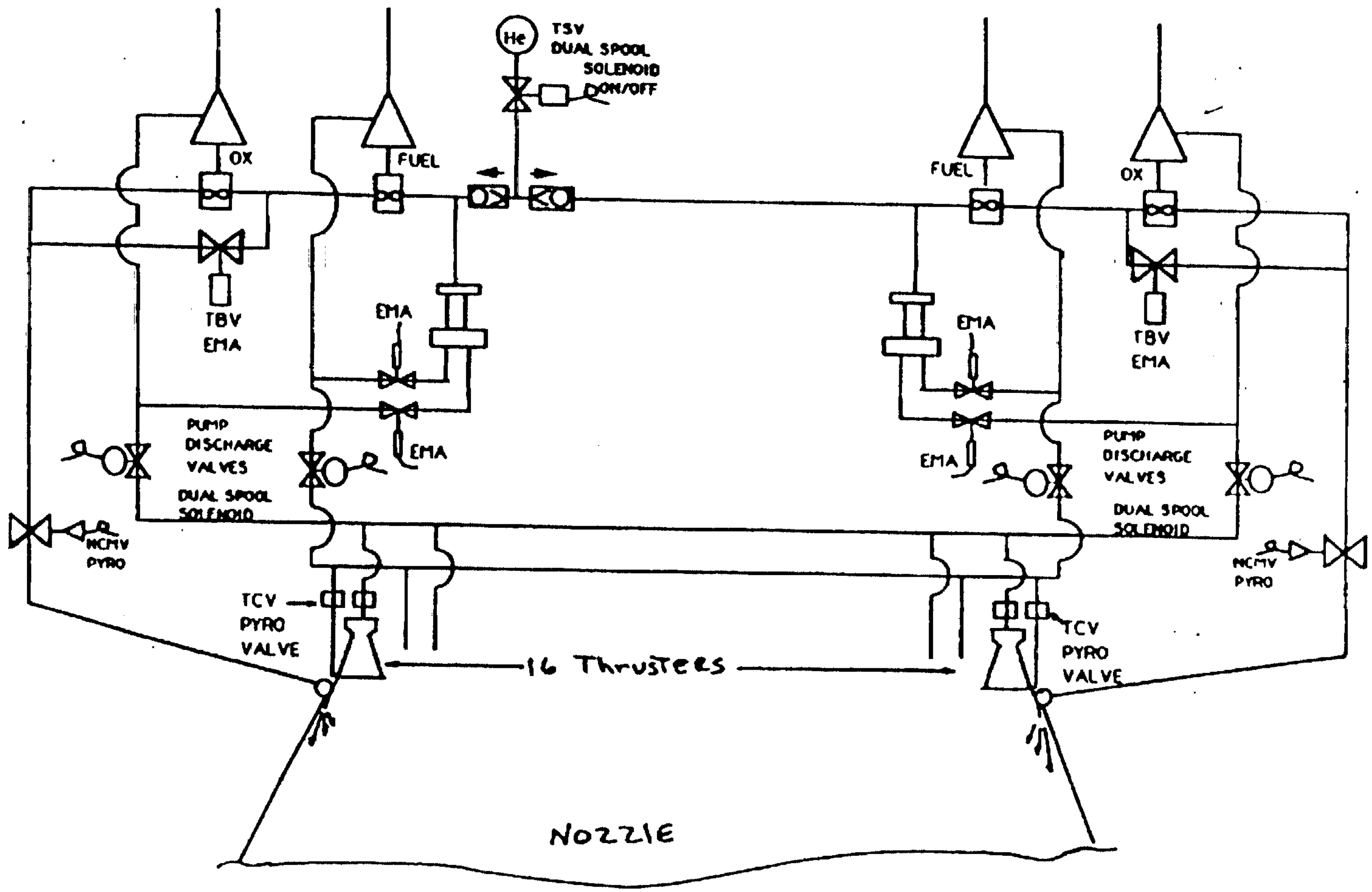
Upper Stage IME Schematic



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Agenda

• Introduction and General Overview	M. Wakefield
• TLI Stage	
- Selected Design	M. Wakefield
- Logic Behind Selection	J. Greenwood
- Requirements Satisfaction	J. Greenwood
• Lunar Lander	
- Selected Design	M. Wakefield
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- Requirements Satisfaction	M. Wakefield
• Reliability Assessment	R. Welborne
• Technology Plan	M. Wakefield
• Conclusions	M. Wakefield

Logic Behind Selection - Upper Stage

- Summary
- IME Type
- Thrust Level
- Number of Combustors
- Power Cycle
- Thrust Vector Control

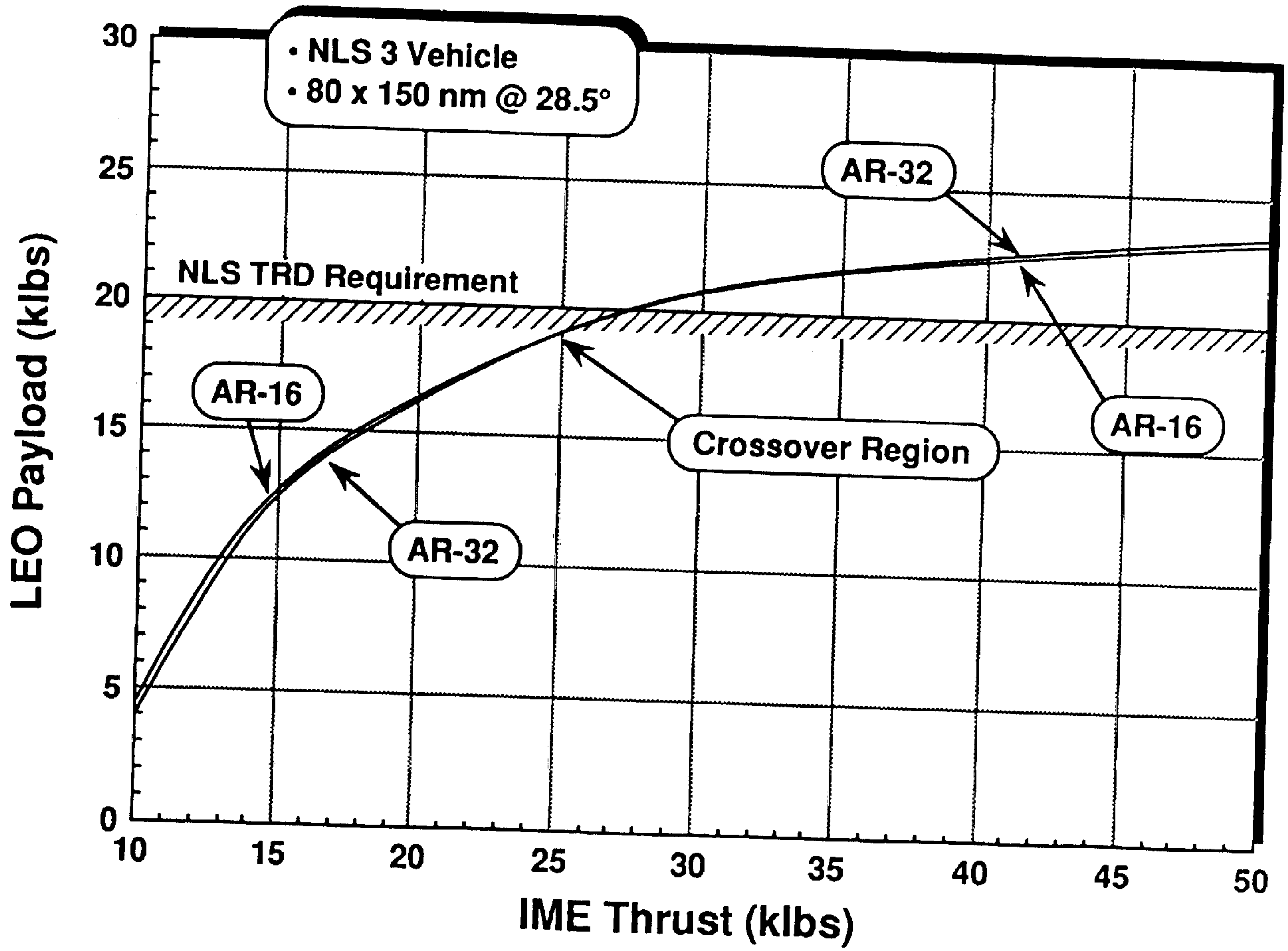
Rationale for Upper Stage IME Selection - Summary

- Expansion-Deflection Nozzle - Best Thrust-to-Weight for This Application
- 16 Combustors - Minimum Number (to Maintain Reliability) that Provides Best Thrust to Weight and TVC by Thrust Modulation
- 2 Turbomachinery Sets - Allows Single Failure Tolerance with Best Reliability
- 30,000 Pounds Thrust - Recommended by Air Force - Alternative RL10 at 20,800 Pounds Performs About Half of the Defined Missions
- Gas Generator Cycle - Result of a Weighting Matrix
- Thrust Chamber Pressure 1414 Psi - Best Fit from Parametrics Code, and Matches Ongoing Programs, Analysis, and Experience

OIME System Options

- In Vacuum - Isp is a Function of Nozzle Area Ratio
- Four Options Available - Limited Length and Diameter
  - High Pc
  - Extendible Nozzle
  - Multiple Engines - Bell Nozzle
  - Unconventional Nozzle - Plug or Expansion-Deflection
- Available Diameter Not Utilized Except with Plug & E-D
- Modular E-D and Plug Engine Concepts Selected to be Evaluated for Detail Configuration, Performance, Reliability, Operability, Cost, etc.

LEO Capability vs 2nd Stage IME Thrust Level



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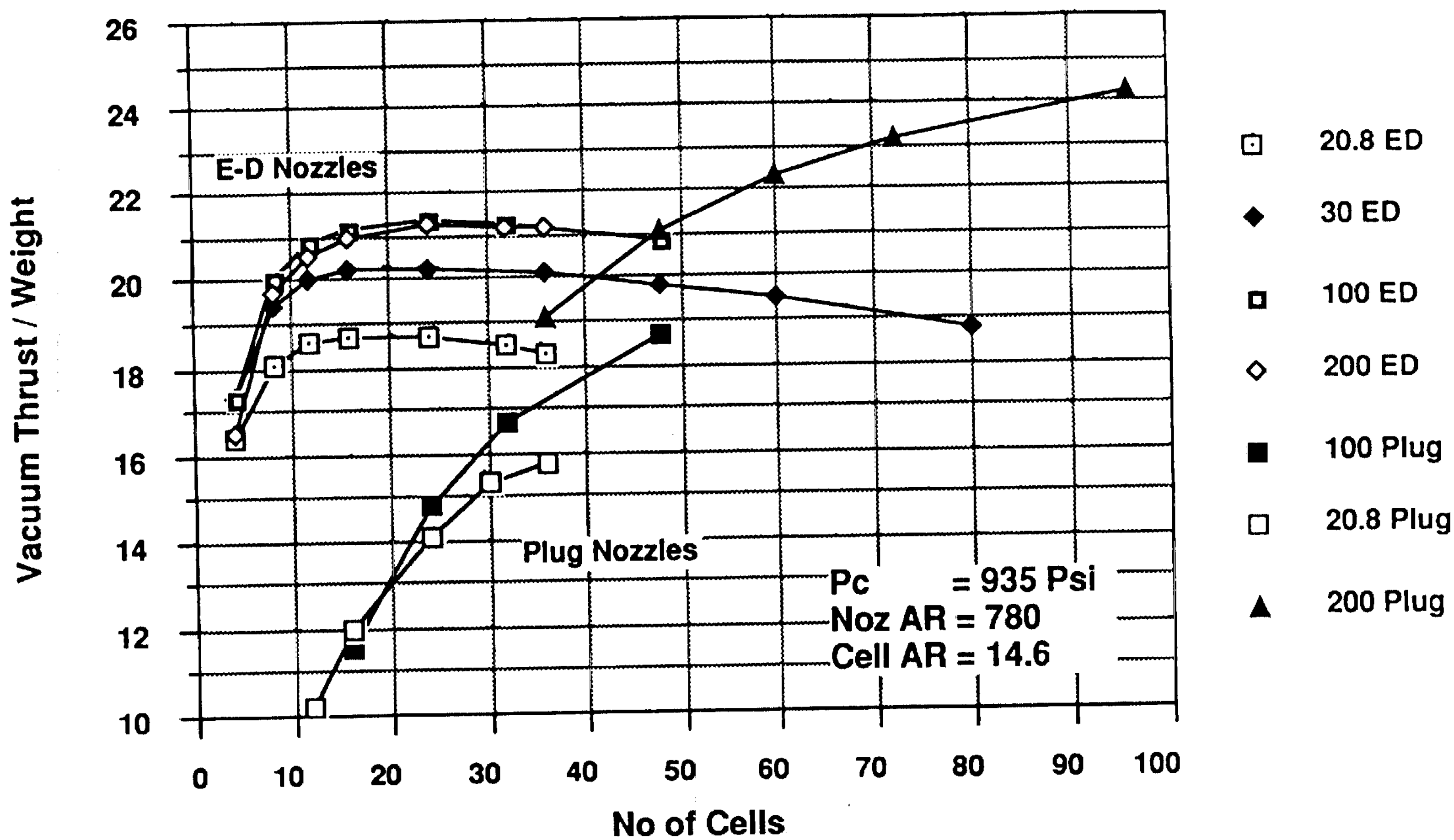
Plug and E-D Nozzle Parametrics



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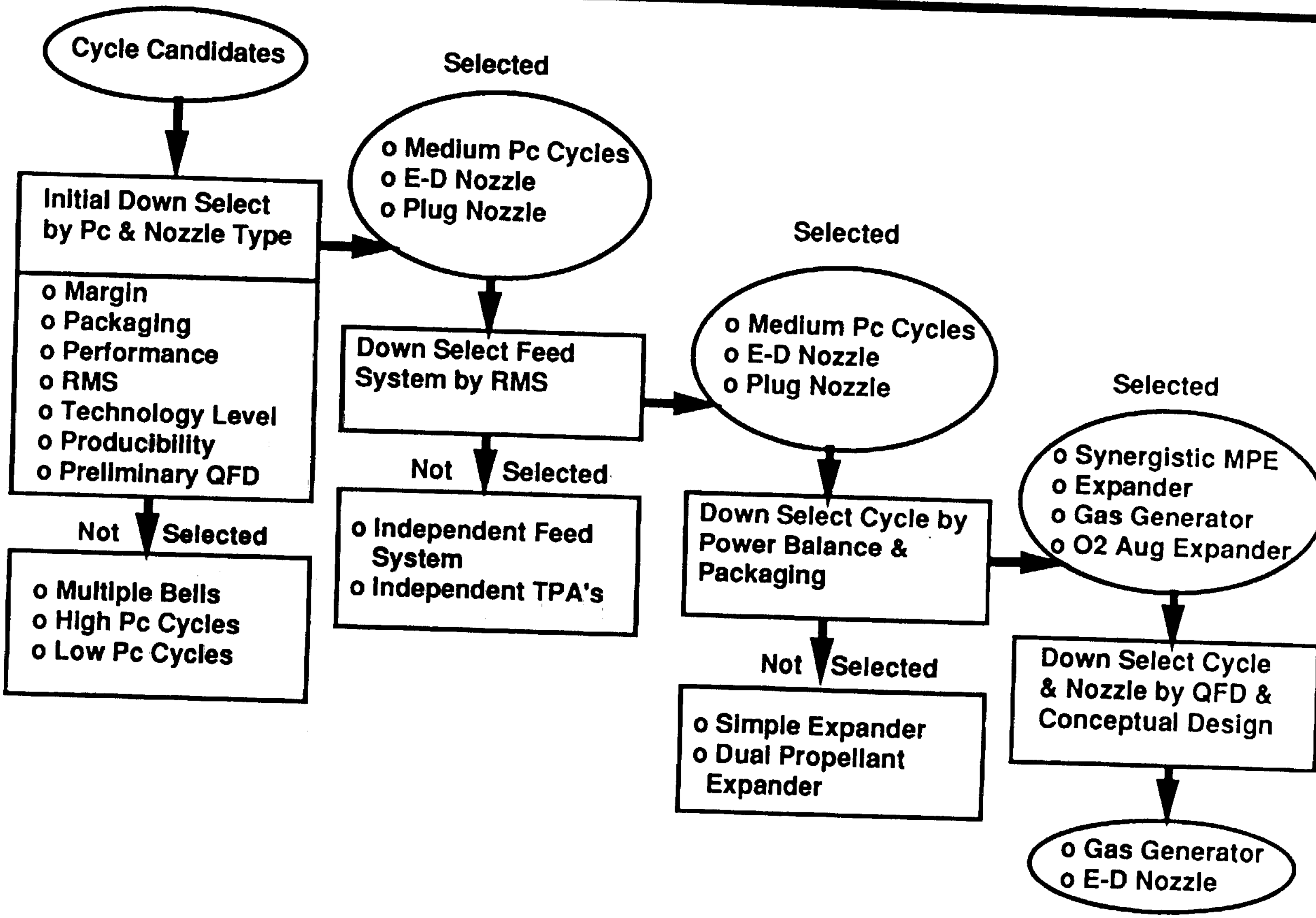
Rationale for Upper Stage IME Cycle Selection



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Upper Stage IME Thrust Vector Control

- Thrust Vector Control Logic is Similar to That Employed on the TLI Stage IME Engine
- Having a Minimum of Eight Segments in a Modular Engine Allows Thrust Vector Control to Be Effective during One or Two Segment-Out Operation. The Upper Stage IME Has Essentially 16 Segments, Due to Having Isolation Valves on all Combustors.
- For the Upper Stage, Gas Generator Exhaust is Used for Nozzle Cooling, So RCS is Used for Roll Control

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IME Matrix

Missions	Primary IME Benefit		Additional IME Benefits
- Mission Characteristics - Requirements/issues			
• **Upper Stage** →	**Improve Reliability** **(w/Weight Penalty)**	+	**Reduced Ops Cost**
- Single Engine - 20-40 Klb Thrust Rqt - Gravity Loss Sensitivity - Mission Flexibility, e.g. LEO - Single Burn GEO - Multiple Burns - TVC - Fault Tolerance Issue - Unmanned			- Elimination of Hydraulics - Elimination of Gimbal System - Increased Component Accessibility - T/W & Isp Allow More Cargo (or lighter vehicle)

Upper Stage Requirements (OIME)

Requirement	Conventional	IME
• Air Force Requirements		
- I/F with NLS Fam, and within 50K NLS T-IV Der Shroud	Yes	Yes
- Deliver 20,000 Pounds to LEO (20K NLS, Deriv. Reqt)	16 ~~11~~,000 Pounds	21,000 Pounds
- Reliability >0.995	0.9987	0.9996
- Responsiveness per ALS 1989 Requirements	Yes	Yes
- Low Cost	TBD	TBD
- Non-Reusable	Yes	Yes
- Isp ≥475 Seconds	449	473
- Mixture Ratio 6.5	5.5	6.5
- Thrust 30,000 (was 20,800 Early in Program)	No	Yes
- Eliminate On-Pad Chilldown	No	Yes
- Inflight Starts ≥3	Yes	Yes
• Additional Requirements of This IME Study		
- Single Engine	Yes	Yes
- 20-40 Klb Thrust	Yes	Yes
- Assess Gravity Loss Sensitivity	Yes	Yes
- Provide Mission Flexibility (Single vs. Mult Burns)	Yes	Yes
- Thrust Vector Control - Eliminate Gimbals	No	Yes
- Incorporate Fault Tolerance	No	Yes
- Assume Unmanned	Yes	Yes
- Improve Reliability	No	Yes

Upper Stage Requirements Satisfaction

Air Force Requirements

- Interface with NLS Family, and within 50,000 Pound NLS 2 T-IV Derived Shroud

  Both Engine Configurations Fit within the Volume Allocated for the Propulsion System. That Volume is 76 inches in Diameter by 72 inches Long, and Both Engines Fit Interchangeably on the Aft end of the LOX Tank.

- Deliver 20,000 Pounds to GEO, Using 20,000 Pound NLS 3 Vehicle

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  The Conventional 20,800 Pound Thrust Engine (Isp 450) will Place ~~11~~,000 Pounds in LEO, Using the 20,000 Pound NLS 3 Vehicle. The 30,000 Pound Thrust IME (Isp 473) Will Place 21,000 Pounds in LEO. This Represents ~~a 90~~% Increase in Capability.

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- Provide Engine System Reliability Greater than 0.995

  Reliability of the Conventional Engine System Has Been Determined to be 0.9987. Reliability of the IME Engine System Has Been Calculated to be 0.9996, a Significant Increase. A Detailed Reliability Analysis is Presented in the Reliability Section of this Report.

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Upper Stage Requirements Satisfaction (Continued)

Air Force Requirements

- Responsiveness per ALS 1989 Requirements

  Both Engine Configurations May Be Made to Meet the Responsiveness Requirements. The Conventional Engine Configuration Requires Modification Such as EMA's, While the IME Has the Opportunity to Incorporate User Friendly Operations during Initial Design.

- Low Cost

  The IME Engine System is Required to be Low Cost Relative to the Conventional RL-10 Engine System. An Assessment of Costs is Presented on the following chart.

- Non-Reusable

  Both Engine Systems Easily Meet the Requirement for Expendable Use. Engine Life Far Exceeds Single Mission Usage Requirements.

- Isp ≥475 Seconds

  The IME System will Achieve and Isp of 473 Seconds, which is 2 Seconds Short of the Requirement. This is Due to the Envelope Constraint, which Limits the Available Expansion Area, Coupled with the Desirability of Using a Moderate Combustion Chamber Pressure, and a Slightly Less Efficient Gas Generator Cycle. This is, However, a Significant Increase Over the 450 Sec Isp of the Conventional Engine System.

IME Costs

- Recurring Costs for each Engine Installation have been Estimated, and are shown on the Summary Chart Comparing each Installation to its Conventional Counterpart.

TLI Stage	$9.0M	(Includes 5.5M for Engine, and 3.5M Delta for Tank)
Lunar Lander	$3.0M	
Upper Stage	$1.5M	

- Nonrecurring Costs are Estimated to be Approximately $300M to $500M. The Least Expensive is the Upper Stage Application. The Lunar Application is More Due to Throttling, and the TLI Application is More Due to the Insulation Issue.

- The Nonrecurring Costs are <u>Very Synergistic</u>. Development of one Concept Results in Development of Most of the Others.

- The Building Block Approach (Single Cell, Group of Cells, Pump Fed Engine, Flight System) Supports Synergy.

Upper Stage Requirements Satisfaction (Continued)

Air Force Requirements

- Mixture Ratio 6.5:1

  The IME Has Been Designed to Operate at a Mixture Ratio of 6.5:1, Compared to 5.5:1 for the Conventional Engine.

- Thrust of 30,000 Pounds

  The IME Engine System is Designed for a Thrust of 30,000 Pounds, Compared with 20,800 Pounds for the Conventional Engine.

- No On-Pad Chilldown

  The IME Engine System Has Been Designed to Perform In-Flight Chilldown, and May or May Not Use Recirculation to Avoid a Performance Penalty.

- Three or More In-Flight Starts

  The IME System Has Been Designed for Multiple In-Flight Starts. Use of a Gas Generator Cycle Helps to Meet this Requirement.

Additional Requirements of This IME Study

- Single Engine

  Both the IME and Conventional Engine Systems are Single Engines, Although the IME Has the Advantage of Single Fault Tolerance.

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Upper Stage Requirements Satisfaction (Continued)

Additional Requirements of This IME Study

- Assess Gravity Loss Sensitivity
  Gravity Loss Sensitivity Was Assessed to Help Determine the Appropriate Thrust Level for the IME Engine System.
- Provide Mission Flexibility
  The IME Engine System is Designed for Multiple Starts.
- Thrust Vector Control - Eliminate Gimbals
  The IME Engine System Has Been Designed to Perform Thrust Vector Control of the Vehicle by Thrust Modulation of Individual Engine Cells.
- Incorporate Fault Tolerance
  The IME Engine System Has Been Designed to Operate at Full Power Level after the Failure of Any Component - i.e. a Turbopump Assembly, a Gas Generator, a Combustor/Nozzle Assembly, or any Valve.
- Assume Unmanned
  The Assumption That the IME Would Not Be Used for a Manned Vehicle Did Not Significantly Influence the Design. The Resulting Configuration Does Not Preclude Eventual Manrating.
- Improve Reliability
  Reliability has Been Improved Significantly, from 0.9987 to 0.9996.

Summary of Upper Stage IME Benefits

ITEM	BENEFIT	
• Dependability	Addition of Single Fault Tolerance Enhances Mission Success	
• Reliability	Increase in Firings / Failure from 769 to 2500 Represents a 325% Improvement	
• Payload to LEO (20K NLS)	Improvement from ~~11~~ 16,000 to 21,000 Pounds Represents a ~~91~~ 31% Increase	
• Specific Impulse	Improvement from 450 to 473 Represents a 5% Increase	
• Thrust Vector Control (TVC)	Incorporation of TVC by Thrust Modulation Eliminates the Gimbal System and the Associated Hydraulic System	
• Cost	Development Cost	$300M$ ROM
	Unit Cost	$1.5M$ ROM

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IME Reliability Assessment Approach

1st Half Study

- Review Engine Failure History and Current Predicted Engine Reliability Figures
- Preliminary IME Configurations Developed for Module-out Comparisons
- Fault Detection Coverage and Correlation Factors Along with Their Sensitivities Assessed
- Reliability Design Guidelines and Recommendations Presented for IME Development

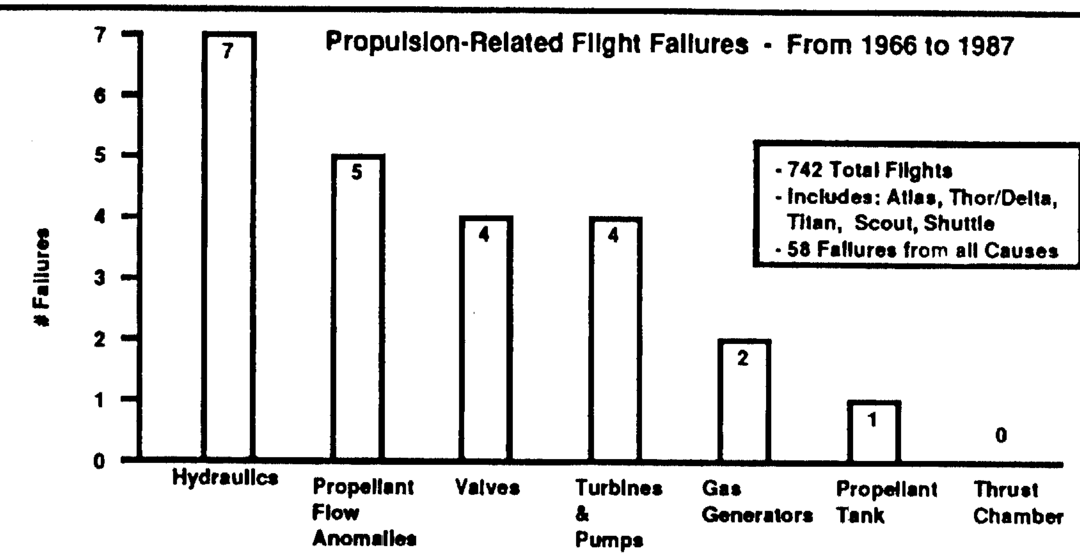
2nd Half Study

- Address Reliability Drivers
- Assess Where We are Today with Fault Tolerant and Fault Avoidance Techniques
- Identify Techniques to Increase Confidence in Reliability Results
- Reliability Calculations Provided for Both IME and Conventional Engine Configurations for the Three Classes of Vehicles.

Results Of Engine Data Study

Propulsion-Related Flight Failures Were Studied for the Period Between 1966 and 1987.

Historical Data Does Not Support Industries' Current Predicted Engine Reliability Figures.



Demonstrated Reliability	.9905	.9932	.9946	.9946	.9973	.9987	>.9987
Predicted Reliability	.9993	>.9999	.9998	.9998	>.9999	>.9999	>.9999

Techniques Needed to Increase Confidence in Predicted Reliability Results.

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Preliminary IME Configuration Reliability Results

Several Turbopump and Combuster/Nozzle Configurations Assessed for Optimum System Reliability

Results:

IME Configuration Using 2 Sets of Turbo Pump Equipment (1 Turbo-Out Capacity) is Optimum for Reliability

IME Configuration Using 3 to 4 Combuster/Nozzle (1 Combuster/Nozzle-Out Capacity) is Optimum for Reliability

Combinations of Up To 20 Combuster/Nozzles are Extremely Reliable

Above Configurations are Fault Tolerant and Can Withstand Multiple Failures

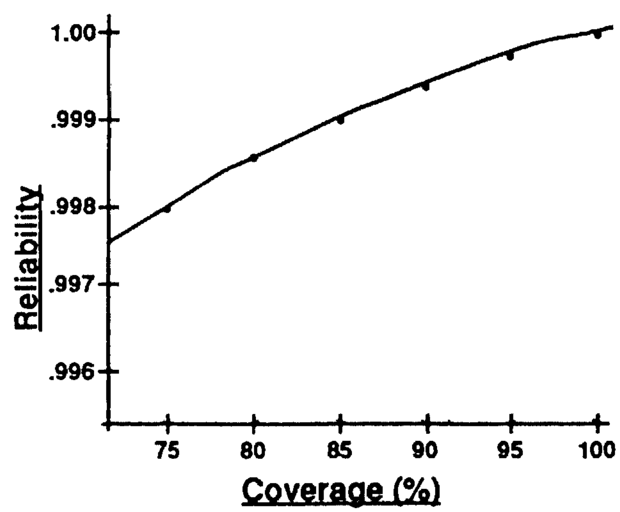
IME Reliability Assessment Preliminary Results

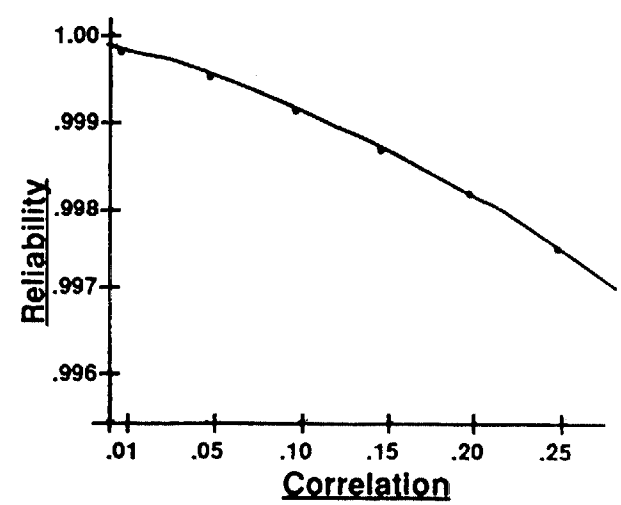
Sensitivity of Coverage and Correlation Factors Assessed

- Fault Detection and Correction Capability Key to Implementing Redundancy, Therefore Coverage Factor has Significant Impact on Engine Reliability Results

- Benefits of High Fault Detection Coverage Well Documented in Avionics Area. IME Will Require Improved Sensor Implementation to Meet High Coverage Requirements

- Very Little Historical Information on Dependent Engine Failures, However, Correlation Factor also has Large Impact on Reliability





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IME Reliability Design Recommendations

Recommendations:

Minimize Fault Propagation (Low Correlation Factor)

Maximize Fault Detection Coverage Thru Improved Parameter Sensing and Instrumentation (High Coverage Percentage)

Eliminate Hydraulic Systems

Optimize Propellant Systems to Improve Reliability

Redundant Turbo-Pump Assemblies Provide Optimum Reliability

4 to 20 Combuster/Nozzle Combinations Provide Optimum Reliability

Potential Reliability Derived IME Benefits

The Following Reliability Benefits Were Derived During the First Half of the IME Study:

- **Provides Fault Tolerant Capability With Capacity to Withstand Multiple Subsystem Failures with Less Performance Reduction than Conventional Engines**
- **Fault Tolerant Aspect Overcomes Poor Historical Performance**
- **New Design Permits Opportunity For Improved Parameter Sensing**
- **Reduced Risk For Long-Duration Manned Missions**
- **Reduced Cost Risks For Expensive Payloads and Upper Stages**

Objectives

- Develop Better Understanding of Reliability Drivers
  - Parameters Which Affect Reliability are Addressed and Techniques to Improve Confidence in Results are Presented.

- Integrated Modular Engines Compared to Redundant Conventional Engine Designs for Each of the Three Classes of Vehicles: Upper Stage, Lunar Lander and TLI.
  - Reliability Calculations Made Based Upon Selected Designs.

Reliability Drivers

Reliability Calculations Based Upon Core Statistical Equations and Reliability Drivers

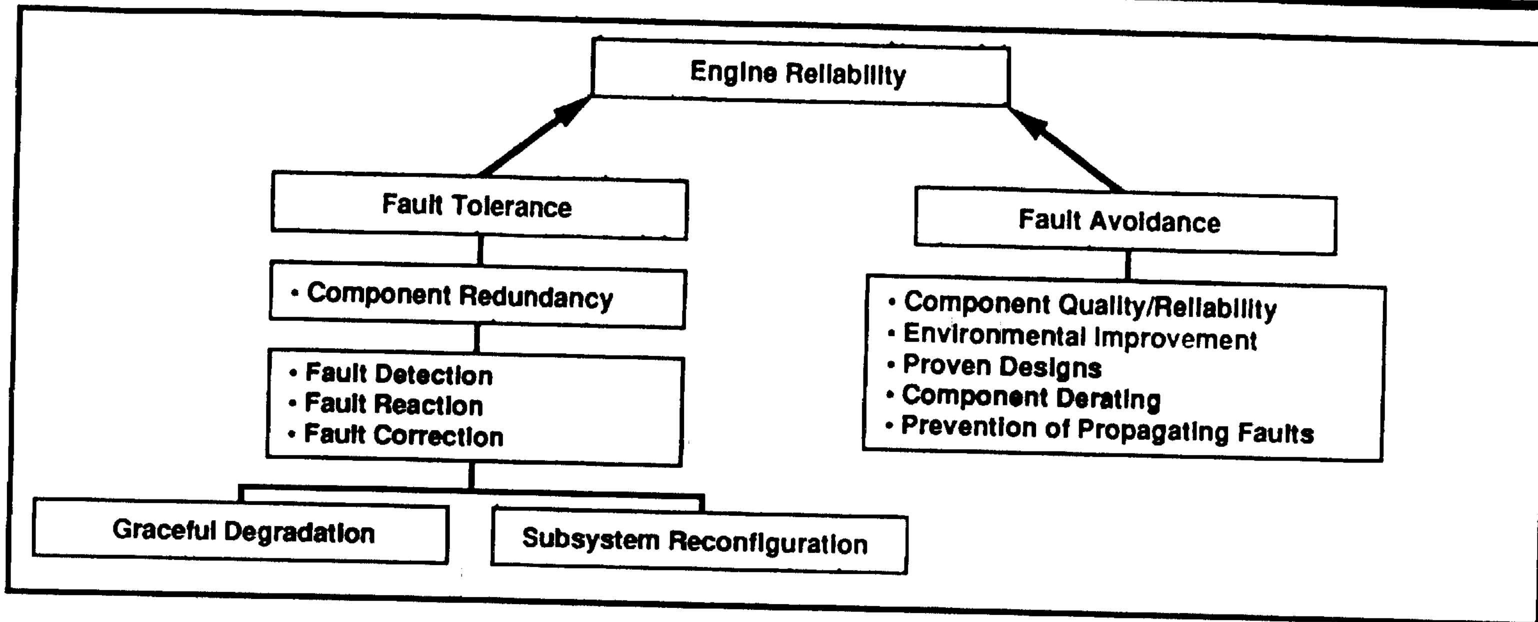
Rsystem = Σ (Core Statistical Equations) x (Reliability Drivers)

Rsystem	Σ (Core Statistical Equations)	(Reliability Drivers)
System Reliability	Statistics Equates Only to Number of Components or Modules Selected and How they are Interconnected	Reliability Drivers are Parameters Which Bias the Statistical Results. Examples of Reliability Drivers: Time Environment (Thermal, Vib, etc.) Operating Procedures Margin Quality of Components Proven Design (Maturity) Level of Testing
System Reliability is Equally Influenced by Both Statistics and Reliability Drivers	Reliability Statistics are Mature Mathematically Derived Formulas	Reliability Drivers are System and Vehicle Design Dependent, as well as Mission Dependent

Accuracy of Reliability Calculation Only as Good as Fidelity of Reliability Driver Information

By Addressing Reliability Drivers with Modular Engine Design, Higher System Reliability can be Achieved than with Conventional Engine Designs

Reliability Drivers



- Dual Path Exists for Improving Reliability Performance
  - Fault Tolerance
  - Fault Avoidance
- Either Path or Pieces of Each Path can be Implemented Separately
- Optimum Approach Follows Both Paths Simultaneously Using Selected Elements of Each
- New IME Design Provides Opportunity to "Design-in" Reliability by Following Optimum Approach

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Reliability Drivers cont'd

Level of Implementation and Techniques Affect Reliability Performance

Fault Tolerance Techniques	Where are We Today?	Techniques for Improving Reliability Confidence
Use of Redundant Components: (Fault Detection, Reaction and Correction)	Engine Reliability Studies have Based Calculations on Occurrence of Benign or Catastrophic Failures Only. This Is Due In Part to Several Reasons: - Launch Vehicles Being the Only Ones to Implement Redundancy. - The Short Mission Duration of these Vehicles (10 min.) has not Required Extensive use of Fault Detection Schemes. - Few Vehicles Have Been Manned. - Orbital Returns Used as Redundancy or Abort Modes in Place of Additional Engines.	The Key to Successfully Implementing Redundant Engine Configurations for Long Duration Missions is the Detection and/or Prediction of Faults and the Rapid Reaction or Prevention of Catastrophic Events. Fault Tolerance Must be Designed-in to Future Propulsion Systems so that Their Level of Reliability Matches the Importance of the Mission.
	RECMS Arrived at an Average Industry Coverage Rate of 78% However, Limited Knowledge Available on Current Level of Attainable Coverage. Level of Analysis and Testing to Obtain this Knowledge is Significant and Costly. Level of Fault Coverage will be Weighed Against Cost. RECMS Recommended a Coverage of No More than 90%, for Launch Vehicles, Based on a Life Cycle Cost Analysis.	Level of Fault Coverage can be Determined by using the Following Methodology: - Perform Detailed Engine FMEA for All Mission Phases. - For Each Failure Mode Identified, Ascertain If/How Detected by Engine Health Mgmt. Design (Failure modes must be detectable within time frame that allows for reaction). - Extensive Testing of Engine and Engine Health Mgmt. System Required to Prove Technologies and Provide Confidence in Fault Detection Techniques. We Must Know VHM Costs and Maximum Level of Coverage Obtainable with Current or Near-Term Technology Before an Accurate Coverage vs. Reliability vs. Cost Trade can be Performed.

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Reliability Drivers cont'd

Fault Avoidance Techniques	Where are We Today?	Techniques for Improving Reliability Confidence
Reduction of Environmental Stresses:	Environmental Stress Greatly Impacts Reliability Results. Thermal and Vibration Environments not well Defined for New and/or Redundant Designs.	Extensive Analysis and Testing will be Required to Assure That Each Material And Design Chosen for the Engine is Properly Selected to Operate Under the Known Environments. Application of Modern Tools, Such as Finite Element Analysis for Static, Dynamic and Thermal Stresses Allow Improved Understanding and Quantification of Environments.
Component Derating (Margin):	Margin Greatly Impacts Reliability. For Example; an Engine Designed for 30,000lbs of Thrust Operating at 28,000lbs will be Less Reliable than the Same Engine Operating at 15,000lbs. Insufficient Data is Currently Available to Derive Reliability Factors for Engine Margins.	Analysis and Testing needs to be Done to Achieve a Better Understanding and Confidence of the Reliability Impacts Associated with Levels of Engine Operation and Margin.
Extensive Environmental Test:	Well Documented Evidence Exists that Shows Environmental Testing Discloses Weak Components and Uncovers Workmanship Defects.	Since Design and Manufacturing Defects are in Effect Attributes of Specific Components and Not a Function of Inherent Life, the Application of Environmental Testing Can Result in Significant Reliability Improvements.
Use of Proven designs:	Proven Flight Tested Designs are Inherently More Reliable.	New Engine Design Should Take Advantage of Proven Components to Reduce Risks Associated with New Technologies and Materials.
Fault Propagation Preventation: (Detection of Impending Failures, Shielding)	New Advanced Technologies are Being Studied and Developed by all of the Rocket Engine Manufacturers for More Effective Engine Monitoring.	These New Technologies Need to Be Tested and Integrated into Propulsion Systems in Order to Take Advantage of Their Expected Reliability Improvements. A New Engine Design Provides an Opportunity to Implement These New Technologies.

IME vs. Conventional Engine Reliability Comparison

The Integrated Modular Engine was Compared to a Conventional Engine Approach for the Three Classes of Vehicles. The Following Data and Assumptions Were Used for the Comparisons:

- Rocket Engine Conditioning Monitoring Study (RECMS) Data Baselined for IME Calculations
- RL-10 Engine Reliability Baselined for Conventional Engine Calculations (Data are Very Similar to RECMS and Provided by P&W)
- Modified Binomial Equation Used for Calculations
- For Redundant Configurations, Coverage Factor of .95 and Correlation Factor of .03 Used as Design Goals
- Assumed that 5 RL-10s Required for Two Fault Tolerance.
- Assumption Made that Smaller Combusters Manufactured in Large Quantities will Have Inherently Higher Reliability

IME vs. Conv. Engine Reliability for TLI Stage

Integrated Modular Engine	Conventional Engine
Reliability: .9988 Successful Engine Firings per Failure: 833	**Reliability:** .9994 Successful Engine Firings per Failure: 1667
Reliability Based on Design with 4 TPA and 64 Combusters. .9994* Reliability for Turbo Pump Assembly. .9998** Reliability for Combuster/Nozzle Set. Reliability of Valves, Lines, Actuators and Manifolds Taken from Industry*. Fault Detection Coverage of .95 Assumed and Correlation Factor of .03 Used. These Values are Goals for New Design.	RL-10X (5 engines) Chosen for Conventional Engine Analysis. Demonstrated RL-10 Reliability (all configurations w/no Failures) Baselined, with Increase in Failure Rates Assumed for Additional Valves, Feedlines, Sensors, Engine Health and Redundancy Management Capability, etc. Fault Detection Coverage of .95 Assumed and Correlation Factor of .03 Used. These Values are Goals for New Design.
Range: .9930 9988	**Range:** .9960 9994
Engine Reliability Calculations are Sensitive to Many Variables. Considering all or a Subset of the Following Parameters, and their sensitivities, will Result in the Above Range of Reliability for the Upper Stage Design: % Fault Coverage, Correlation Factor, Confidence Limits in Rel. Calculation, Maturity of Technology, etc.	If Atlas/Centaur-70 Failure Attributed to RL-10, the Reliability is .9989 and Successful Engine Firings per Failure to Fire Becomes 909. RL-10C is Also Sensitive to the Same Variables as the IME Design.
Other Considerations: IME Design is Dual Fault Tolerant, but Generally Provides Higher Thrust After Failure(s).	Other Considerations: New Design for 35K Thrust, Application of Historical Reliability May not be Appropriate. RL-10C TLI Stage Design is Dual Fault Tolerant.

References:
* Rocket Engine Conditioning and Monitoring Study (RECMS), Prime Contractor-Pratt and Whitney.
** RECMS Data Baselined w/Reliability Improvements Assumed as a Result of Projected Increase in # of Production Units.
*** Data Provided by Pratt & Whitney 4/92.

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IME vs. Conv. Engine Reliability for Lunar Lander

Integrated Modular Engine	Conventional Engine
Reliability: .9995 Successful Engine Firings per Failure: 1957	**Reliability: .9994** Successful Engine Firings per Failure: 1667
Reliability Based on Design with 4 TPA and 16 Combusters. .9994* Reliability for Turbo Pump Assembly. .9998** Reliability for Combuster/Nozzle Set. Reliability of Valves, Lines, Actuators and Manifolds Taken from Industry*. Fault Detection Coverage of .95 Assumed and Correlation Factor of .03 Used. These Values are Goals for New Design.	RL-10X (5 engines) Chosen for Conventional Engine Analysis. Demonstrated RL-10 Reliability (all configurations w/no Failures) Baselined, with Increase in Failure Rates Assumed for Additional Valves, Feedlines, Sensors, Engine Health and Redundancy Management Capability, etc. Fault Detection Coverage of .95 Assumed and Correlation Factor of .03 Used. These Values are Goals for New Design.
Range: .9950 9995	**Range: .9960 . . . . . .9994**
Engine Reliability Calculations are Sensitive to Many Variables. Considering all or a Subset of the Following Parameters, and their sensitivities, will Result in the Above Range of Reliability for the Upper Stage Design: % Fault Coverage, Correlation Factor, Confidence Limits in Rel. Calculation, Maturity of Technology, etc.	If Atlas/Centaur-70 Failure Attributed to RL-10, the Reliability is .9989 and Successful Engine Firings per Failure to Fire Becomes 909. RL-10X is Also Sensitive to the Same Variables as the IME Design.
Other Considerations: IME Design is Dual Fault Tolerant, but Generally Provides Higher Thrust After Failure(s). Less Susceptible to Plume Impingement on Lunar Surface.	Other Considerations: New Design for Throttling & VHM, Application of Historical Reliability Data May Not be Appropriate. RL-10X Lunar Lander Design is Dual Fault Tolerant.

References:
* Rocket Engine Conditioning and Monitoring Study (RECMS), Prime Contractor-Pratt and Whitney.
** RECMS Data Baselined w/Reliability Improvements Assumed as a Result of Projected Increase in # of Production Units.
*** Data Provided by Pratt & Whitney 4/92.

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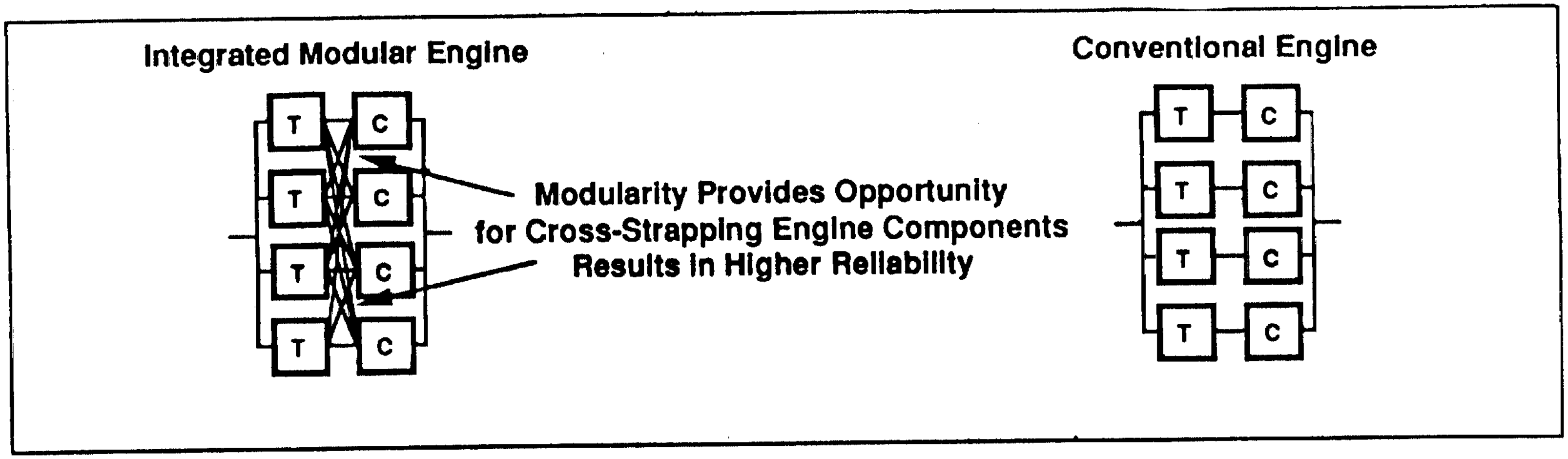
IME vs. Conv. Engine Reliability for Upper Stage

Integrated Modular Engine	Conventional Engine
Reliability: .9996 Successful Engine Firings per Failure: 2500	**Reliability:** .9987 Successful Engine Firings per Failure: 769
Reliability Based on E-D Design with 2 TPA and 16 Combusters. .9994* Reliability for Turbo Pump Assembly. .9998** Reliability for Combuster/Nozzle Set. Reliability of Valves, Lines, Actuators and Manifolds Taken from Industry*. Fault Detection Coverage of .95 Assumed and Correlation Factor of .03 Used. These Values are Goals for New Design.	RL-10A-4 Chosen for Conventional Engine Analysis. Demonstrated RL-10 Reliability (all configurations) w/no Failures, and based 81 Missions, 186 Engines and 384 Firings. If Atlas/Centaur-70 Failure Attributed to RL-10, the Reliability Is .9974 and Successful Engine Firings per Failure to Fire Becomes 385.
Range: .9975 9996	**Range:** .9900 9987
Engine Reliability Calculations are Sensitive to Many Variables. Considering all or a Subset of the Following Parameters, and their sensitivities, will Result in the Above Range of Reliability for the Upper Stage Design: % Fault Coverage, Correlation Factor, Confidence Limits in Rel. Calculation, Maturity of Technology, etc.	P&W Publishes a "Predicted" RL-10 Reliability of .9996*** Based Upon Calculations using the Total Number of Accountable Test and Flight Firings (~1700). The Lower End of the Reliability Range Is Based Upon the Demonstrated RL-10 Reliability (w/one failure) using a 90% Confidence Bounds.
Other Considerations: IME Design Is Single Fault Tolerant	Other Considerations: RL-10 Upper Stage Design Is Zero Fault Tolerant. RL-10A-4 Does Not Have Thrust Sufficient for All Defined Missions. Application of Historical Reliability Data May not be Appropriate for Higher Thrust.

References:
* Rocket Engine Conditioning and Monitoring Study (RECMS), Prime Contractor -Pratt and Whitney.
** RECMS Data Baselined w/Reliability Improvements Assumed as a Result of Projected Increase in # of Production Units.
*** Data Provided by Pratt & Whitney 4/92.

Reliability Conclusions and Recommendations

The Ability to Cross-Strap Engine Components is the Prime <u>Inherent</u> Reliability Benefit of the Integrated Modular Engine



Other Potentially Significant Reliability Benefits or Gains:

- **Withstands Multiple Subsystem Failures with Less Performance Reduction than Conventional Engines**
- **Smaller Combusters Manufactured in Large Numbers Could be Very Reliable**
- **IME Offers Opportunity to Implement Thrust Vector Control without the use of Hydraulics, EMAs, etc.**

Reliability Conclusions and Recommendations (cont'd)

- Statistical Analysis of IME and Preliminary Reliability Assessments Provide Promising Results Indicating Modular Concept Can Be More Reliable than Redundant Conventional Engine Design.

  - A Detailed Reliability Analysis is Required During a Phase B/C Program to Ascertain a More Definitive Assessment of the Reliability Performance for the IME.

- Engine Health Management is an Enabling Technology for Missions of Extended Duration Requiring Engines with Redundant Components Which are Subjected to Multiple Engine Firings. The Ability to Detect Impending Failure Modes is the Key Element to Successfully Implementing a Redundant Conventional or IME Design.

  - A Detailed Engine Health Analysis Study (Order of Magnitude of the RECMS) is Recommended for Engine Configurations Applicable to the Next Generation of Upper Stage, TLI and Lunar Lander Vehicles.

RW920612-17B

Agenda

• Introduction and General Overview	M. Wakefield
• TLI Stage	
- Selected Design	M. Wakefield
- Logic Behind Selection	J. Greenwood
- Requirements Satisfaction	J. Greenwood
• Lunar Lander	
- Selected Design	M. Wakefield
- Logic Behind Selection	J. Greenwood
- Requirements Satisfaction	J. Greenwood
• Upper Stage	
- Selected Design	M. Wakefield
- Logic Behind Selection	M. Wakefield
- Requirements Satisfaction	M. Wakefield
• Reliability Assessment	R. Welborne
• **Technology Plan**	**M. Wakefield**
• Conclusions	M. Wakefield

MW920602-13

Approach to Technology Plan

- Plan IME Implementation for "Best Fit" Vehicle
- Identify all Required Technologies
- Estimate the Scope of Implementing IME
  - Facilities
  - Labor
  - Materials
- Prepare an Integrated Schedule
- Identify Technologies for Growth to Other IME Applications

Best IME Application

- TLI, Lunar Lander, Upper Stage
  - Performance is Considerably Improved. Reliability and Safety are Maintained or Improved.

- Lunar Lander Distinction
  - Additional Advantages for Lunar Lander Application Are: 1. Site Alteration Reduction - a Significant Driver for Viking and Apollo Missions, and 2. Reduced Landed Height of Payload and Personnel.

- The Technology Plan Will Key off of Lunar, and Pick up Other Applications as Appropriate
  - System Performance Requirements and Issues
  - Component Requirements and Issues

IME System Technology Issues

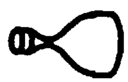
To Support Lunar Lander Vehicle

Item	Issue	Where Are We Today
Efficiency of Large Plug Nozzles	High Expansion Ratio Truncation Efficiency Effect of GG Filling Exhaust Plume Boundary Layer Growth	Basic Codes Exist. Codes Need Contemporary View and Need to Be Validated.
Thrust Vector Control of Plug Custer	Thrust Vector Angle and Vector Offset Module(s) Out Control Effect on Total Thrust Effect of Engine Throttling on TVC Capability Cross Connection Between TVC and Throttling	Codes Not Known to Exist. Need to Be Developed and Validated.
Throttling Behavior of Plug Custer	Efficiency During Throttling Effect of TVC on Engine Throttling (Assume Combustor on/off)	Codes Not Known to Exist. Need to Be Developed and Validated.
Landing Site Impingement	Soil Displacement Soil Fluidization Effects of TVC and Engine out Throttling	Codes are Contemporary, Need to Generate Data for Specific Applications

IME System Technology Issues (Cont)

Item	Issue	Where Are We Today
Startup Transient	Plume Behavior Time History of Thrust Vector Ignition of Multiple Chambers	Codes Not Known to Exist. Need to Be Developed and Validated
Gas Generator Start	Elimination of Start Gas Supply, Throttling	Conventionally Requires Gaseous Helium or Start Cartridge. Bootstrap Doable, but Requires Development.
Reliability	Assessment of Non Conventional Engines and Components Mission Profile and Environmental Issues	New Hardware Requires New Database. Available Data Primarily Launch Vehicle Related.
VHM Planning and Integration	VHM Architecture Sensor Technologies Engine Controller Integration into ICHM	Preliminary Strategies Demonstrated
To Support Other Vehicle Applications		
Efficiency of Large E-D Nozzles	Boundary Layer Growth, Channeling of Combustor Exhausts	Codes Require Test Validation
Fluid Mechanics Transient Analysis	System Interactions, Throttling, TVC, Failures	Top Level Analysis

IME Plug Nozzle Component Technology Issues

Component		Available Now	Required Enhancement	Next Generation Enhancement
Structure		Composites, Berylium Alloys Al/Li Alloys	None Required	Metal matrix Composites
Combustors (Injector, Chamber, Primary Nozzle)		NASP Platelet, Steel Clad	Formed Platelet	Metal Matrix Composites Cladding
Plug Nozzle		None at This Size	Large C/SiC Nozzles	TBD
TPA's		RL-10 600 Psi RS-44 1540 Psi	1500 Psi, Hydrostatic Brgs, Broad Range, or AETB 1500 Psi	AJ 4Stg, Dual Spool, Hydro Brgs
Feedlines		Steel, Aluminum, Composites	None Required	Metal Matrix Composites
Igniters		Intermittent Spark	Continuous Spark	Laser, or Other Advanced Tech.

MW920614-04

IME Plug Nozzle Component Technology Issues (Cont)

Component	Available Now	Required Enhancement	Next Generation Enhancement
Gas Generator	None in this Range	Design for Small Engines - Bootstrap Start, Low or no GHe	TBD
Valves	Helium or Propellant Operated	Electro-Mechanical Operators	TBD
Controllers	Conventional	VHM Compatible Fault Tolerant	Neural Network
Instrumentation	Conventional	High Reliability VHM Compatible Non-Intrusive	Smart Sensors

MW920615-01

IME "Other" Nozzle Component Technology Issues

Component		Available Now	Required Enhancement	Next Generation Enhancement
Insulation for Using Existing Tank Surfaces for Expansion		None	NASP Derivatives or Metallic MLI	TBD

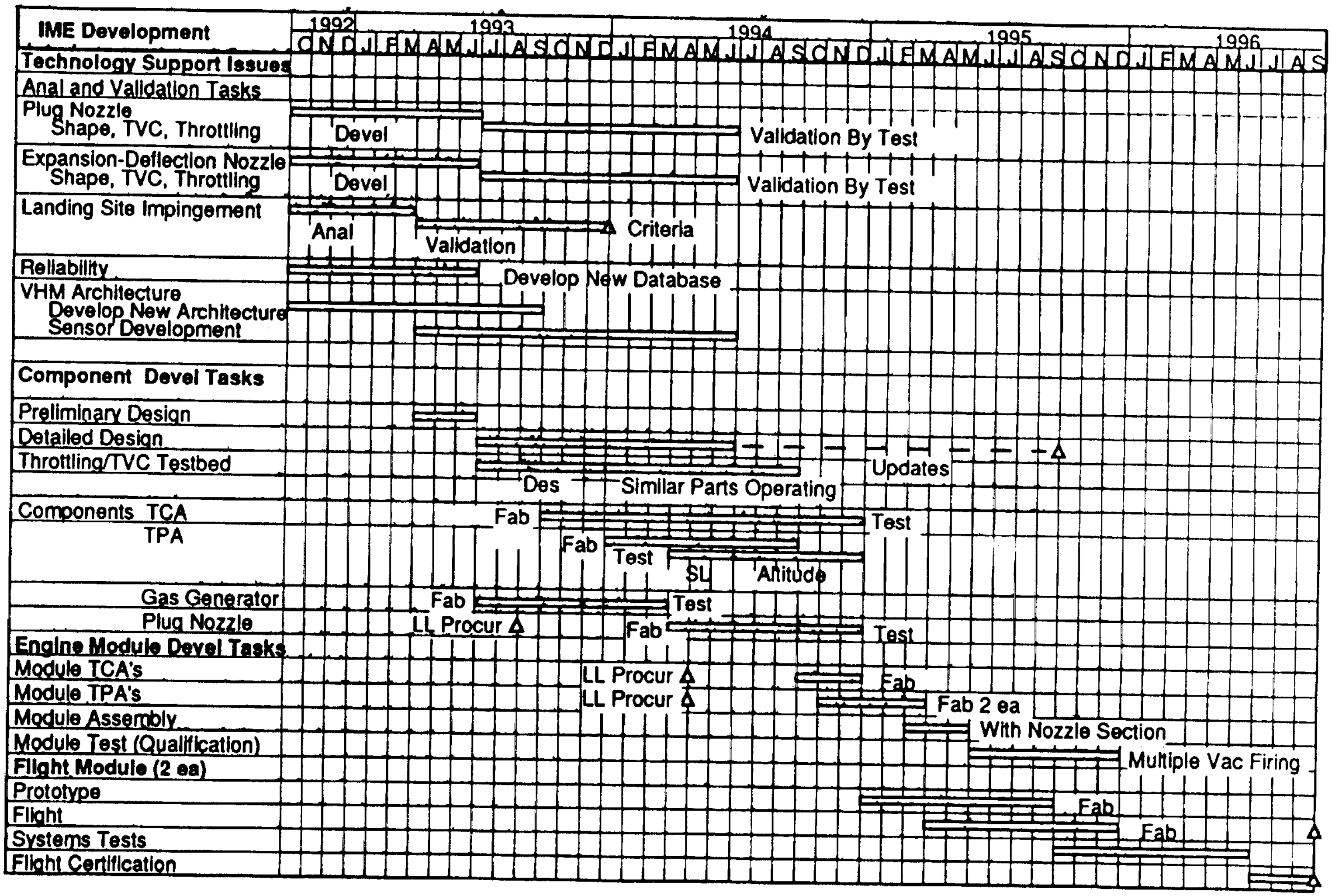
MW920615-03

IME Thruster Commonality

Vehicle	TLI	Lunar Lander	Upper Stage
Thrust, Lbf	2734	2063	1875
Area Ratio	30:1	30:1	16:1
Chamber Pressure, Psi	1500	1500	1414
Exit Shape, in	2.1X13.7	1.5X13.5	TBD

Fabrication of an IME for One Application May Provide Hardware that is Applicable to Other Applications

IME Technology Schedule



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Agenda

• Introduction and General Overview	M. Wakefield
• TLI Stage	
- Selected Design	M. Wakefield
- Logic Behind Selection	J. Greenwood
- Requirements Satisfaction	J. Greenwood
• Lunar Lander	
- Selected Design	M. Wakefield
- Logic Behind Selection	J. Greenwood
- Requirements Satisfaction	J. Greenwood
• Upper Stage	
- Selected Design	M. Wakefield
- Logic Behind Selection	M. Wakefield
- Requirements Satisfaction	M. Wakefield
• Reliability Assessment	R. Welborne
• Technology Plan	M. Wakefield
• Conclusions	M. Wakefield

MW920602-14

Conclusions

- The IME Can, Due to Redundant Components, Have Better Reliability.
- TVC Can Be Met by Differential Throttling of an IME.
- Use of Existing Vehicle Surfaces for Expansion Increases Performance, but Raises Insulation Issues.
- Each of the Three IME Applications Developed Showed Benefit.
- The Lunar Lander Application was Found to be Most Beneficial.
  - Reduced Site Alteration
  - Lower Payload and Crew
- The Site Alteration Issue May Make Use of a Conventional Engine System Impractical for the Large Lunar Landers Envisioned. A Study is Needed Which Establishes Criteria.
- Many of the Components Envisioned During This Study are of a Size and Range that Development of a Single IME Would Likely Make Others Practical.
- Development of a New Engine Represents an Opportunity to Properly Incorporate VHM, Which Helps Reinforce Reliability.

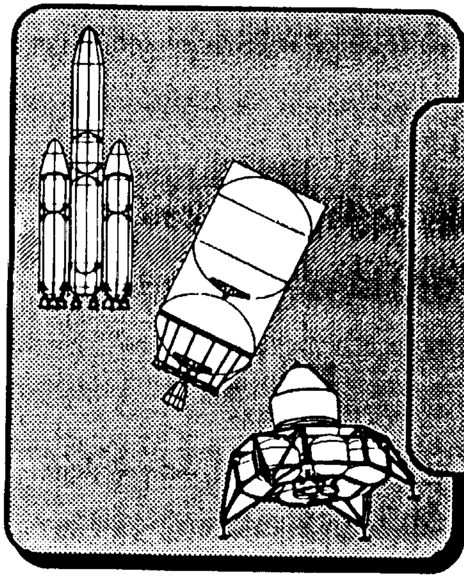
Conclusions (Continued)

- Technology Steps Advocated Provide an Orderly Approach to Development of an IME Philosophy and Hardware that Can Become the Next Generation of Chemical Propulsion Engines.

- The Cost of Obtaining Failure Signatures to Support VHM Will be More Costly for Conventional Engines than for IME Engines.

Technical Directive 09

Upper Stage Evolution Study



Technical Directive 09 Overview

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Topics

MSFC

- Introduction
- Task Descriptions
- Key Groundrules & Assumptions
- Design Reference Missions (DRMs)
- Configuration Summaries
- The STV Upper Stage Strategic Plan

Introduction - *Purpose*

MSFC

- Incorporate the Lessons Learned from Previous Upper Stage Work
- Gain a Better Understanding of the Government's Upper Stage Needs
- Identify the Growth and Commonality Issues Associated with the Next Generation of Upper Stages
- Evaluate Potential Upper Stage System and Subsystem Concepts
- Support MSFC's Study of the First Lunar Outpost

Initial Task - *Review of Previous Work*

Groundrules and Assumptions

- Are They Still Valid?
- Should They Be?
- Are Any Missing?

System Drivers

- Why Do They Drive the System?
- How Do They Drive the System?
- Should They Drive the System?

Lessons Learned

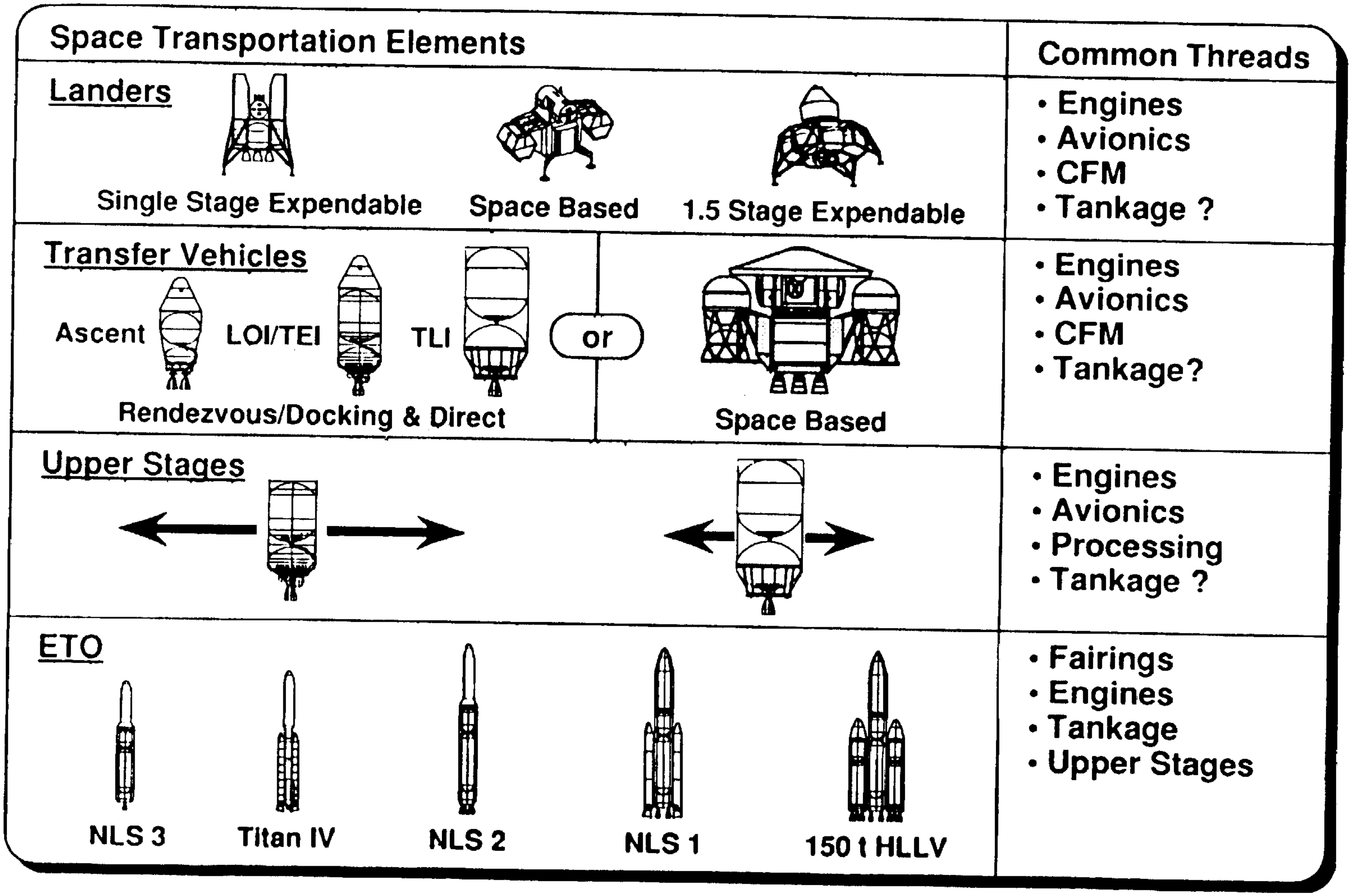
- What Should Be Done?
- What Shouldn't Be Done?

Key or Enabling Technologies

- What Areas Need to Be Developed?

Space Transportation Elements

MSFC



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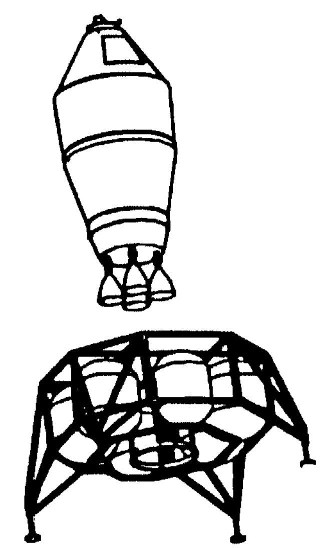
MSFC's EXPO Support Task

MSFC

Provide MSFC with Information and Analyses to Enhance Their Role in JSC's EXPO Studies (especially FLO)

- Parametrics
  - Mission Analysis
  - Programmatics
  - Design

- Vehicle Concepts
  - Upper Stages
  - Landers (became TD11)

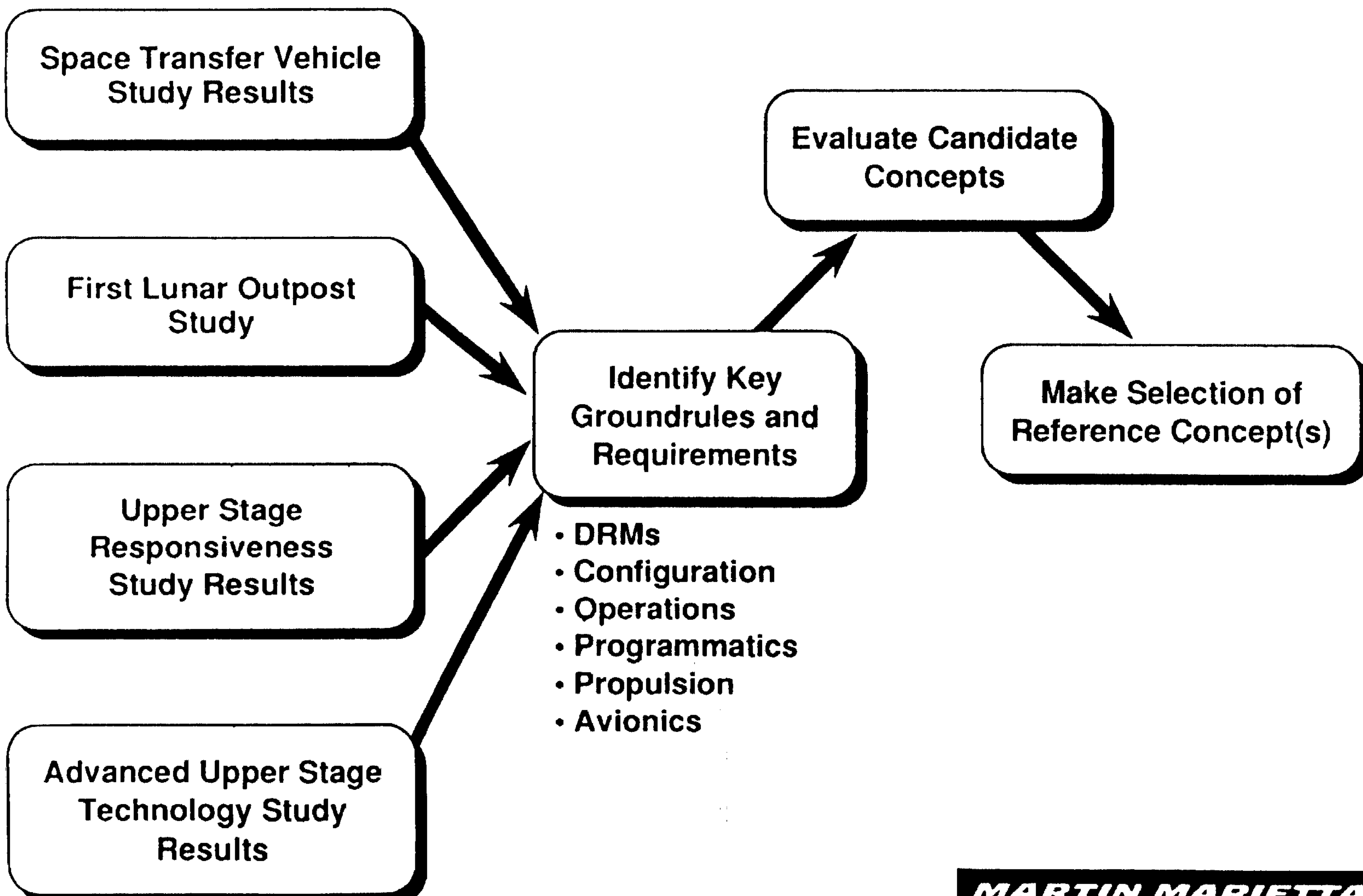


HLLV Upper Stage Analysis Task

Conduct Parametrics, Sensitivities, Analyses, and Trade Studies to Define the Characteristics of Upper Stages to Be Used in Conjunction with a Family of Heavy Lift Launch Vehicles

Mission Definition	Design Reference Missions
Configuration Analysis	Sizing, Performance, Commonality, Growth
Upper Stage Definition	Reference Configurations & Layouts
Launch Processing	Timelines, Facilities, Approach
Interfaces	System Level, Electrical and Mechanical
Avionics	Architecture, Adaptability, Automation
Programmatics	Cost and Schedule
Propulsion	Main and RCS Propulsion
Thermal	Insulation Requirements

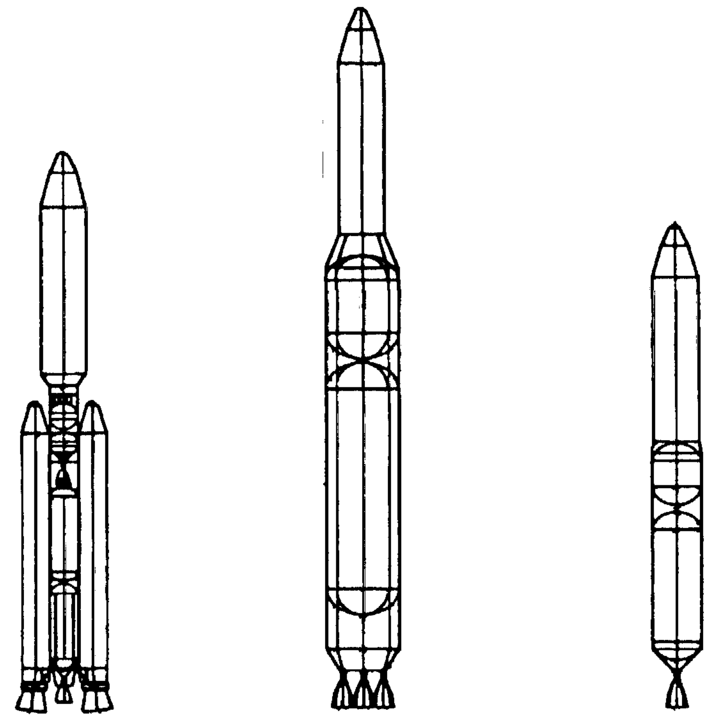
HLLV Upper Stage Study Approach



Key Groundrules & Assumptions

- Configuration
  - MSFC Provided the Four HLLV Options
  - The Upper Stage Is a Free Standing, Load Carrying Structure (for the HLLVs)
  - 20% Dry Mass Contingency
- Operations
  - Upper Stage Operations Make Maximum Use of Existing Facilities
  - Operations Utilize Automated Checkout with AGE and BIT
- Programmatics
  - The Upper Stage Uses Existing Hardware Where Applicable
  - All Required Technology Shall Be Flight Qualified by PDR
  - 1998 ILC, 1999 First Flight
- Propulsion
  - Liquid Oxygen and Liquid Hydrogen Are the Propellants (RL10A-4 & J-2S)
  - Single Engine Out Capability Exists in Multi-Engine Configurations
- Avionics
  - A Single Avionics Suite Shall Be Capable of Performing All DRMs
  - VHM Supports Mission Recovery from Planned Recoverable Fault States

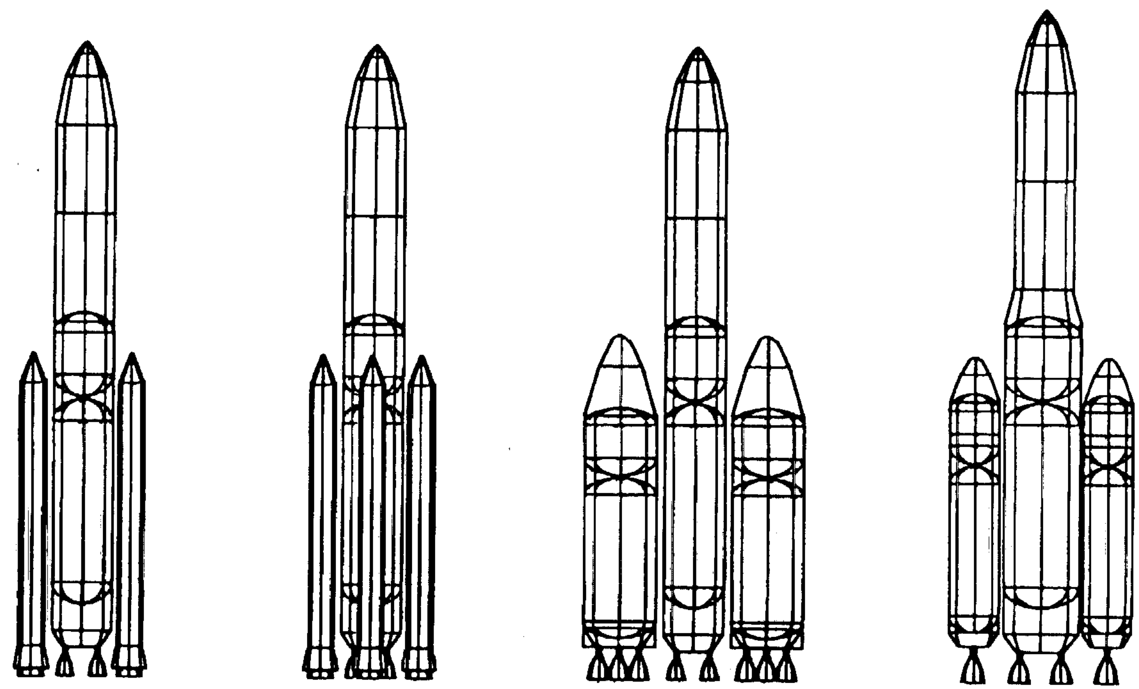
ETO Systems - *Titan IV and NLS 2 & 3*



Vehicle	Titan IV	NLS 2	NLS 3
Core	AZ 50/N2O4	"ET" LOX/LH2	New LOX/LH2
Diameter (m)	3.1	8.4	5.5
Booster	2x - SRMU's	N/A	N/A
Diameter (m)	3.3	N/A	N/A

ETO Systems - *Heavy Lift Launch Vehicles*

MSFC



Class	70 tonne	110 tonne	150 tonne	170 tonne
Core	"ET" LOX/LH2	"ET" LOX/LH2	"ET" LOX/LH2	New LOX/LH2
Diameter (m)	8.4	8.4	8.4	10.5
Booster	2x - ASRM's	4x - ASRM's	2x - New LOX/RP	2x-New LOX/RP
Diameter (m)	4.0	4.0	10.1	6.7

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Design Reference Missions

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Upper Stage Reference Missions - *Approach*

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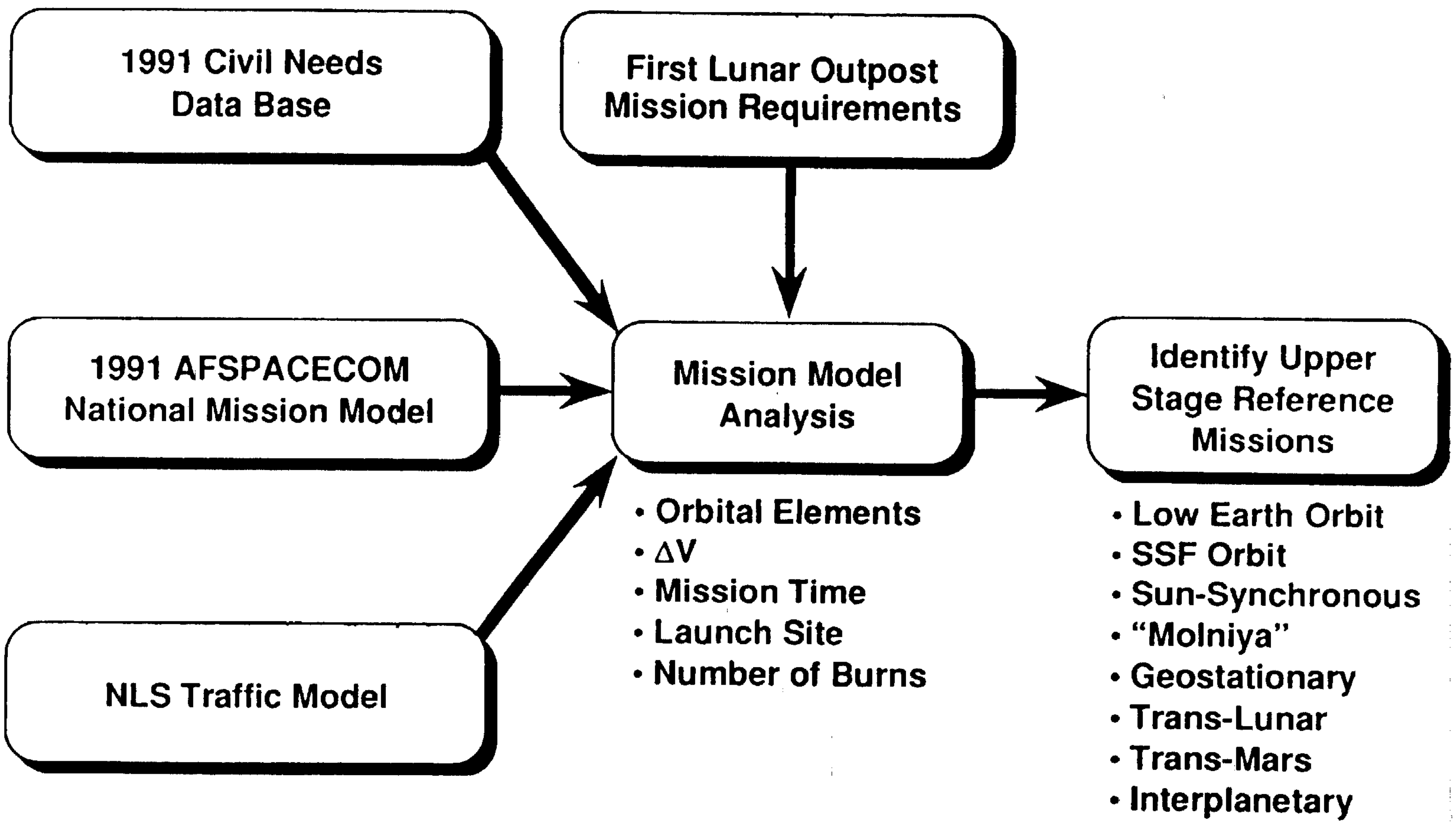

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Description of Reference Missions

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Reference Mission	Perigee Altitude (km)	Apogee Altitude (km)	Inclination (deg)	Typical Orbital ΔV (m/s)
1. Low Earth Orbit	150 - 350	150 - 350	28.5 - 57	70
2. SSF Orbit	407	407	28.5	200
3. Sun-Synchronous	500 - 900	500 - 900	97.4 - 99.2	5900 (ETR)
4. "Molniya"	180 - 900	39500 - 40200	63.4	2700
5. Geostationary	35790	35790	0 - 65	4300
6. Trans-Lunar	185 - 450	390000 - 525000	28.5 - 57	3200
7. Trans-Mars	185 - 450	C3 = 8 - 36	28.5 - 57	4200 (C3 = 22)
8. Interplanetary	185 - 450	C3 = 10 - 50	28.5 - 57	4500 (C3 = 30)

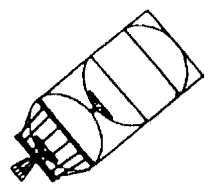
Upper Stage Performance Matrix

Vehicle	DRM 1 LEO (tonnes)	DRM 2 Space Station (tonnes)	DRM 3 Sun Sync (tonnes)	DRM 4 Molniya (tonnes)	DRM 5 GSO (tonnes)	DRM 6 TLI (tonnes)	DRM 7 TMI (tonnes)	DRM 8 Interplan-etary (tonnes)
HLLV1								
HLLV2								
HLLV3								
HLLV4								
NLS2								
NLS3								
TITAN IV								

"LEO" Missions

"High Energy" Missions

- Initial Sizing Based on "LEO" and "High Energy" Missions
- Engine Configurations Included:
  - 1 & 2 RL10A-4s for the Titan and NLS Upper Stages
  - 6 & 10 RL10A-4s for the HLLVs
  - 1 & 3 J-2S's for the HLLVs



Upper Stage Configurations

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Upper Stage Propulsion Options

MSFC

Engine Option	Type	Thrust (klbs)	Isp (sec)	Potential Upper Stage Size
RL10A-3	Cryo	16.5	444.4	S, M, L
RL10A-4		20.8	448 - 452	S, M, L
RL10B-2		22	456 - 468	S, M, L
J-2S		265	436	L
RS-44		16 - 20	~480	S, M, L
IME		20 - 200	465 - 475	S, M, L
NERVA Derived	Nuclear	25 - 75	870 - 925	L
Particle Bed		20 - 200	900 - 1000	S, M, L
Thermionic		0.2 - 1	~850	S, M
AJ10-118	Storable	9.6	319	S
OMS		6	320	S
XLR-132		3.7 - 15	340 - 347	S

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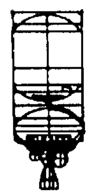
Upper Stage System Description

MSFC

Small

Characteristics			Payload Capabilities (t)		
			LEO	GSO	TLI
Total Mass	[illegible]	Titan IV (U)	[illegible]	[illegible]	[illegible]
Usable Propellant	[illegible]	NLS 2	[illegible]	[illegible]	[illegible]
Length	[illegible]	NLS 3	[illegible]	[illegible]	[illegible]
Max. Diameter	[illegible]				

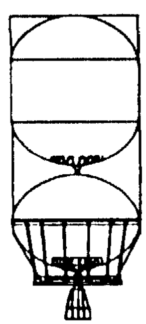


Medium

Characteristics			Payload Capabilities (t)		
			LEO	GSO	TLI
Total Mass	26.2 t *	Titan IV (U)	NA	6.3	9.3
Usable Propellant	22.7 t	NLS 2	36.9	8.7	13.3
Length	10.0 m	NLS 3 †	10.2	1.3	2.8
Max. Diameter	4.3 m				

* w/1 x RL10A-4

† w/2 x RL10A-4



Large

Characteristics (w/1 x J-2S)			Payload Capabilities (t)		
			LEO	GSO	TLI
Total Mass	123.2 - 168.9 t	70 t HLLV	116	26	40
Usable Propellant	110.0 - 150.0 t	110 t HLLV	147	37	56
Length	50.0 - 57.2 m	150 t HLLV	208	52	79
Max. Diameter	8.4 m	170 t HLLV	187	51	76

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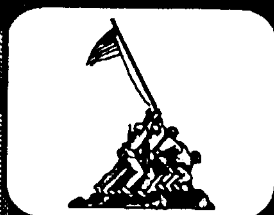
Upper Stage

Strategic Plan



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Task Objectives

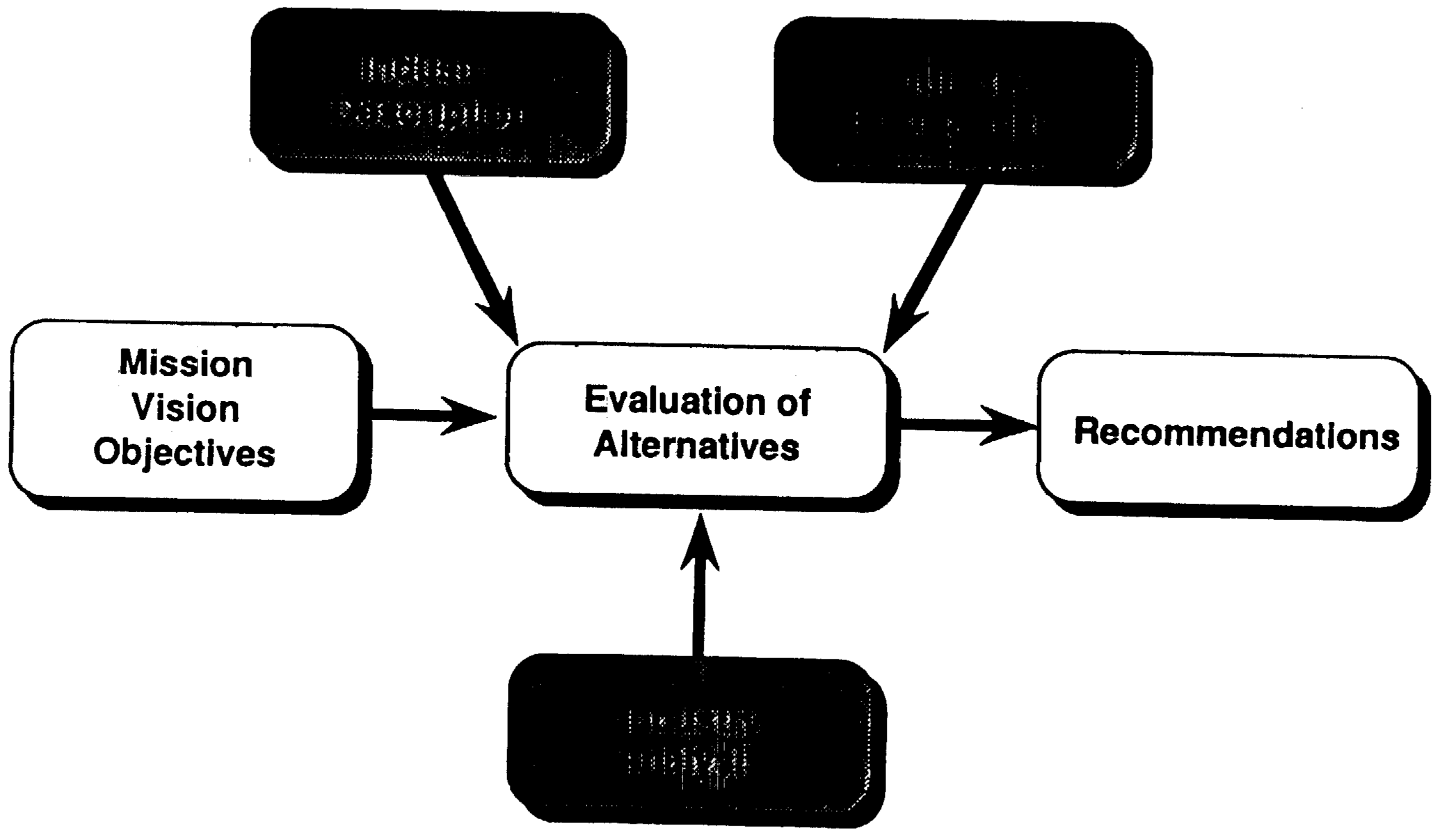
- Realize and Identify the Environmental Changes
  - Political
  - Societal
  - Business

- Develop a Strategy Responsive to These Changes

Get People to Think About the Problem from Another Perspective, Looking in Other Areas for Solutions

Upper Stage Strategic Plan - *Approach*

MSFC



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Upper Stage Development

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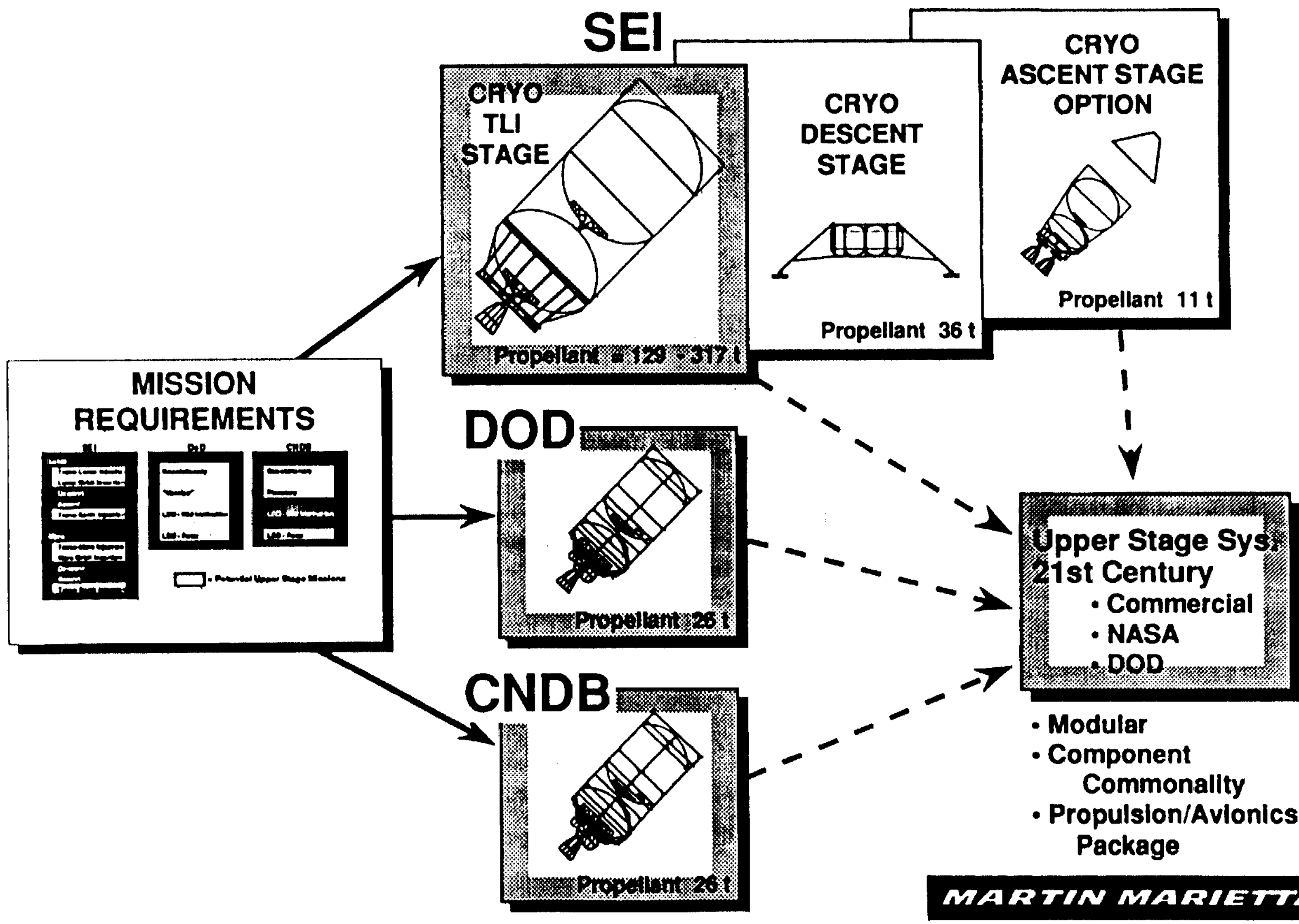

- Modular
- Component Commonality
- Propulsion/Avionics Package

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Strategic Goals

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Near Term (1993 and 1994)

- Perform a Comprehensive Integration of Government Agency Requirements
- Establish National and International Scenarios for Future Space Transportation
- Define Upper Stage Characteristics That Are Responsive to Changing Government Needs
- Evaluate the Application of Current Transportation Systems to Future Needs
- Define New Upper Stage Concepts to be Flown on the Next Generation of Launch Vehicles
- Incorporate Innovative Solutions to Development, Validation, and Procurement

Long Term

- Shape and Strengthen Our Technological Foundation to Maintain Our Leadership
- Further Develop the World's Spacefaring Capabilities
- Increase Our Competitiveness in the World Marketplace, Especially High-Tech
- Contribute to the Inspiration and Education of Society
- Keep Long Term Growth in Mind, Learn from Our Mistakes
- Do Not Force Unrealistic Schedule and Budget Goals

Government Organization Options

Option	Advantages	Disadvantages
Create an Alliance Between Current Agencies of the U.S. Government	• Eliminates Redundancy • Pooled Resources - Skills - Technology - Money • Shared Risk • Requires Commitment	• Conflicting Objectives • Loss of Total Control • Takes Time to Develop
Create a New Agency to Oversee the Development of the New Upper Stage(s)	• Bypasses Bureaucracy • Provides Proper Focus • Open to New Ideas • No Interest Conflicts • Provides Total Control	• Friction with Incumbents • Takes Time to Develop • New Reporting Level • Limited Resources • Risky to the Organization
Make One of the Existing U.S. Government Agencies Responsible for the New Upper Stage(s)	• Provides Proper Focus • Provides Total Control • Agencies Are In Place • Infrastructure Exists	• Friction with Other Players • Limited Resources • Risky to the Organization • May Not Support All Missions

Business Approach Options

MSFC

Option	Advantages	Disadvantages
Create an Alliance of Aerospace Contractors to Develop and Build the New Upper Stage(s)	• Pooled Resources • Shared Risk • Requires Commitment • U.S. Competitiveness • Spreads the Wealth	• Conflicting Objectives • Loss of Total Control • Takes Time to Develop • International Reprisal • Reluctance to Share Knowledge
Conduct the Upper Stage Program as a Fully Commercial Venture, with the Contractors Organized as They Wish	• Reduces Bureaucracy • Open to New Ideas • No Interest Conflicts • Provides Total Control • Reduces Cost • Accelerated Cycle	• Limited Resources • Risky to the Contractor(s) • Risky to the Government • Resistance to Change • Mission Model May Be Too Small
Use the Conventional Prime Contractor Approach, with a New Procurement Process	• Provides Some Control • Within Comfort Zone • Infrastructure Exists • Reduces Bureaucracy • Reduces Cost • Accelerated Cycle	• Risky to the Government • New Process Developed • Resistance to Change

Recommendations

Government Organization

Create an Coalition Between the Current U.S. Government Agencies Involved in Upper Stages and Streamline the Development Process

- Pool the Nation's Resources
- Cooperation and Commitment Are Essential
- Increase Effectiveness, Productivity, and Success
- Create a Source of Public Pride, Support Will Follow

Business Approach

Form an Alliance of Upper Stage Contractors to Develop and Manufacture the Upper Stage(s) in an Environment with Minimal Government Intervention

- Removes Burden of Risk from the Government
- Alliance Spreads Risk Among Participants
- Lower Cost to the Government and Increased Profitability to the Contractors

Summary

- The Completion of this Technical Directive Has Positioned Us to Start TD12 (Upper Stage Concepts & Rqmts) on a Dead Run
- The Time Has Been Taken to Learn from and Build on Previous Work
  - STV
  - USRS
  - AUSTS
- Understanding the Changing Environment Is Crucial, We Must Learn to Adapt Our Thinking

Technical Directive 10

Propulsion Avionics Module Study

PA Module Presentation Topics

MSFC

- P/A Module Overview
- Systems Engineering
- Configuration & Subsystem Details
  - Structures
  - Propulsion
  - Avionics
- Ground & Flight Operations
- Programmatics
- Summary & Conclusions

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Programmatics

P/A Module

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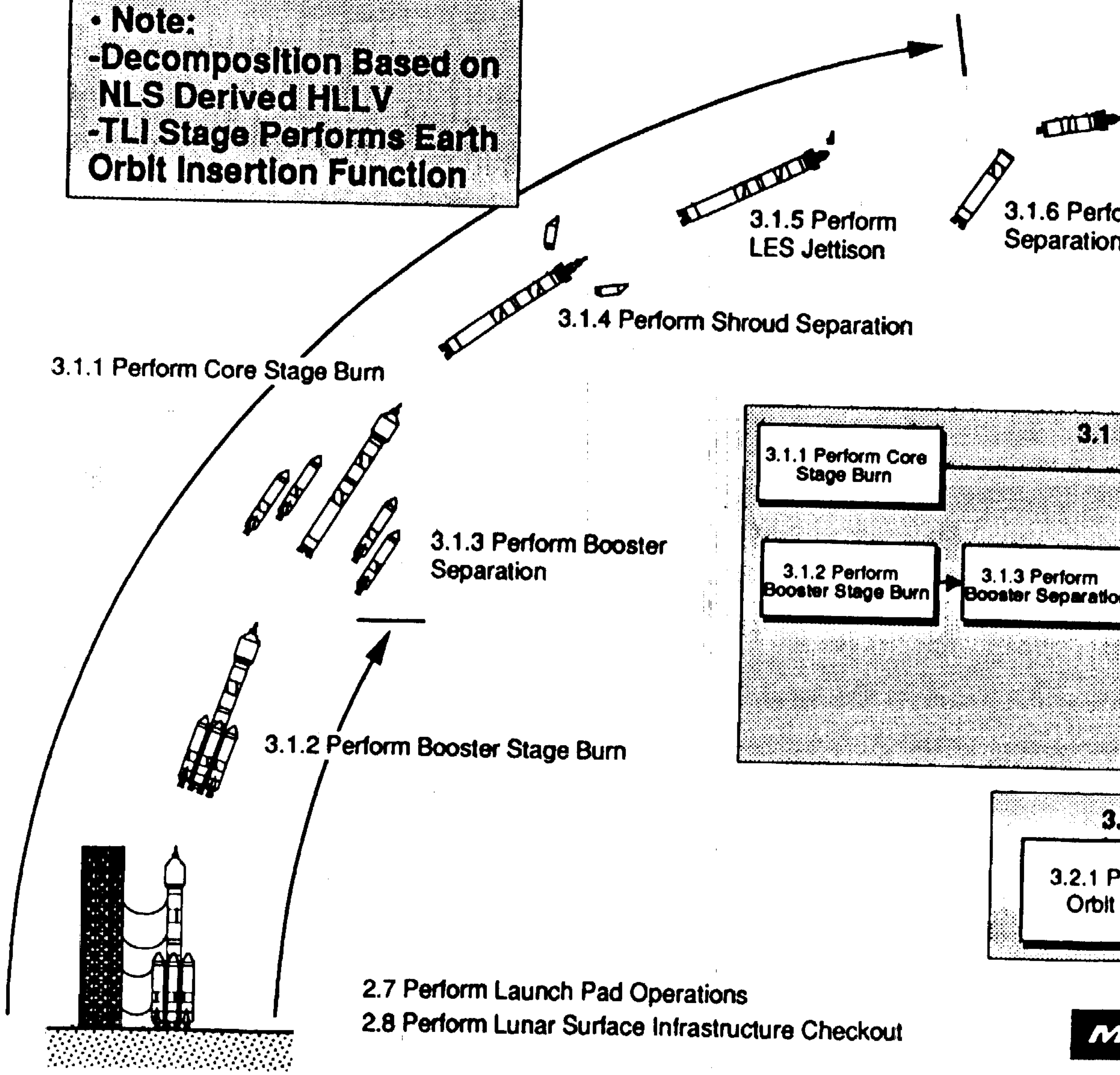
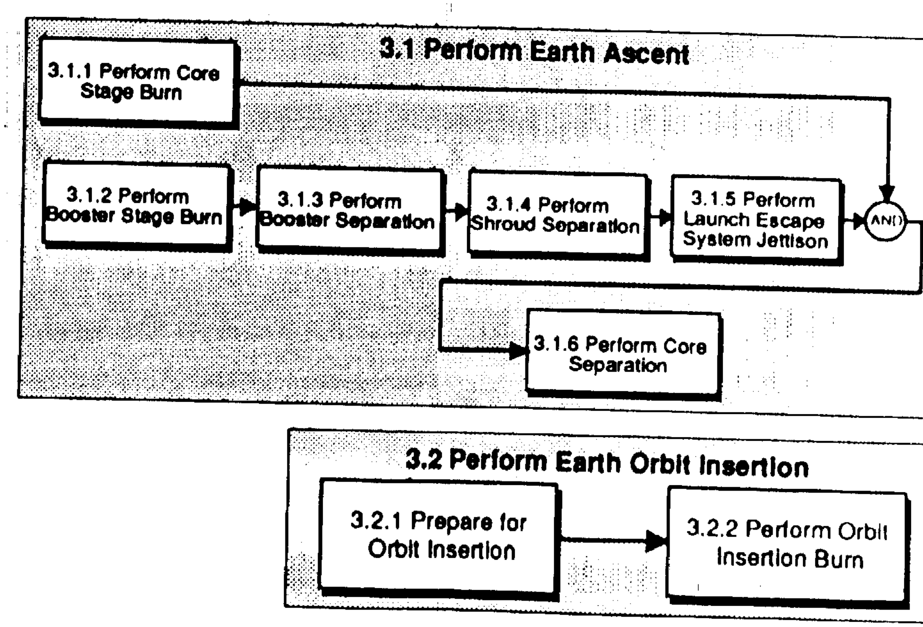
Lower Level Decomposition-NLS Derivative

MSFC

- Note:
  - Decomposition Based on NLS Derived HLLV
  - TLI Stage Performs Earth Orbit Insertion Function





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003

JCu920930-01A

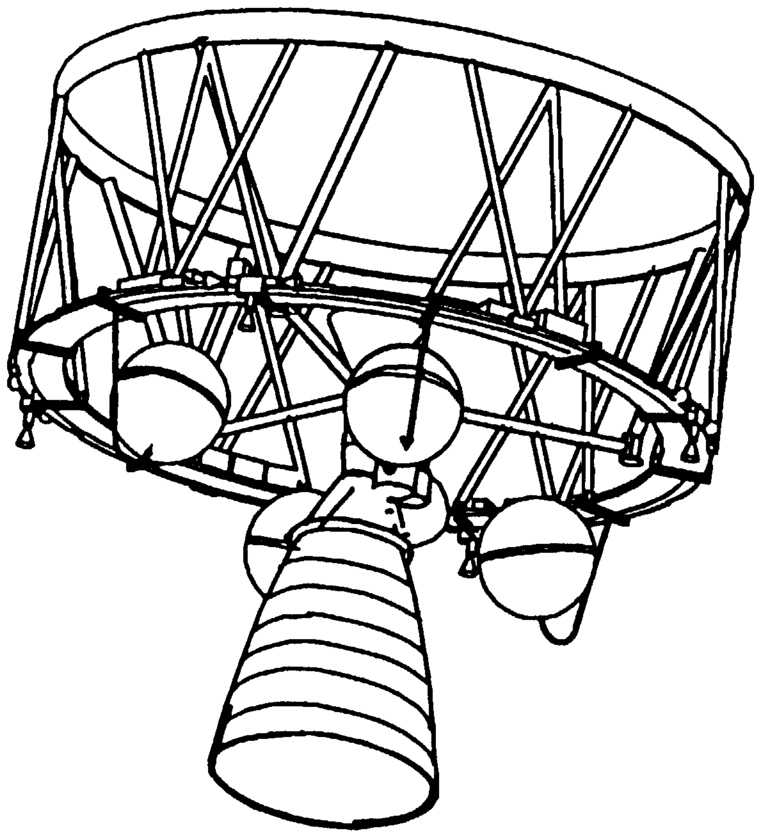
Reference Missions - P/A Module

MSFC

Class	Reference Mission	Perigee Altitude (km)	Apogee Altitude (km)	Inclination (deg)	Typical Orbital ΔV (m/s)
Ascent/Descent stage	1. Lunar Orbit Insertion	100 - 400	100 - 400	0 - 90	900 - 1100
	2. Lunar Descent	100 - 400	100 - 400	0 - 90	1850 - 2000
	3. Lunar Ascent	100 - 400	100 - 400	0 - 90	1900 - 2100
	4. Trans-Earth (Lunar)	100 - 400	100 - 400	0 - 90	900 - 1100
Impact Asses.	5. Mars Orbit Insertion	185 - 450	33840	0 - 90	10 - 20
	6. Mars Descent	185 - 450	33840	0 - 90	280
	7. Mars Ascent	185 - 450	33840	0 - 90	5400
	8. Trans-Earth (Mars)	185 - 450	C3 = 8 to 55	0 - 90	1000 - 4200
TLI/Upper Stage	9. Low Earth Orbit	150 - 350	150 - 350	28.5 - 57	70
	10. SSF Orbit	407	407	28.5	200
	11. Sun-Synchronous	500 - 900	500 - 900	97.4 - 99.2	5900 (ETR)
	12. "Molniya"	180 - 900	39500 - 40200	63.4	2700
	13. Geostationary	35790	35790	0 - 65	4300
	14. Trans-Lunar	185 - 450	390000 - 525000	28.5 - 57	3200
	15. Trans-Mars	185 - 450	C3 = 8 - 36	28.5 - 57	4200 (C3 = 22)
	16. Interplanetary	185 - 450	C3 = 10 - 50	28.5 - 57	4500 (C3 = 30)

P/A Module - Mass Properties Breakdown

MSFC



Mass Breakdown

Component	Mass (kg)
Primary Structure	484
Secondary Strct	676
Avionics	340
Contingency (20%)	300
Dry Mass	1,800
Engine & Mnt. Strct	3,488
RCS Prop	590
Total Stage	6,575

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RS920819-01A

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Systems Engineering

P/A Module

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JC920910-03A

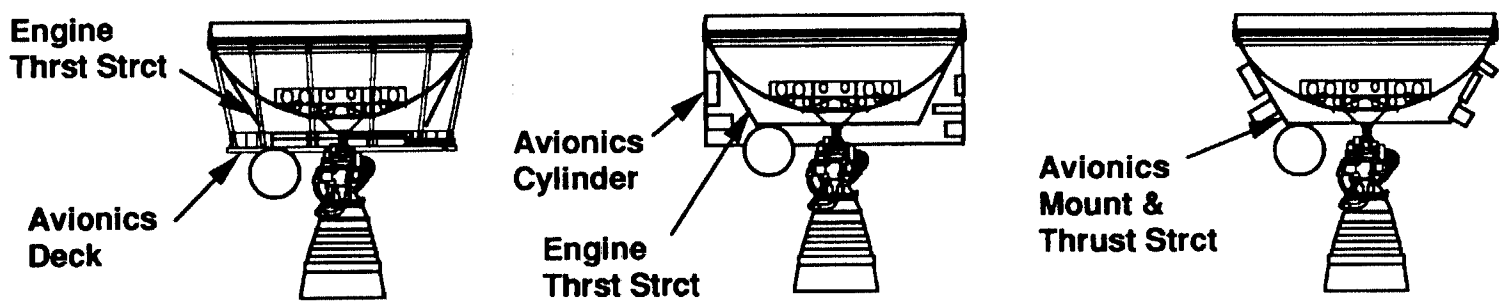
Key Mission Requirements & Groundrules

MSFC

- Configuration
  - MSFC Provided the Four HLLV Options
  - The PA Module Is a Free Standing, Load Carrying Structure
  - 20% Dry Mass Contingency
- Operations
  - *Flight H/W Received in Pre-Tested Configuration Ready for Final Processing*
  - *PA Module Processing Operations Make Maximum Use of Existing Facilities*
  - Operations Utilize Automated Checkout with AGE and BIT
- Programmatics
  - The PA Module Uses Existing Hardware Where Applicable
  - All Technology Will Be at a Technology Readiness Level of Six by PDR
  - 1998 ILC, 1999 First Flight
- Propulsion
  - Liquid Oxygen and Liquid Hydrogen Are the Propellants
  - Single Engine Out Capability Exists In Multi-Engine Configurations
- Avionics
  - *PA Module Avionics Shall Provide Functions for LV, Stage, and PA Module*
  - *A Modular Avionics Suite Shall Be Capable of Performing All DRMs*
  - VHM Supports Mission Recovery from Planned Recoverable Fault States

Upper Stages - P/A Module Candidate Config.

MSFC

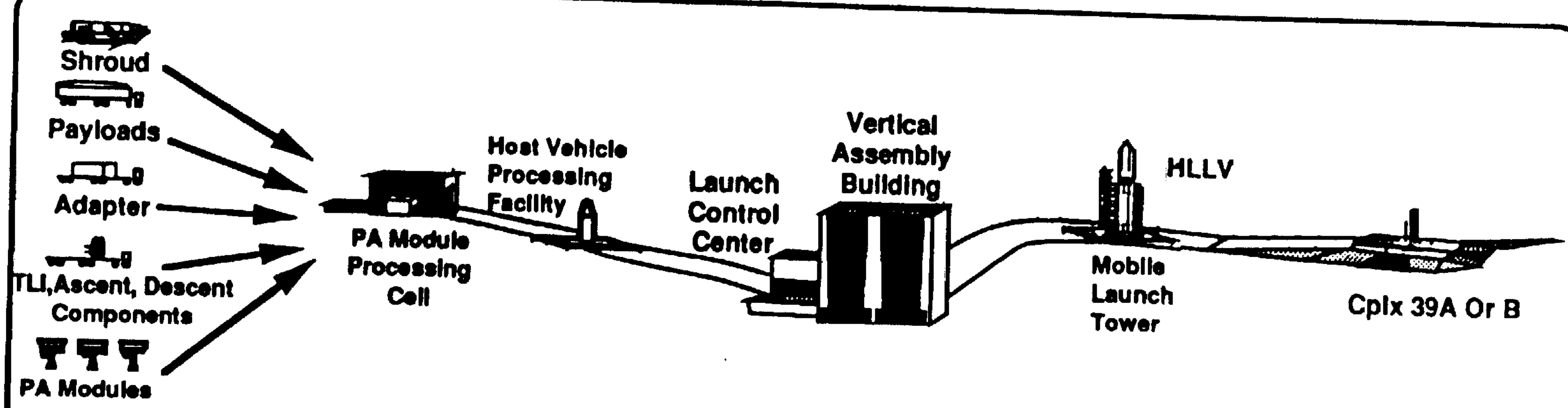


Option #1	Option #2	Option #3
Segmented Horizontal Avionics Deck	Segmented Vertical Cylinder Avionics Mount	Segmented Thrust Structure Avionics Mount

Requirements Impacts and Influences

MSFC

- Flight H/W Received in Pre-Tested Configuration Ready for Final Processing
- PA Module Processing Operations Make Maximum Use of Existing Facilities



* Host Vehicle Defined as TLI/Upper Stage, Ascent Vehicle, or Descent Vehicle

Propulsion Avionics Module Does NOT Require a Dedicated Facility.
PA Module Processing Done in Separate Cell of Host Vehicle Facility or as a Serial Sequence in the Overall Vehicle Processing

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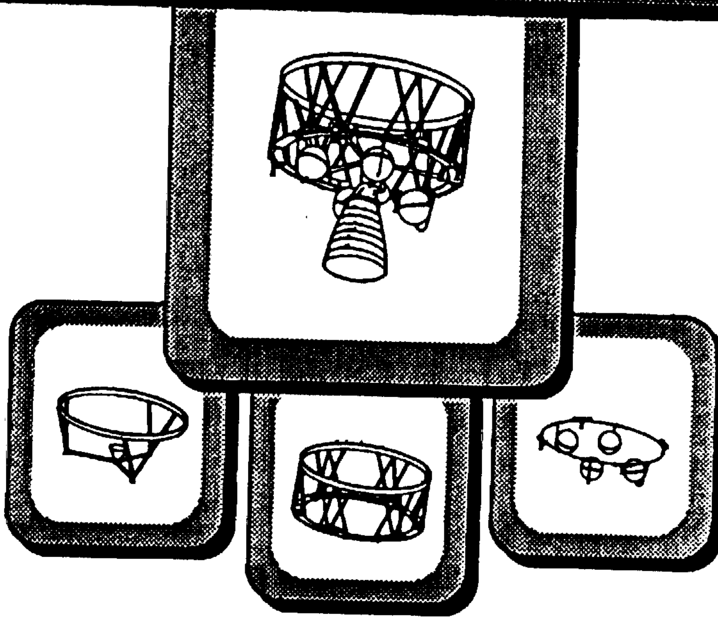
Reference Missions - P/A Module

MSFC

Class	Reference Mission	Perigee Altitude (km)	Apogee Altitude (km)	Inclination (deg)	Typical Orbital ΔV (m/s)
Ascent/Descent stage	1. Lunar Orbit Insertion	100 - 400	100 - 400	0 - 90	900 - 1100
	2. Lunar Descent	100 - 400	100 - 400	0 - 90	1850 - 2000
	3. Lunar Ascent	100 - 400	100 - 400	0 - 90	1900 - 2100
	4. Trans-Earth (Lunar)	100 - 400	100 - 400	0 - 90	900 - 1100
Impact Asses.	5. Mars Orbit Insertion	185 - 450	33840	0 - 90	10 - 20
	6. Mars Descent	185 - 450	33840	0 - 90	280
	7. Mars Ascent	185 - 450	33840	0 - 90	5400
	8. Trans-Earth (Mars)	185 - 450	C3 = 8 to 55	0 - 90	1000 - 4200
TLI/Upper Stage	9. Low Earth Orbit	150 - 350	150 - 350	28.5 - 57	70
	10. SSF Orbit	407	407	28.5	200
	11. Sun-Synchronous	500 - 900	500 - 900	97.4 - 99.2	5900 (ETR)
	12. "Molniya"	180 - 900	39500 - 40200	63.4	2700
	13. Geostationary	35790	35790	0 - 65	4300
	14. Trans-Lunar	185 - 450	390000 - 525000	28.5 - 57	3200
	15. Trans-Mars	185 - 450	C3 = 8 - 36	28.5 - 57	4200 (C3 = 22)
	16. Interplanetary	185 - 450	C3 = 10 - 50	28.5 - 57	4500 (C3 = 30)

Configuration & Subsystem Details



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P/A Module - Agenda 9/92

MSFC

- Methodology
  - Objectives & Approach
  - Point Of Departure Design
- Downselect Process
  - Candidate Configuration
  - Matrix Summary
- Configuration Details & Analysis
  - Baseline Configuration
  - Structural / Functional Groupings
  - Main Propulsion
  - Avionics
  - RCS
- Summary

Propulsion Avionics Module - Approach

MSFC

Objectives: 1.) To Develop, Through Innovative Design & Comparative Analysis, The Configuration Options For a Modular Propulsion Avionics System That Satisfies The Largest Number of Applications Associated With an Upper Stage & Lunar/Mars Missions

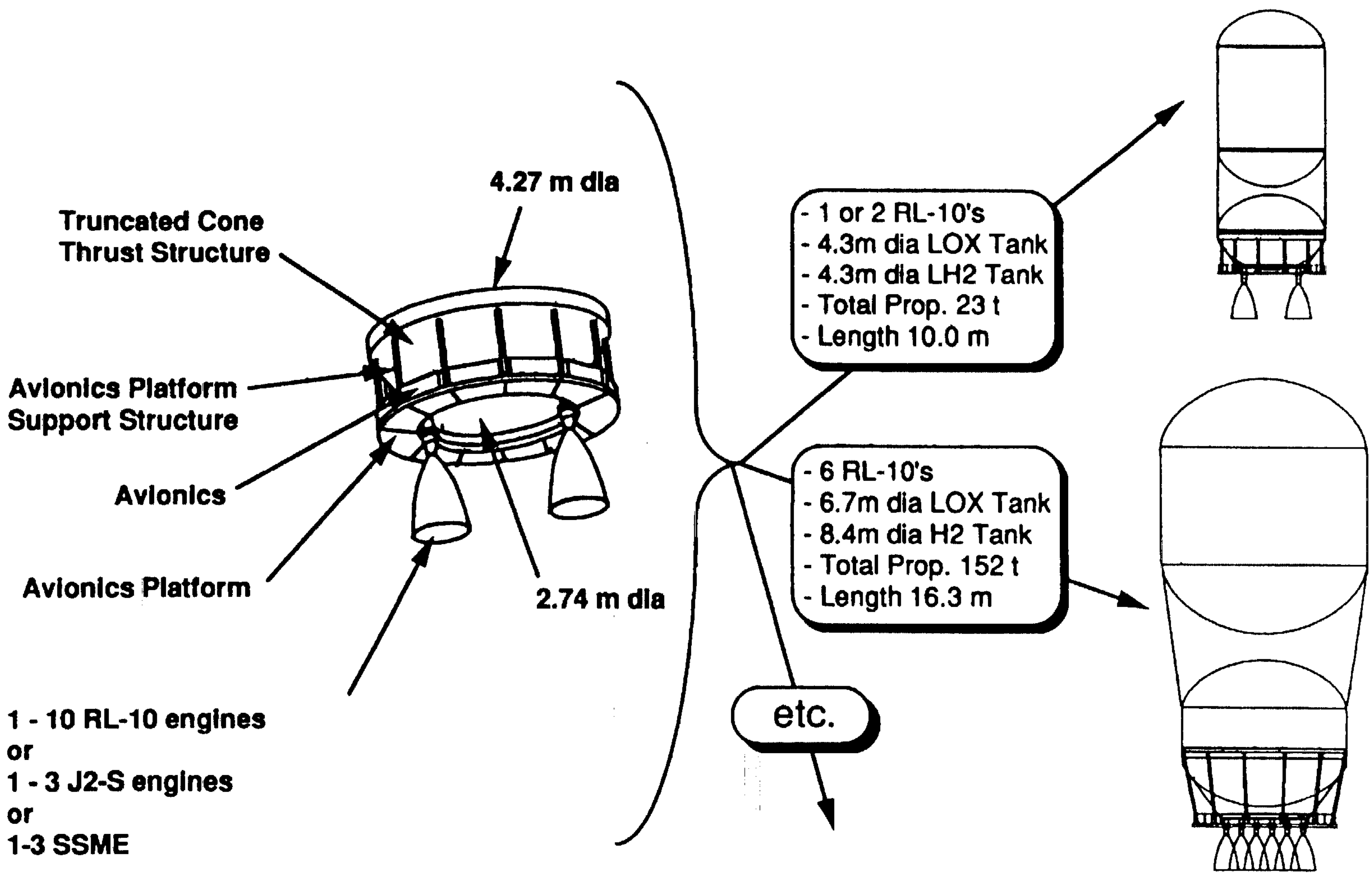
2.) To Perform All Analyses And Designs With Growth, Evolution And Adaptability as Underlying Thrusts While Maximizing Commonality Across Configurations, As Opposed To A Single Point Design With Difficult Growth At Best

Approach:

1. Identification of Key Drivers For Configuration Design
2. Preliminary Configurations Analysis Based on Selected DRM's
3. P/A Module Configuration Trades Based on Adaptability and Functionality
4. P/A Module Configuration Sizing Utilizing Trade Results & ETO Constraints
5. Configuration Selection From Detailed Analysis & Definition

US - P/A Module Point Of Departure

MSFC



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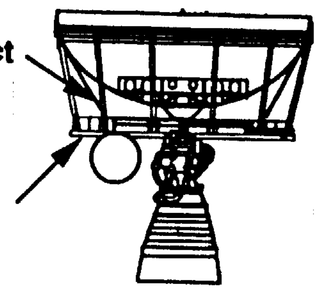
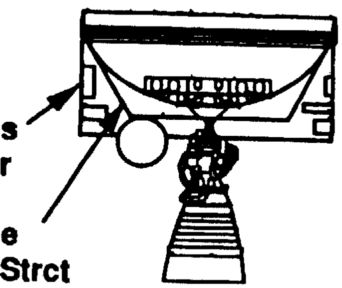
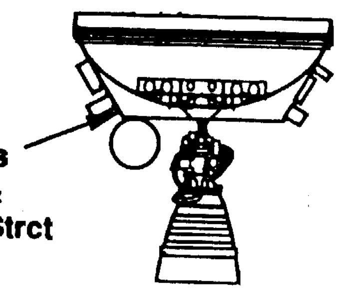
Upper Stages - P/A Module Candidate Config.

MSFC







Option #1
Segmented
Horizontal Avionics
Deck

Option #2
Segmented Vertical
Cylinder Avionics
Mount

Option #3
Segmented Thrust
Structure Avionics
Mount

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RS920423-04A

P/A Module - Down Select Matrix

MSFC

Issues	Horizontal Deck	Vertical Mount	Thrust Strct. Mnt
Thrust Strct	Same	Same	Unacceptable Acoustic Environment For Avionics
Mount Strct	5% Heavier Than Vertical Mount	Lighter Weight	
Support Strct	5% Lighter Than Vertical Mount	Longer Attach Point For Horizontal Struts	
Thermal Blnkt	13 % Heavier Than Vertical Mount	Cylinder Section Reduces Blanket Length	
Avionics	6% Heavier Due to Heater Batteries	Cylinder Mnt Requires No Add. Heaters	
RCS System	Same	Same	
Manufac.	[redacted]	Increased Complex. - Mnting Strct to Avionics	
Ground Proc.	Same	Same	
Check-Out	Same	Same	
Maintenance	[redacted]	Cylinder Disassembly Required.	

[redacted]

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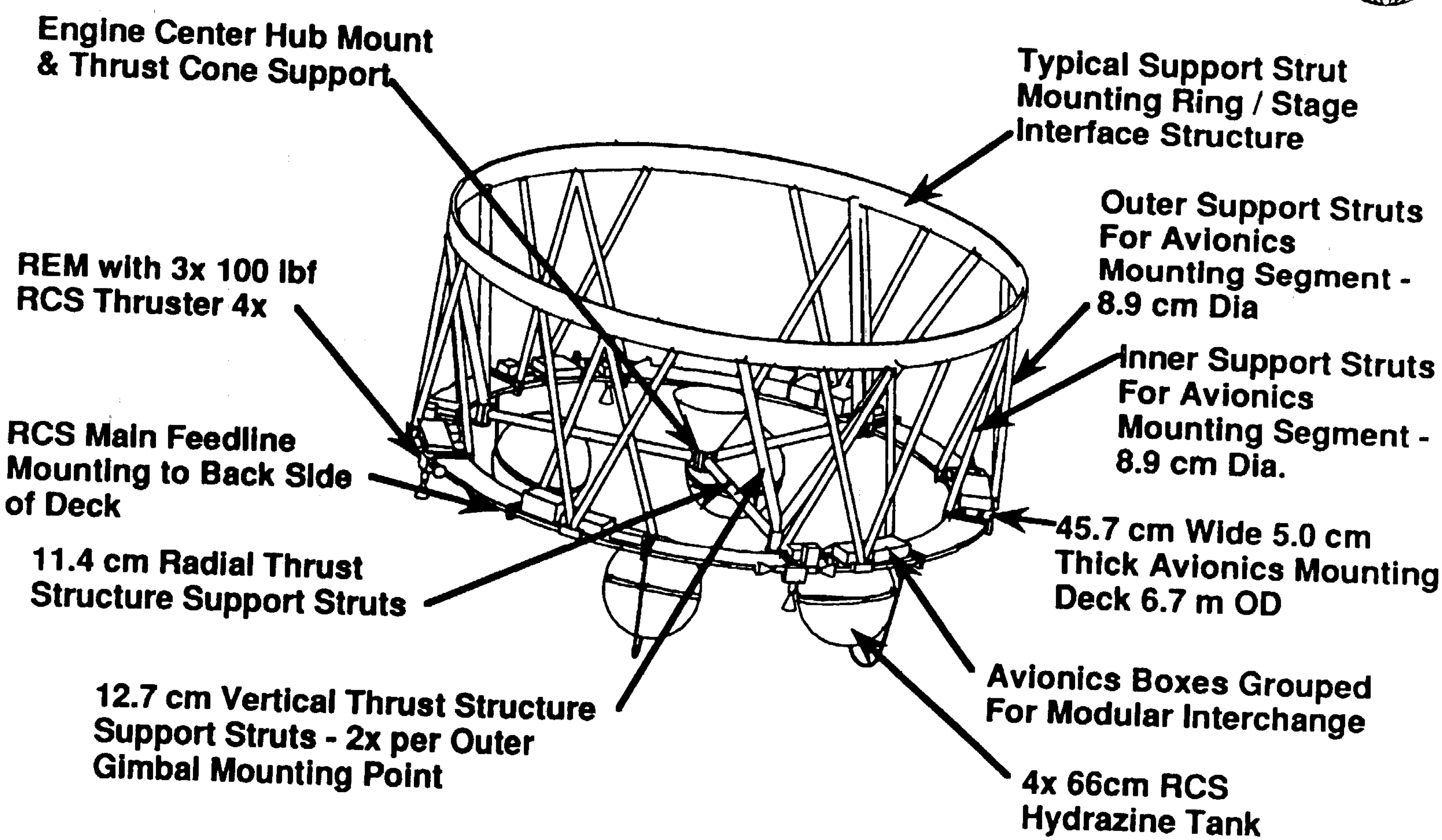
RS920604-01A

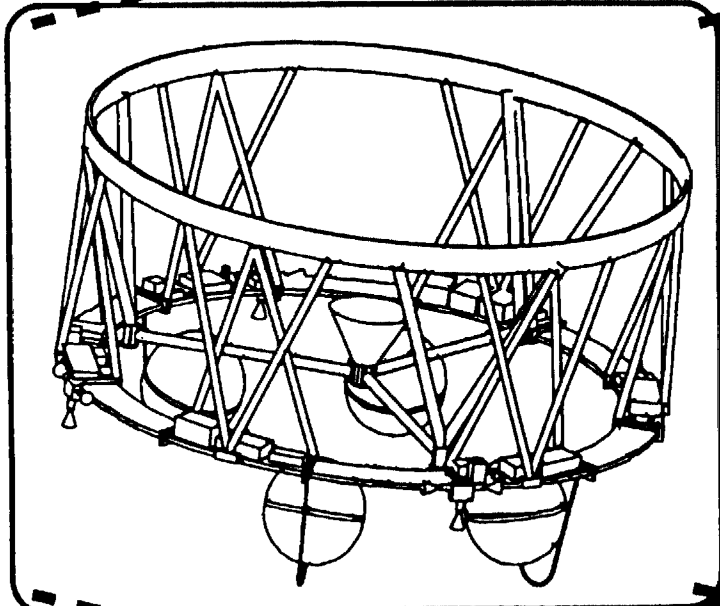
P/A Module - Baseline Configuration

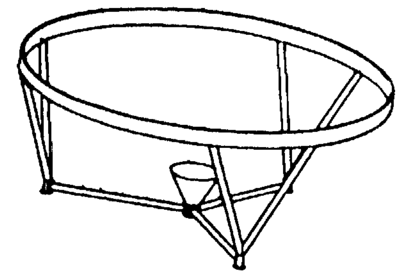
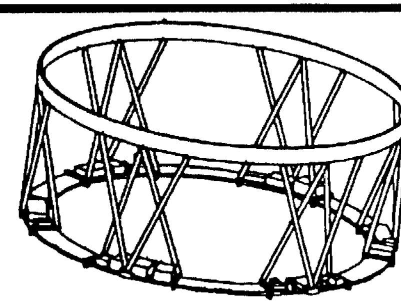
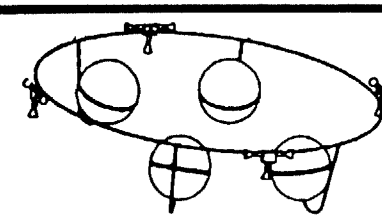
MSFC



P/A Module - Main Structural Grouping

MSFC



Description	
Radial Strut 11.4 Dia* Vert. Strut 12.7 Dia* Cntr Thrust Cone Cntr Hub 650,000 lbf Outer Pnt 265,000 lbf	
Basic Elem. Cnt = 5	Engine Thrust Structure
Inner Strut 8.9 Dia* Outer Strut 8.9 Dia* Avionic Mount Sgmt Ring Spacer Sgmt Grouped Avionics RCS Mounts	
Basic Elem. Cnt. = 6	Avionics Mount
Hydrazine Tanks Main Feed 1.3 Dia* Rocket Eng. Mod 4x 20 lbf -or- 100 lbf	
Basic Elem. Cnt. = 4	Reaction Control System

* All Units in "cm" Unless Otherwise Noted

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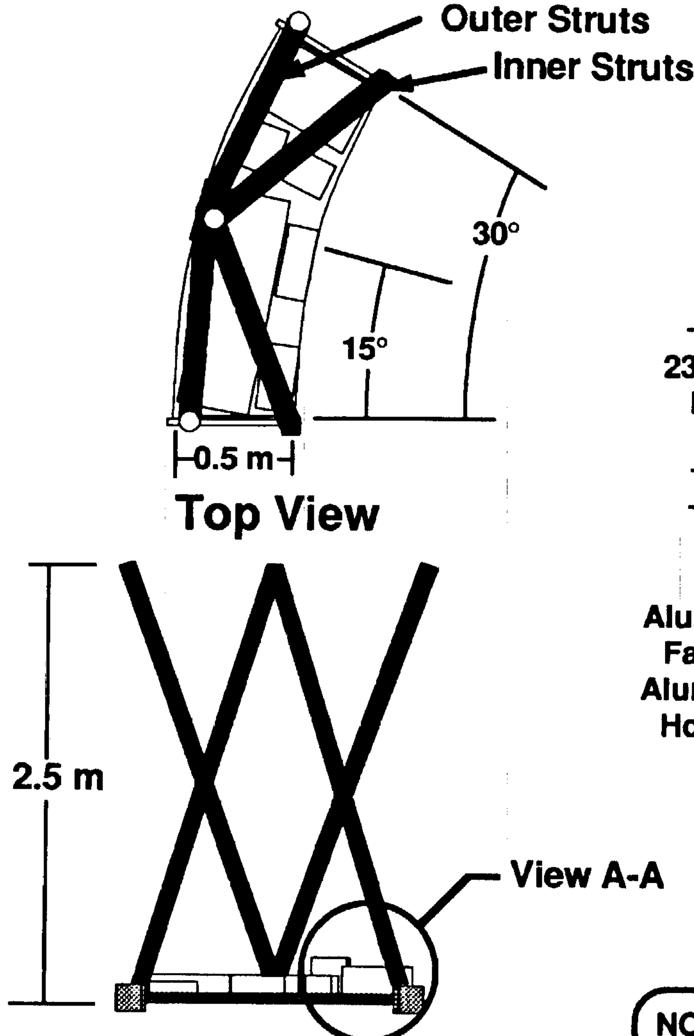
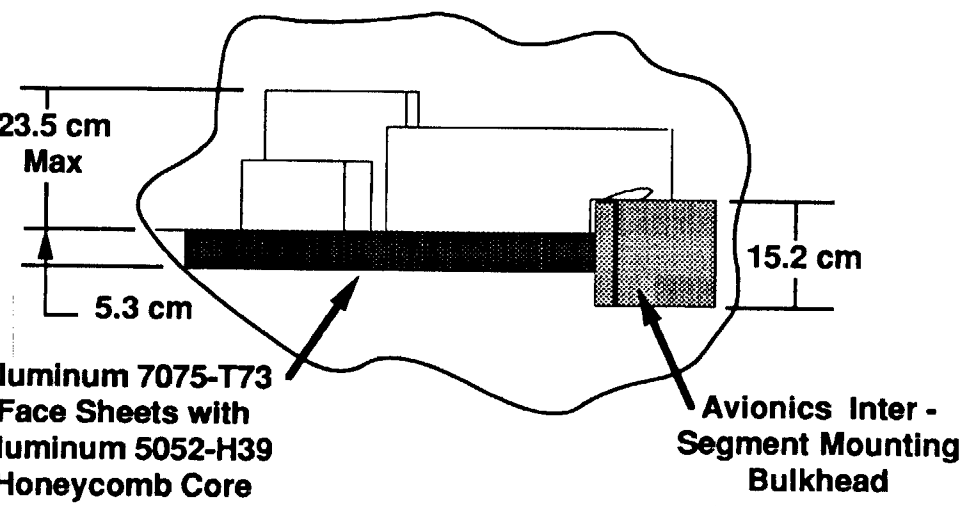
RS920818-01A

Avionics Deck Config. (Segment #1) - cont.



Top View

Front View
(Rotated)

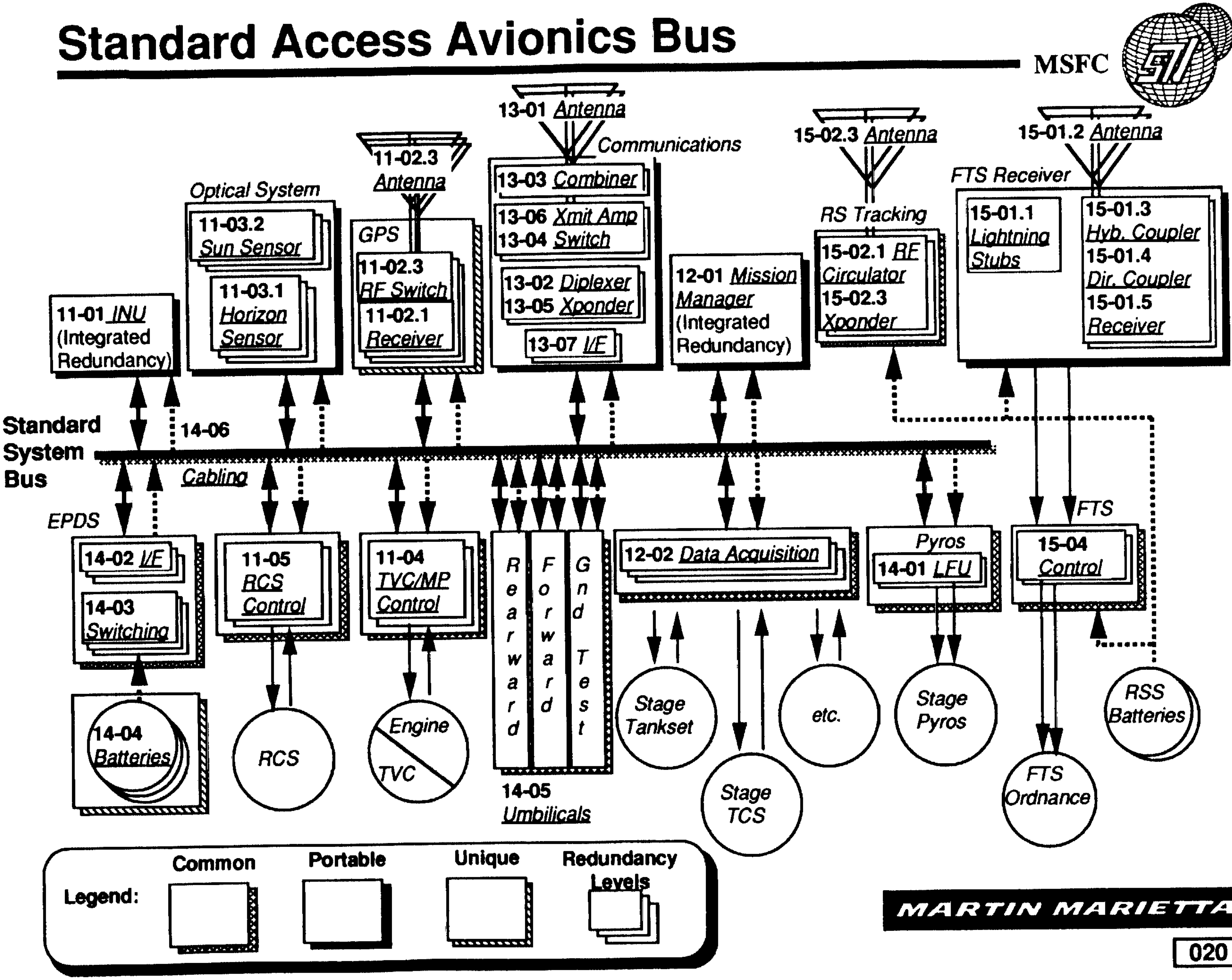


View A-A
(Struts Removed For Clarity)

NOTE: All Struts are Attached with Mono Ball End Fittings

Standard Access Avionics Bus

MSFC



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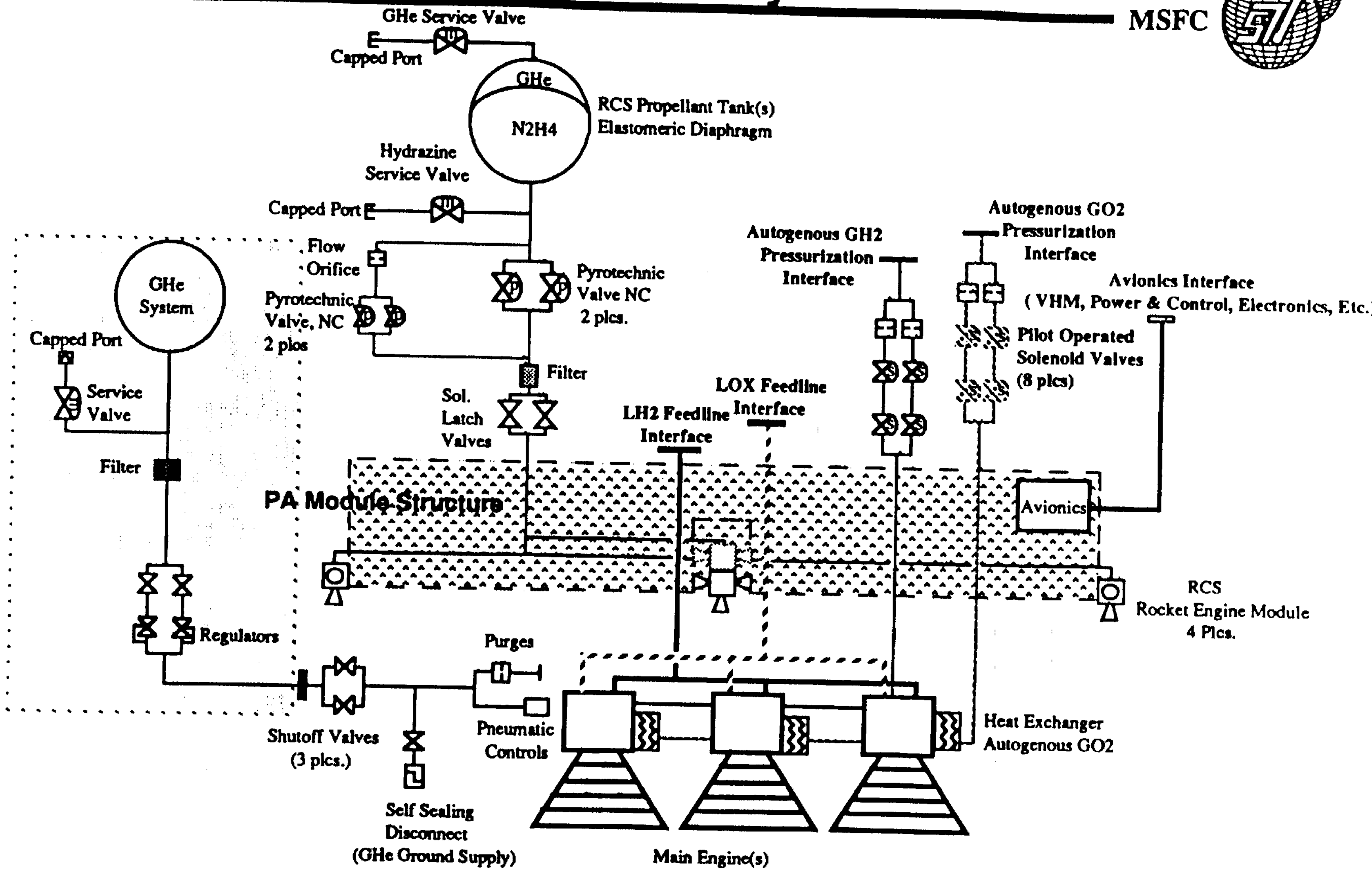
020

RM920818-01

PA Module Propulsion Subsystem

MSFC



- GHe Bootstrap Prepressurization Prior to Engine Start is not Carried as Part of the PA Module

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021

PP920609-01

Main Engine Operational Characteristics

MSFC

Candidate Engines	Engine Type	Inlet Pressure Requirement	Open Loop MR Dispersion	Propellant Utilization Capability	Start Transient	Shutdown Transient	Specific Isp, Vacuum Thrust	Thrust Vector Control	Autogenous GH2 Capability	Autogenous GO2 Capability	Gimbal Capability	Pneumatic Press. Requirement	Expansion Ratio Chamber Press.
RL10	Expander Cycle	30 psia Fuel 45 psia Oxid	2% @ 5.5 MR	Yes	2 sec	1 sec	450 sec, 20.8 Klbf	Hydr.	.07 lb/sec	No Current Capability	6 °	470 psia	84:1 565 psia
J-2S	Gas Generator	30 psia Fuel 39 psia Oxid	1 % @ 5.5 MR	Yes	5 sec	3 sec	436 sec, 265 Klbf	Hydr.	1 lb/sec	1 - 3 lb/sec	7.5°	485 psia	40:1 1200 psia
SSME	Staged Combustion	32 psia Fuel 45 psia Oxid	1% @ 6.0 MR	Yes	4.2 sec	2.5 sec	453.5 sec, 470 Klbf	Hydr.	TBD	TBD	10°	875 psia	77.5:1 3006 psia

- STME Characteristics not Included Due to Lack of Available Data
- Although the SSME is Relatively more Complicated, Engine Operational Characteristics does not Impose Greater Complexity on the Design of the Propulsion System

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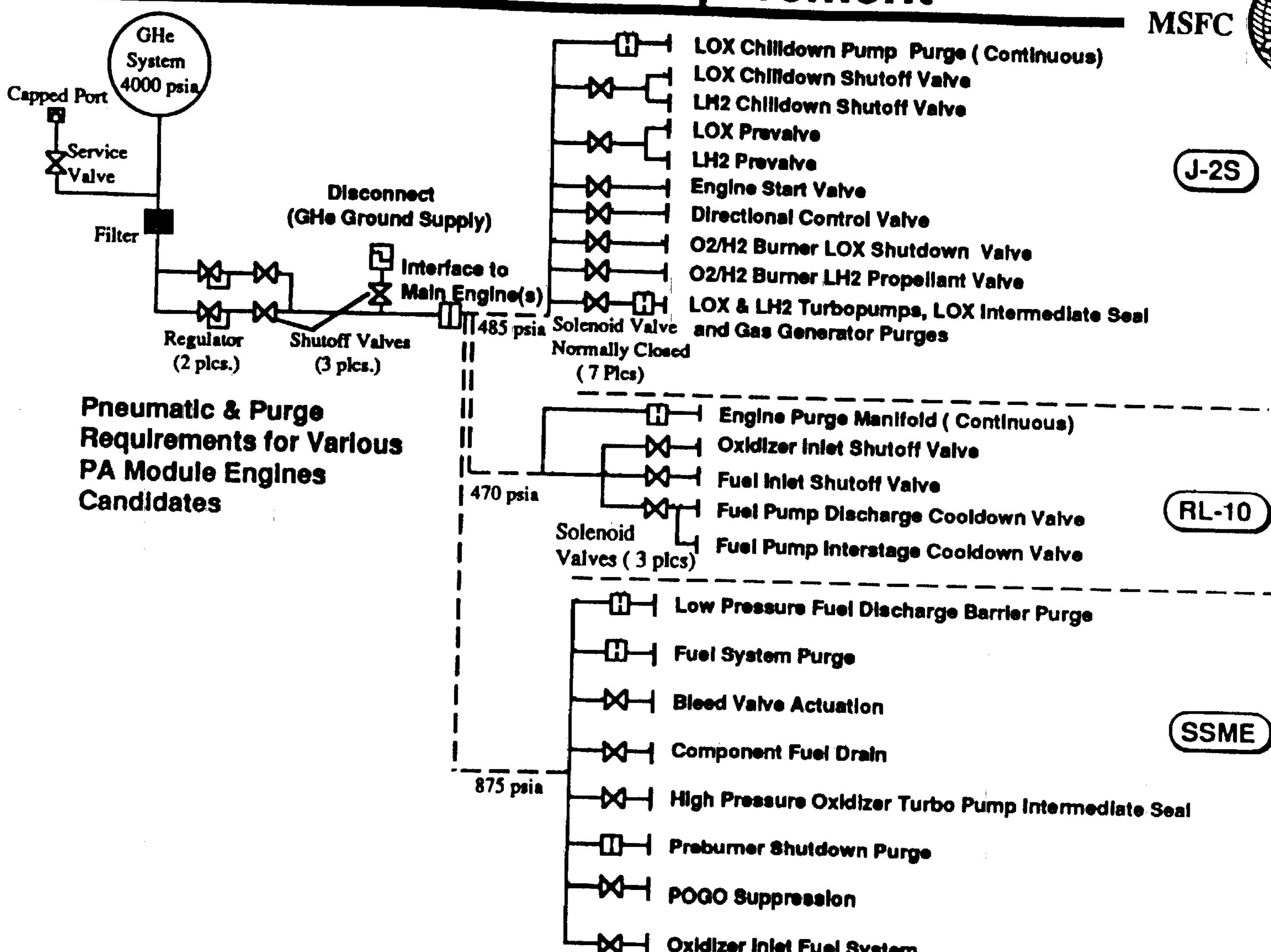
PA Module Engine GHe Requirement

MSFC



Pneumatic & Purge Requirements for Various PA Module Engines Candidates

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PP920609-01

Main Engine Operations

MSFC

- Childown Operations
  - Required to Achieve Proper Temperature Level for Starting Condition
  - Childown Typically Performed during On-Orbit Coast Except for Suborbital Burn where Childown is Performed during Ascent to Minimize Gravity Loss
  - Propellant Usage Requirements for Childown Varies with each Engine Candidates
  - Expended Propellants can be Dumped Overboard During Childown Operations (SSME is an Exception since it May Require Use of Propellant Recirculation Line)

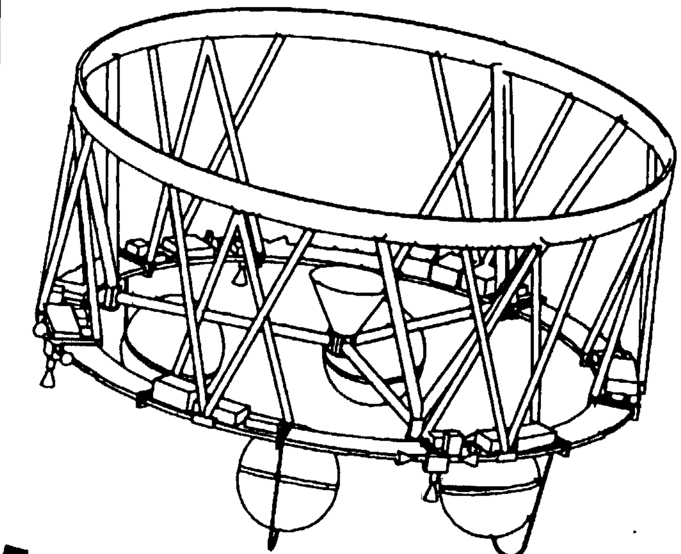
- Tank Pressurization
  - Tanks Prepressurized with GHe Supplied by the Ground System Just Prior to Launch
  - GHe Bootstrap Pressurization Required for all Main Engine Candidates and is Provided by the On Board Vehicle GHe System
  - Pressurization During Engine Burn Performed Autogenously for both LO2 and LH2 Tanks

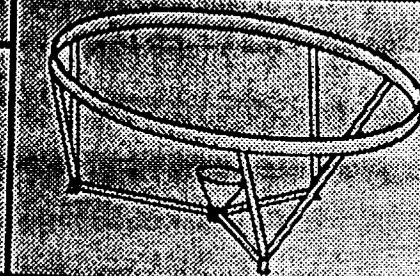
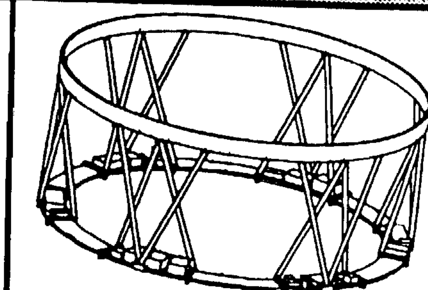
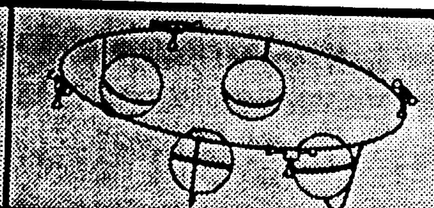
- Start-up/Shutdown
  - Modifications Required on the SSME for On-Orbit Start/Restart Capability
  - No Significant Impact on Start-up Transients for Multi Engine Configuration
  - Main Engines Specifies Fuel Rich Shutdown to Avoid Potential Damage
  - All Engine Candidates Have Demonstrated Benign Shutdown Making them Suitable for Restart

- Restart
  - Use RCS Thrusters to Settle Propellant
  - Vehicle Health Management to Assess Engine Condition Prior to Restart

P/A Module - Main Structural Grouping

MSFC



Description*	
Radial Strut 11.4 Dia* Vert. Strut 12.7 Dia* Cntr Thrust Cone Cntr Hub 650,000 lbf Outer Pnt 265,000 lbf	
Basic Elem. Cnt = 5	Engine Thrust Structure
Inner Strut 8.9 Dia* Outer Strut 8.9 Dia* Avionic Mount Sgmt Ring Spacer Sgmt Grouped Avionics RCS Mounts	
Basic Elem. Cnt. = 6	Avionics Mount
Hydrazine Tanks Main Feed 1.3 Dia* Rocket Eng. Mod 4x 20 lbf -or- 100 lbf	
Basic Elem. Cnt. = 4	Reaction Control System

* All Units in "cm" Unless Otherwise Noted

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RS920818 01A

P/A Module Avionics Design Philosophy

MSFC

- In The Conceptual Design Phase It Is Appropriate To Address Innovative Approaches
- Configuration Control Planning Can Be Programmed In the Conceptual Design Phase
  - Support All Currently Identified Upper Stage DRMs with a Common Configuration
  - Identify Potential Axes of Variation in the Configuration and Emphasize Modularity in These Areas
  - Work for Compatibility with Commercial Standards, Components, and Systems with the Idea of Streamlining Adaptation to Availability Changes, New Technology
- By Taking Advantage of the Inherent Isolation Characteristics of a Fiberoptic Information Bus, the Rationale for Physical Separation of Flight-Critical and Non-Flight Critical Busses Is Weakened, and May No Longer Apply
  - Integration and Checkout Costs Are Reduced by Minimizing Interconnection Wiring Complexity
  - The Single Bus Allows a More Simple and Consistent Approach to Redundancy Management, which Is Compatible with the Application of Flight-Time VHM

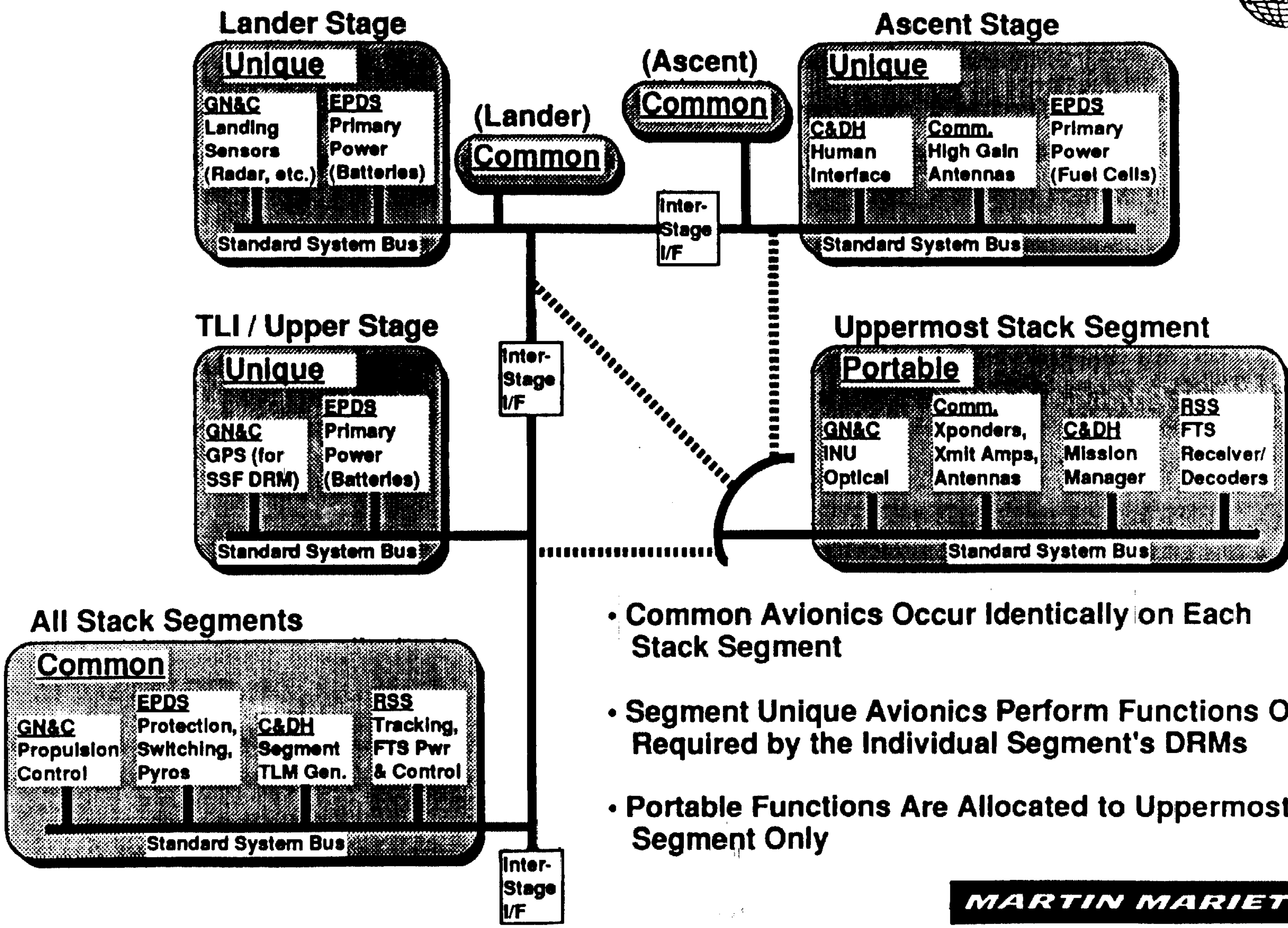
Avionics Modularity, Portability and Evolution

MSFC

- Space Transportation Avionics Designs Are Driven Primarily by Reliability and Functional Coverage, Secondarily by Costs and Weight
- Traditionally, the Approach to Space Transportation Avionics Has Been To Provide a "Lean" System, To Minimize Design and Test Costs, and To Create System Determinism and High Reliability through Simplicity
- Distributed Systems Technology and the Increasing Incidence of Embedded Systems In Many Areas of Hardware Are Changing the Nature of "Lean" and "Simple" Systems
- Modularity, Function Portability, and the Ability to Adapt to Changing Requirements in an Evolutionary Manner and Minimize the System Impacts of Change Are Outgrowths of the Acceptance that Rapid Technological Change and Increasing Rates of Obsolescence Are Facts of Life in Electronics and Most Other Industries
- Integration, Test and Configuration Costs Are Reduced, Adaptation of New Equipment into the System Requires Less Redesign Through the Use of Standard I/F's, Protocols
- Allows for Controlled Configurations while Maintaining Flexibility
- Supports Incremental Reuse and Qualification Concepts, by Providing Framework for Adaptation
- Portability for GN&C, Communications for Upper Stages Is Driven by Stage Disposal Strategies

Common, Portable, and Unique Avionics

MSFC



- Common Avionics Occur Identically on Each Stack Segment
- Segment Unique Avionics Perform Functions Only Required by the Individual Segment's DRMs
- Portable Functions Are Allocated to Uppermost Segment Only

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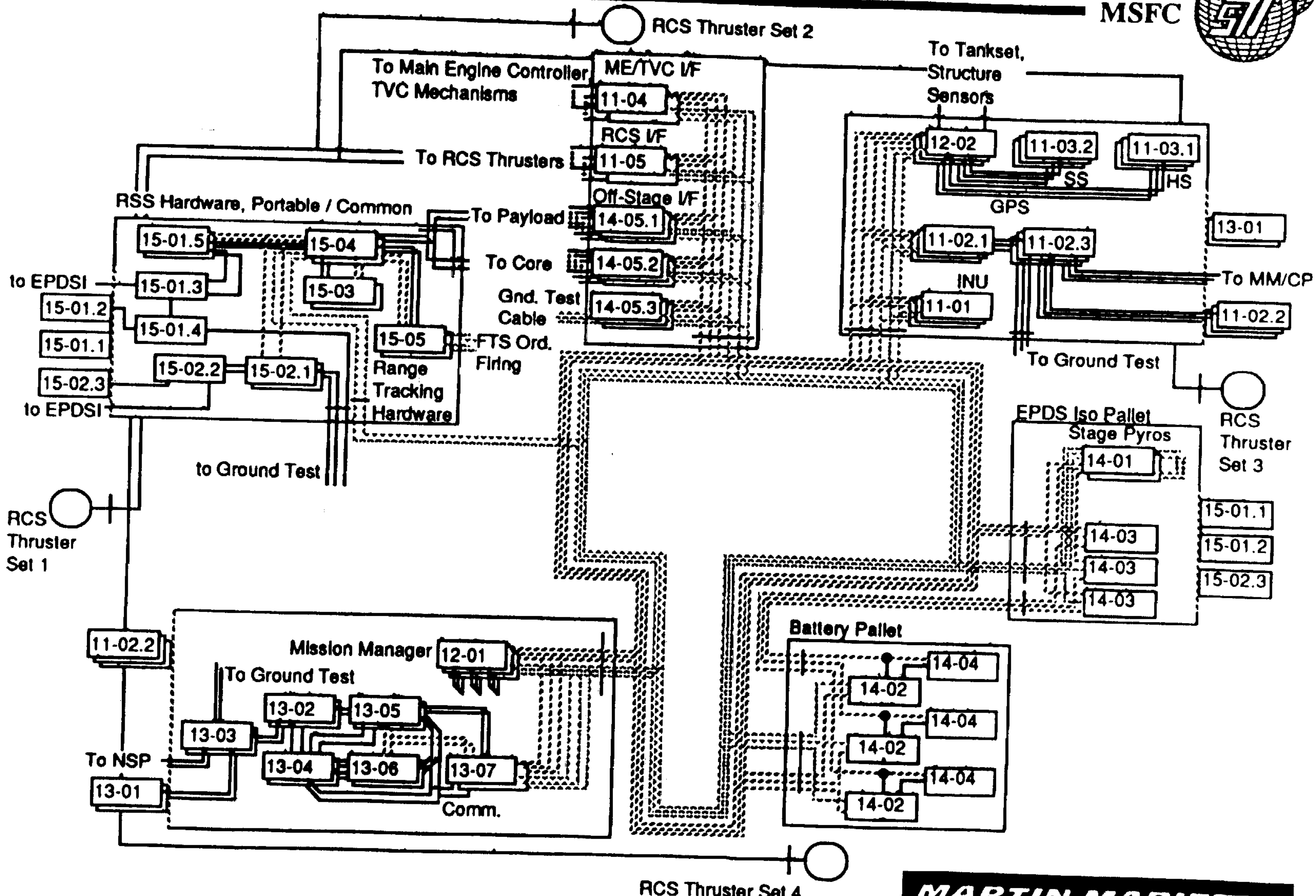
RM920611-01

Avionics Cabling/Bus Design

MSFC



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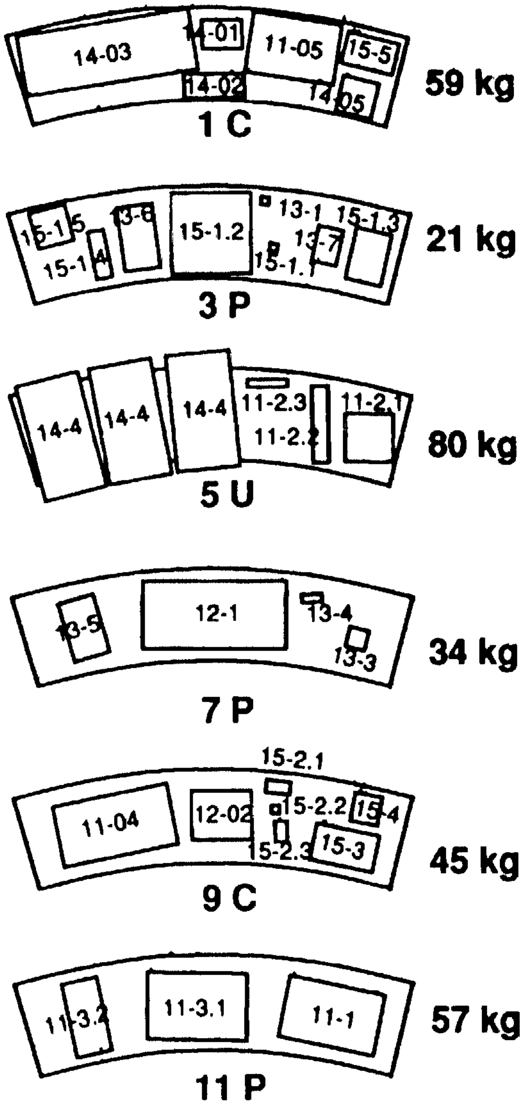
030

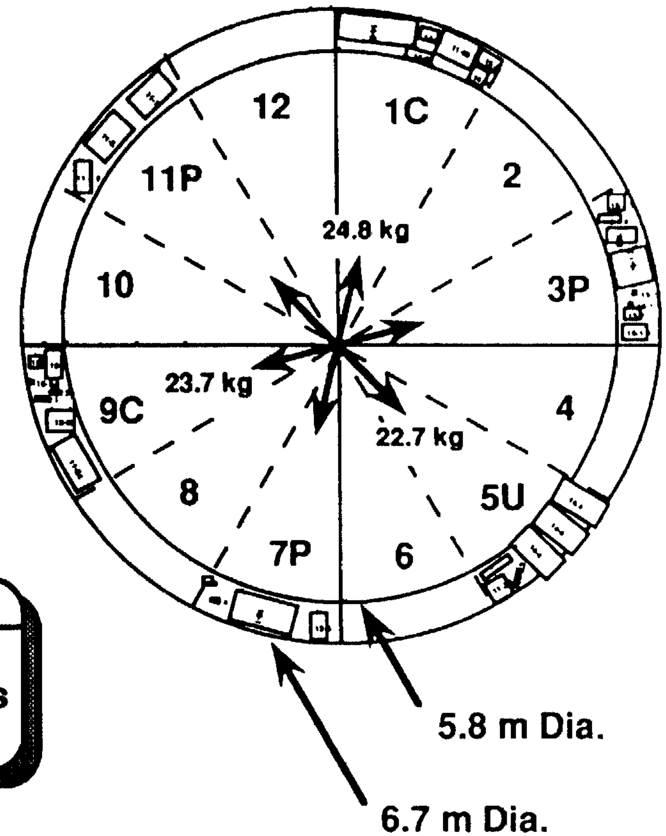
RM921008-02

TLI Stage Avionics Deck Layout

MSFC



SUB-SYS	CODE
GN&C	11-xx
Mission Manager	12-xx
Comm.	13-xx
Power	14-xx
RSS	15-xx



Designation
<u>U</u>nique Elements
<u>C</u>ommon Elements
<u>P</u>ortable Elements

Even # Segments Are Variable Support Structure

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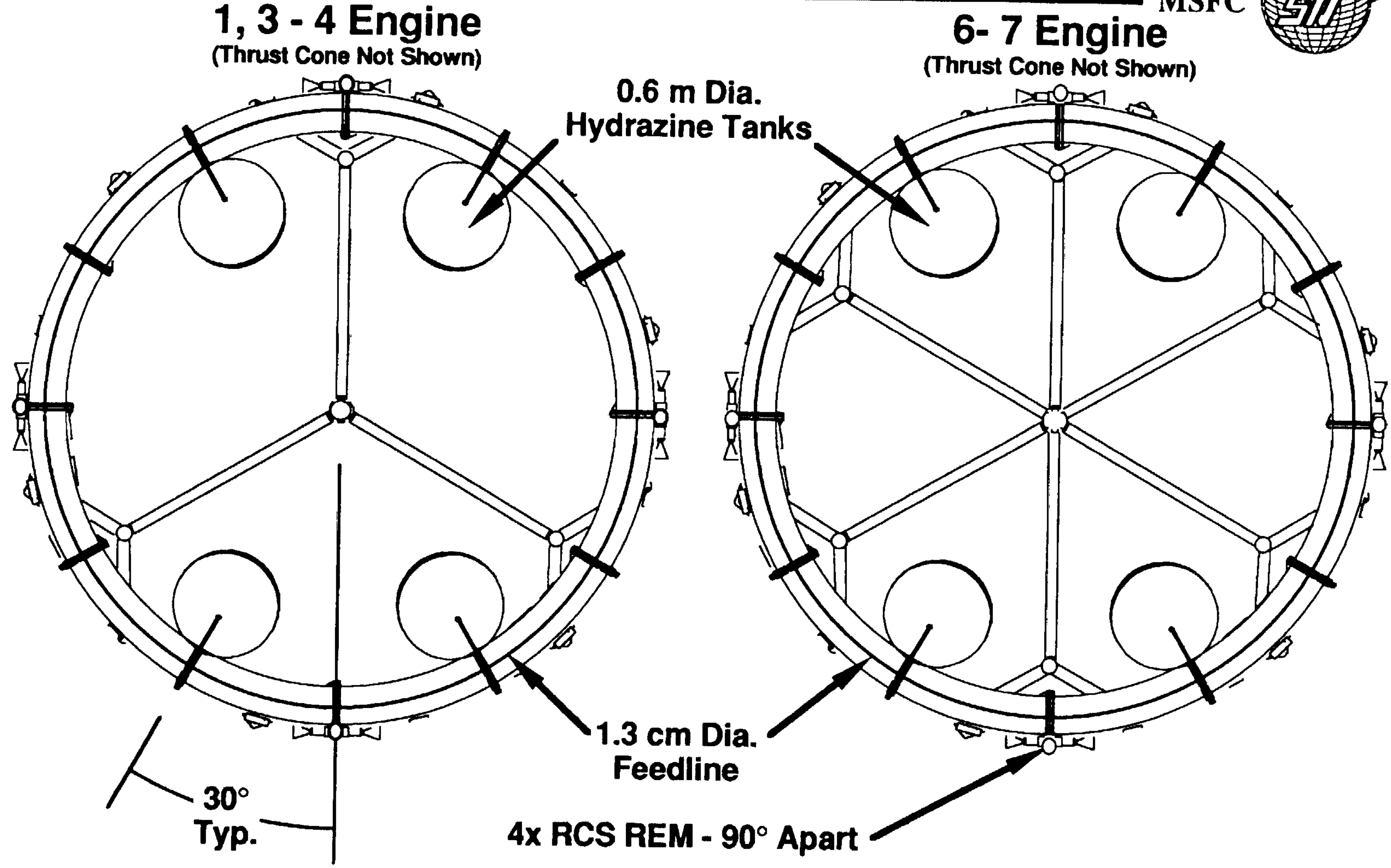
031

RS920618-01B

Detailed Configuration - RCS Placement

MSFC



For Engine Configurations From 1 to 7; No Relocation Needed of RCS System Outside of Possible Addition of Prop. Tanks

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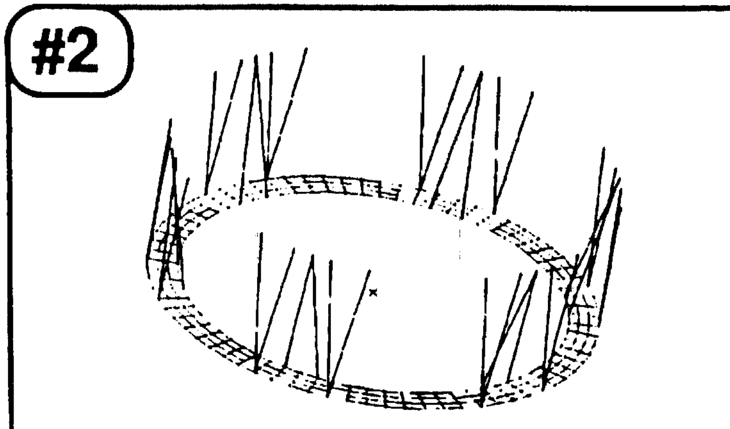
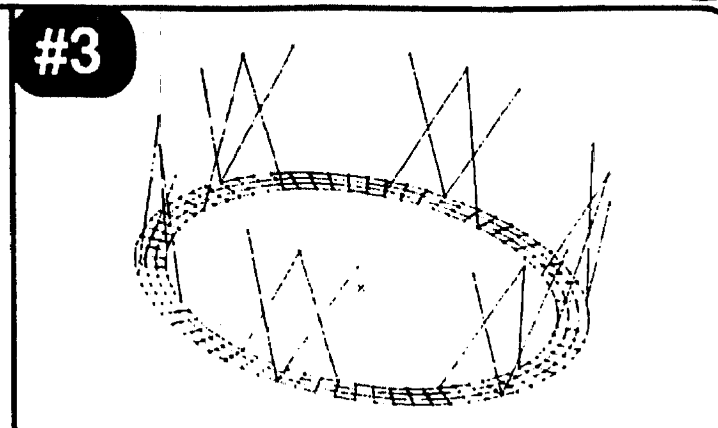
P/A Module - Avionics Deck Strut Options

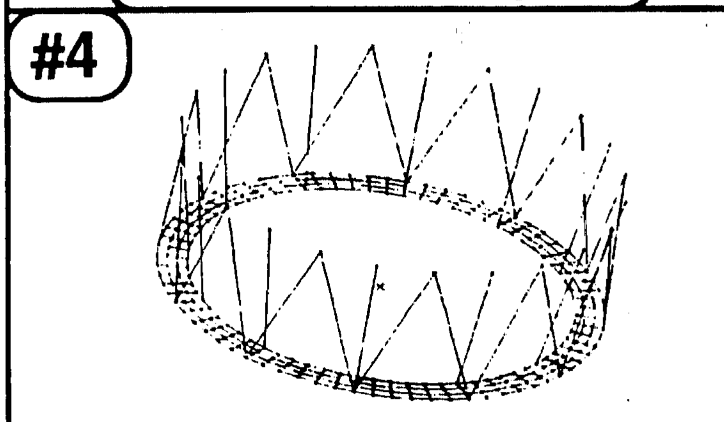
MSFC



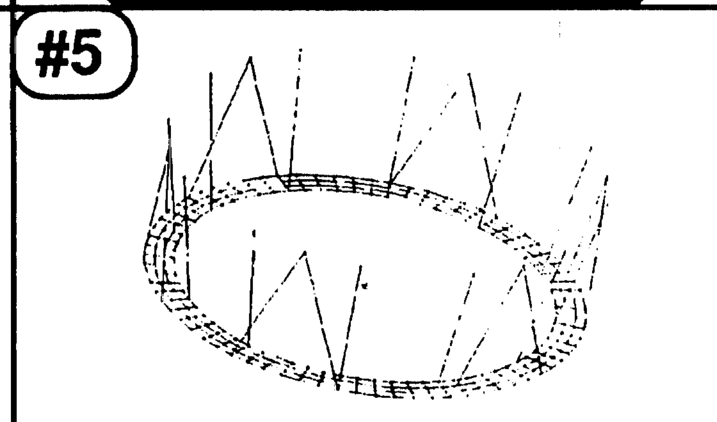
36 Strut - Avionics Segment



24 Strut - Avionics Segment

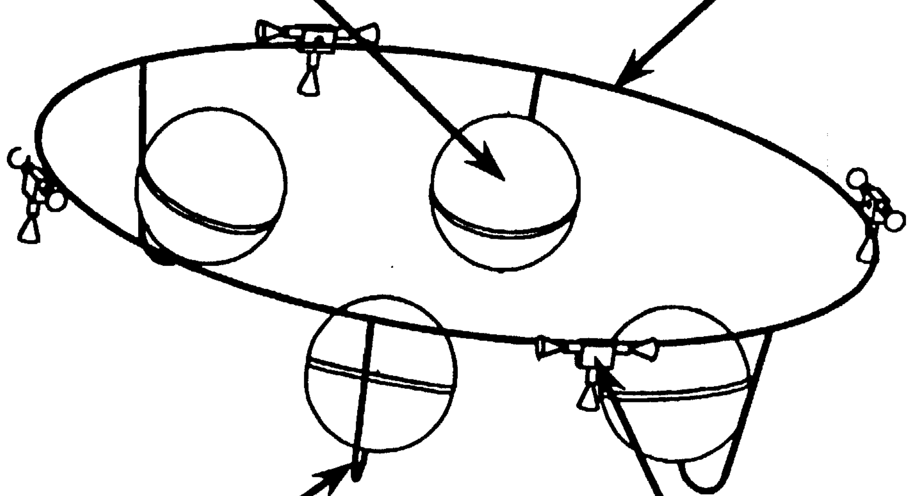


36 Strut - Bulkhead Only



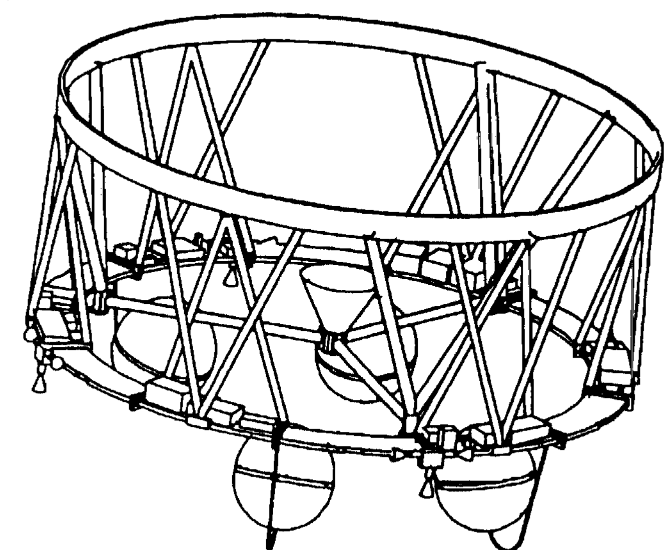
24 Strut - Bulkheads Only

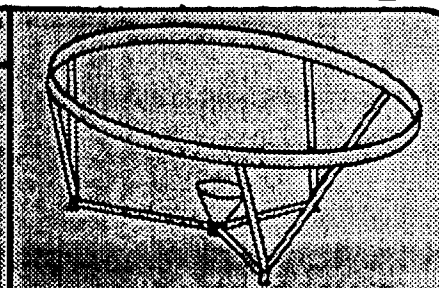
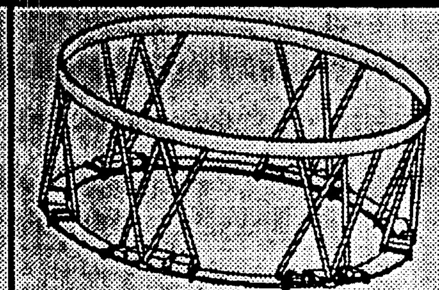
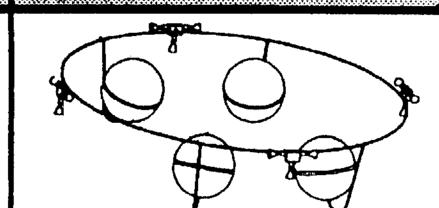
P/A Module - RCS Configuration



P/A Module - Main Structural Grouping

MSFC



Description	
Radial Strut 11.4 Dia* Vert. Strut 12.7 Dia* Cntr. Thrust Cone Cntr Hub 650,000 lbf Outer Pnt 285,000 lbf	
Basic Elem. Cnt. = 5	Engine Thrust Structure
Inner Strut 8.9 Dia* Outer Strut 8.9 Dia* Avionic Mount Sgmt Ring Spacer Sgmt Grouped Avionics RCS Mounts	
Basic Elem. Cnt. = 6	Avionics Mount
Hydrazine Tanks Main Feed 1.3 Dia* Rocket Eng. Mod 4x 20 lbf -or- 100 lbf	
Basic Elem. Cnt. = 4	Reaction Control System

* All Units in "cm" Unless Otherwise Noted

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Detailed Configuration - RCS Placement

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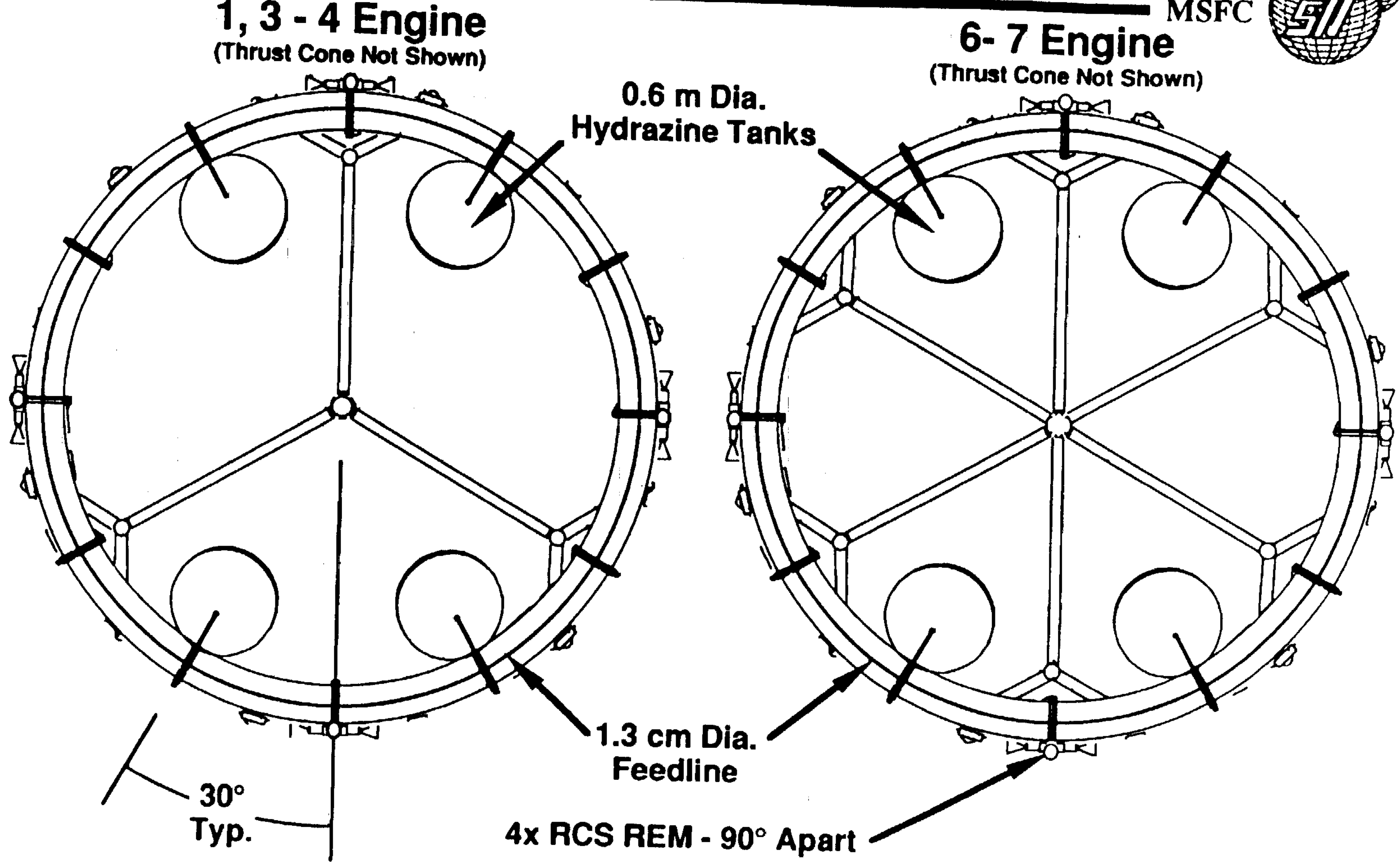

For Engine Configurations From 1 to 7; No Relocation Needed of RCS System Outside of Possible Addition of Prop. Tanks

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Reaction Control System (RCS) Modularity

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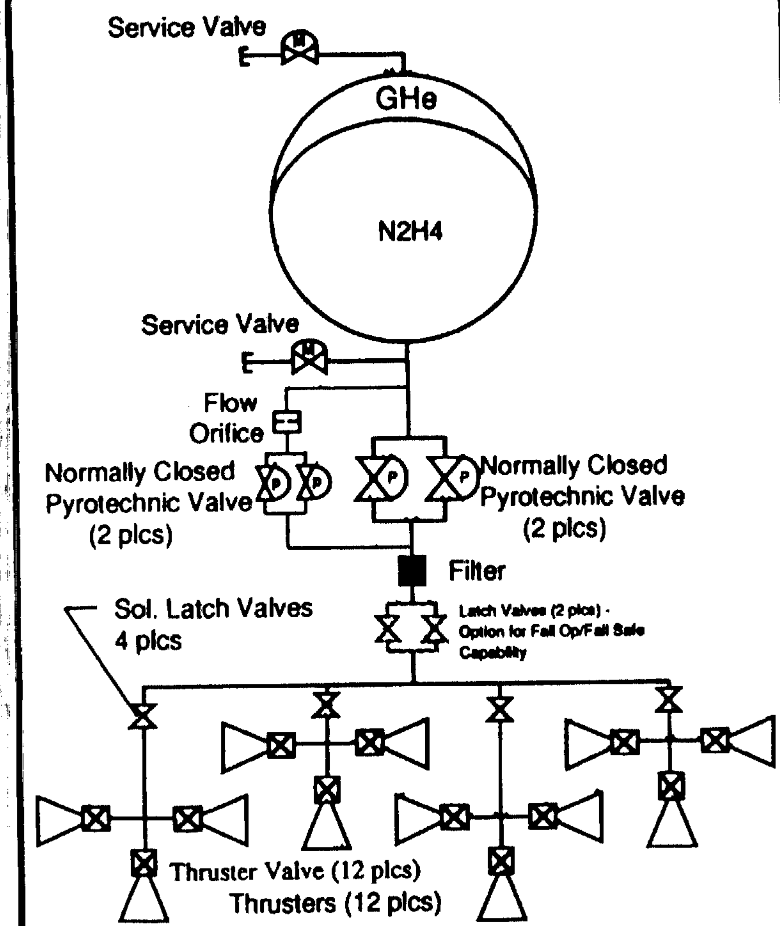
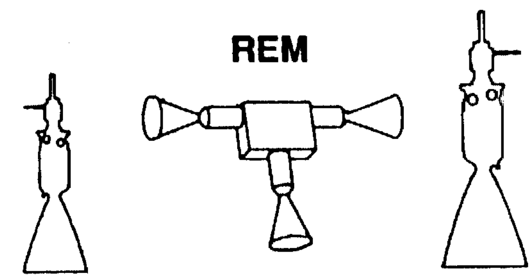


Description & Commonality



20 lbf	Characertistics	100 lbf
25 - 9	Thrust / SS (lbf)	129-46
355 - 105	Feed Pres. (psia)	420-100
115 - 45	Chmbr Pres. (psia)	155-56
0.11-0.043	Flow Rate (lbm/sec)	0.53-0.20
0.90	Weight (lbm)	4.11
229 - 222	ISP (lbf-sec/lbm)	239-233

Commonality:

- Hydrazine Tanks Can be Added and Deleted as Required
- Thrusters Can be Interchanged with no structural Impact
- Feedlines Diameter is Same for Both Thrusters
- Single Fault Tolerance System

PA Module Reaction Control System

MSFC

Issues:

<u>Upper Stage/TLI</u>

- Propellant Load Flexibility Required to Accomplish Various Mission Maneuvers
- Large Propellant Requirement for TLI Stage due to Overall Weight during Main Propellant Settling
- Potential Use of GO2/GH2 RCS to Achieve Propulsion System Commonality

<u>Lander</u>

- Additional Degrees of Freedom Requirement - 16 Thrusters as Opposed to 12 (Addition of 4 Forward Facing thrusters)
- Main Propellant Slosh During Pitch Over Maneuver

<u>Ascent</u>

- Thermal Control Requirement for the Hydrazine Propellant due to Lunar Temperature Extremes, 170° - 700 °R
  - Freezing Point 495 °R
  - Boiling Point 698°R
- Use of Biprop RCS if Storable Fuel is Used for the Main Propulsion in order to Achieve Propellant Commonality

Issues Associated with RCS Design for Various Vehicle Applications have been Identified. These Issues would have to be Addressed in the Design if the Goal is to Achieve a Common System for all the Stages

Configuration - Summary

Modular Propulsion / Avionics System

Accomplishments:

- Performed Multiple Trade Studies and Analyses at Different Subsystem Levels Resulting in The Current Design
- Generated a Database That Bounds the Design of Secondary Structure, Due to Acoustics Environment, For HLLV's
- Generated a New Way of Grouping The Avionics That Enhances The Build Process Flow Cycle
- Developed a <u>Modular</u> Approach to Structural Build - up of The P/A Module

Key Issues:

- Improved Test and Assembly Process Due to Modular Grouping of Avionics
- Reduced Limitations of Multiple Engine Adaptation
- The P/A Module is a Robust System, Well Suited For Growth and Evolvability Into Multiple Vehicle Applications

Ground & Flight Operations

P/A Module

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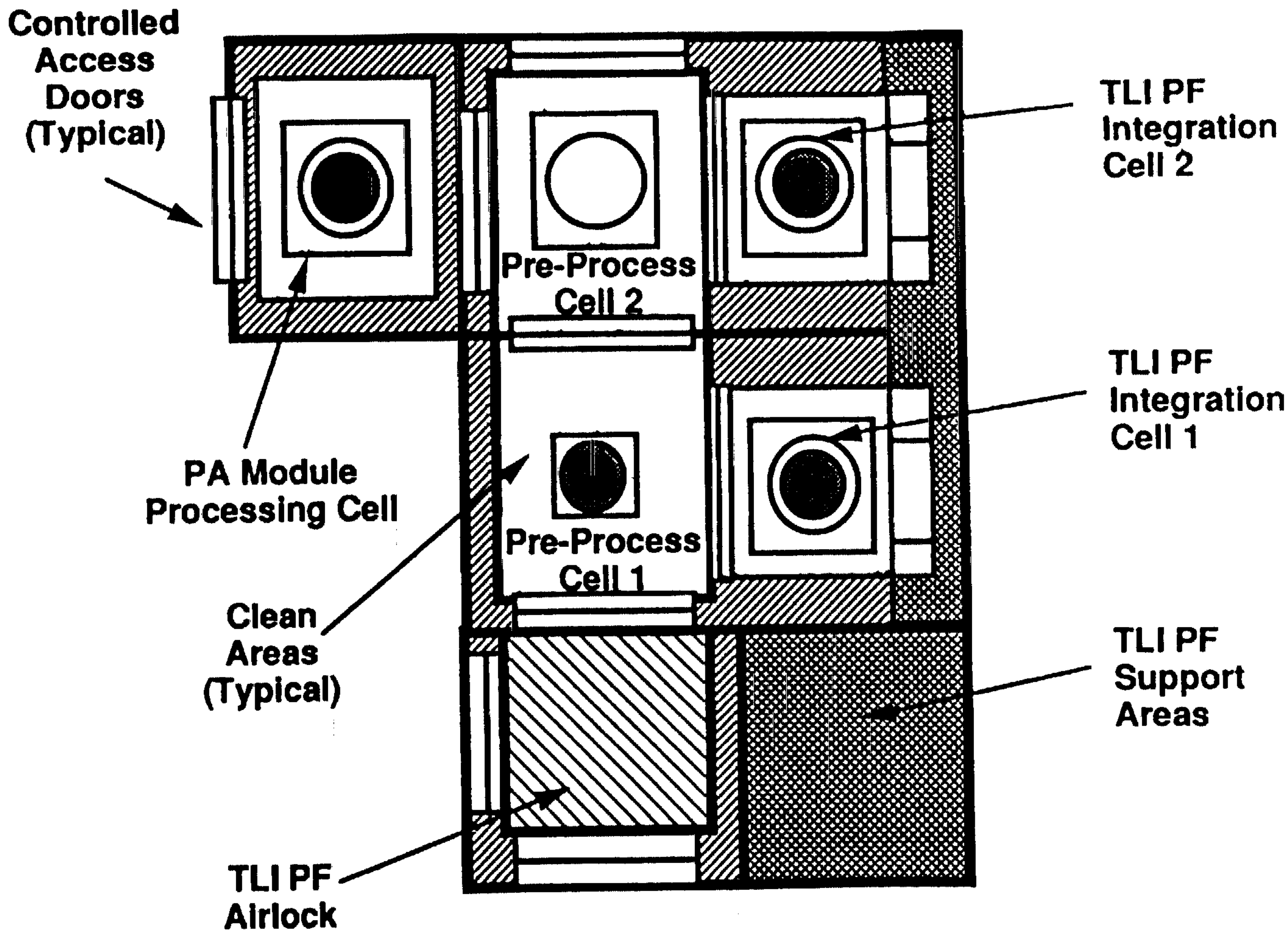
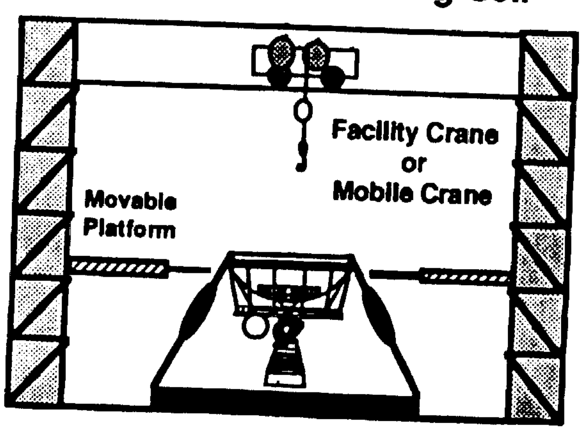
TLI PF With PA Module Processing Cell



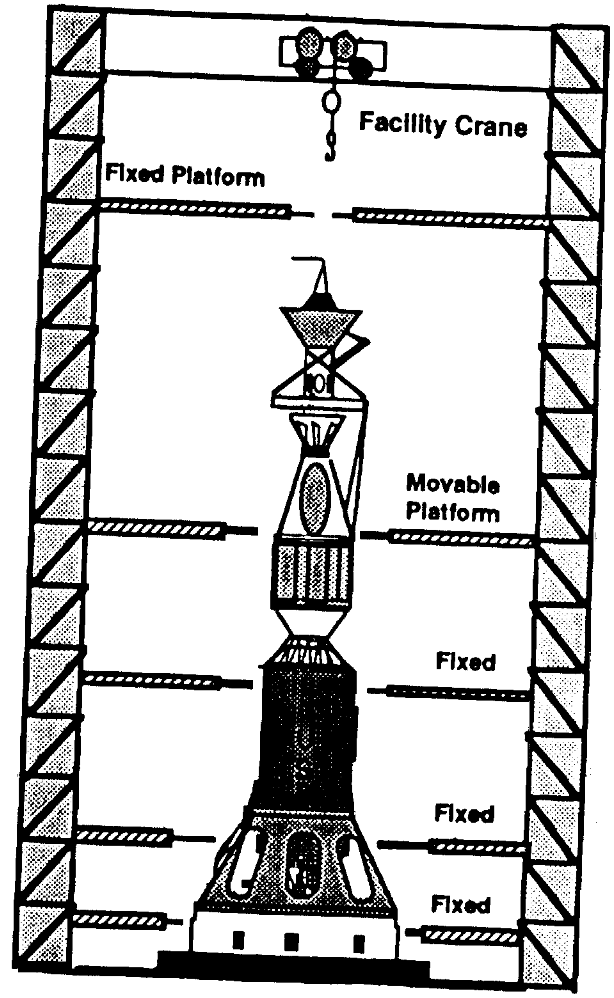
PA Module Processing Cell Activities

The PA Module Processing Cell is about 1/3 the Height of the USPF Integration Cell. This Reduced Height Provides the Opportunity to be Flexible in the Location of the Cell.

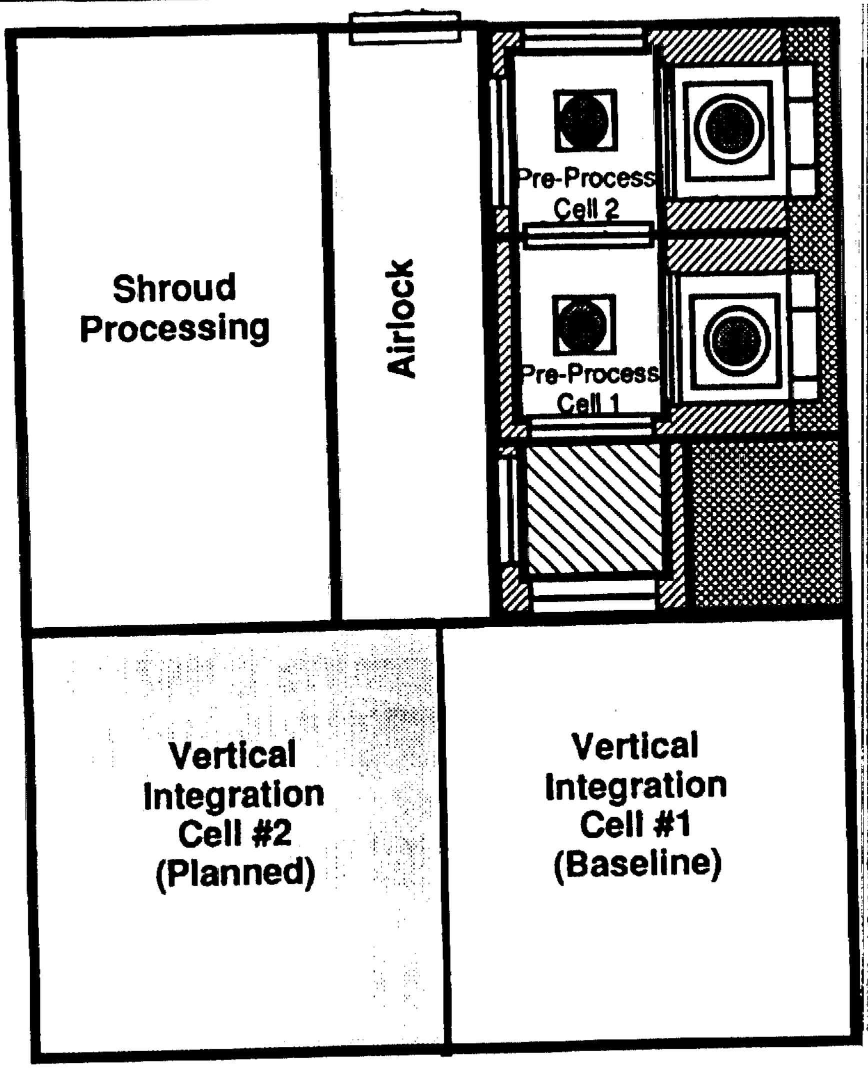
PA Module Processing Cell



USPF Integration Cell



TLI PF With Vertical Integration Building



- Approach is Consistent with the VIB Philosophy of Fully Integrated Processing
- PA Module Processing Can be Accomplished in Separate Cell or in Pre-Process or Integration Cells
- Processing Timelines May be Reduced Due to Less Transport Time Between Facilities

Aerospace Ground Equipment Description

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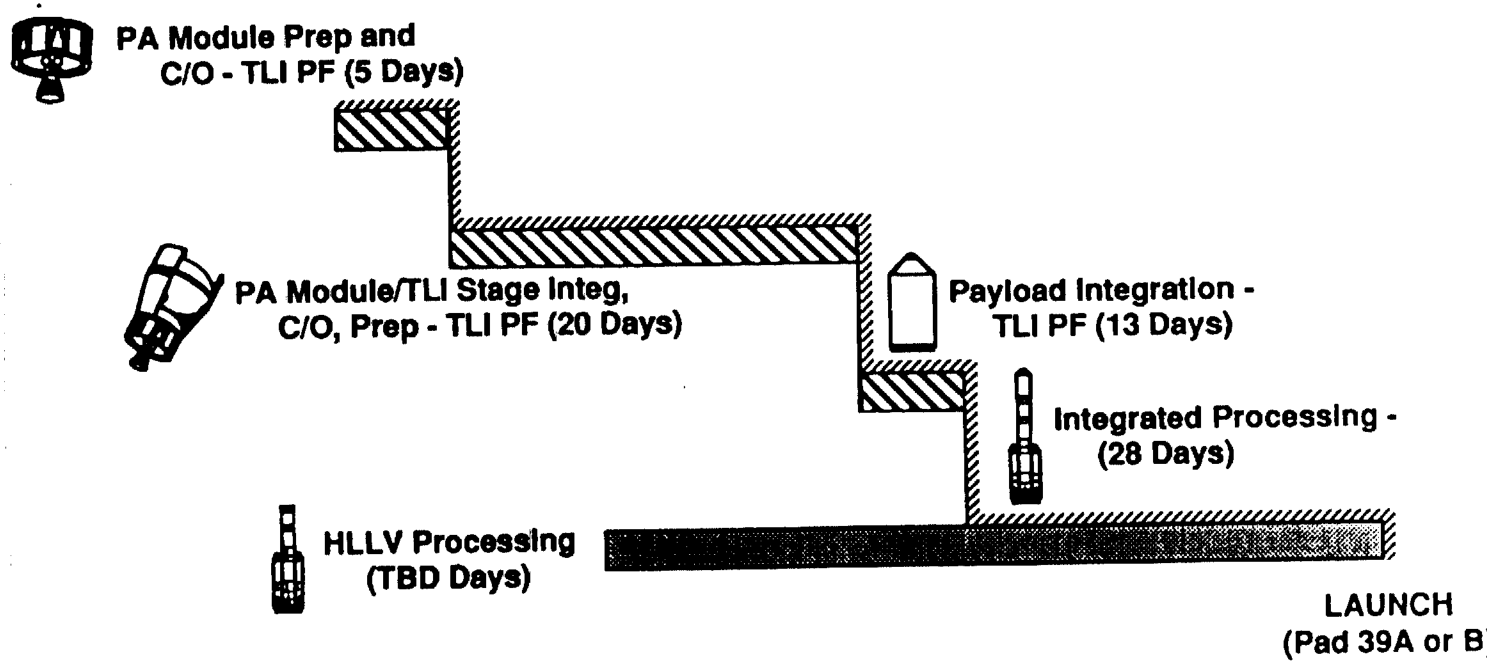
Electrical AGE Description	Factory	PAMPF*	MLP	LP	Quantity	Notes/Comments
Host Vehicle Simulator	X	X			6	3 Mission Classes
MLP/Mast Simulator	X	X			3	
Radio Frequency Test Set						
AVE Simulator						
Workstations						
Transducer Test Set						
Airborne Battery Simulator						
Electrical Circuit Test Set						
Low Voltage Continuity Tes						
Propellant Tanking System						
Pneumatic Control Unit						
Ground Station						
Ordnance Firing System Te						
Antenna Hats (Couplers)						
Launch Vehicle Simulator						

Mechanical AGE Description	Air Field	Factory	PAMPF*	MLP	LP	Quantity	Notes/Comments
Forward & Aft Shipping Adapter	X	X	X			3	Three Sets
PA Module Cradle Assy	X	X	X			3	
Tractor/Trailer	X	X	X			3	GFE
Aft Adapter Shipping Container	X	X	X			3	
RCS Tank Installation/Removal Set		X	X			4	
RCS Tank Shipping Container		X	X			4	
Pneumatic Service Cart (40-3500 PSIG)		X	X	X	X	4	
Access Platforms		X	X			5	
Load Positioner		X	X			4	GFE
PA Module Lifting Sling		X	X			4	
Electric Fork Lift		X	X			4	
Aft Adapter Lifting Sling		X	X			4	

Mechanical and Electrical AGE Element, Quantity, and Location Requirements Have Been Identified

Operational Timeline Summary

TLI PF @ 2 Shifts/Day
VAB @ 2 Shifts/Day
Launch Pad @ 2 Shifts/Day (3 Shift for Terminal Countdown)



66 Days Serial Processing Required for Integrated TLI/PA Module Stage (Payload Processing Completed at Separate Facility)

PA Module Flight Operations

Key Factors Affecting Flight Operations

Overview

- Mission Type/Duration
- Manned vs. Unmanned
- Level of Autonomy
- Communications
- PA Module to Payload Interfaces and Functional Allocation
- Human Factors

PA Module Flight Operations

MSFC

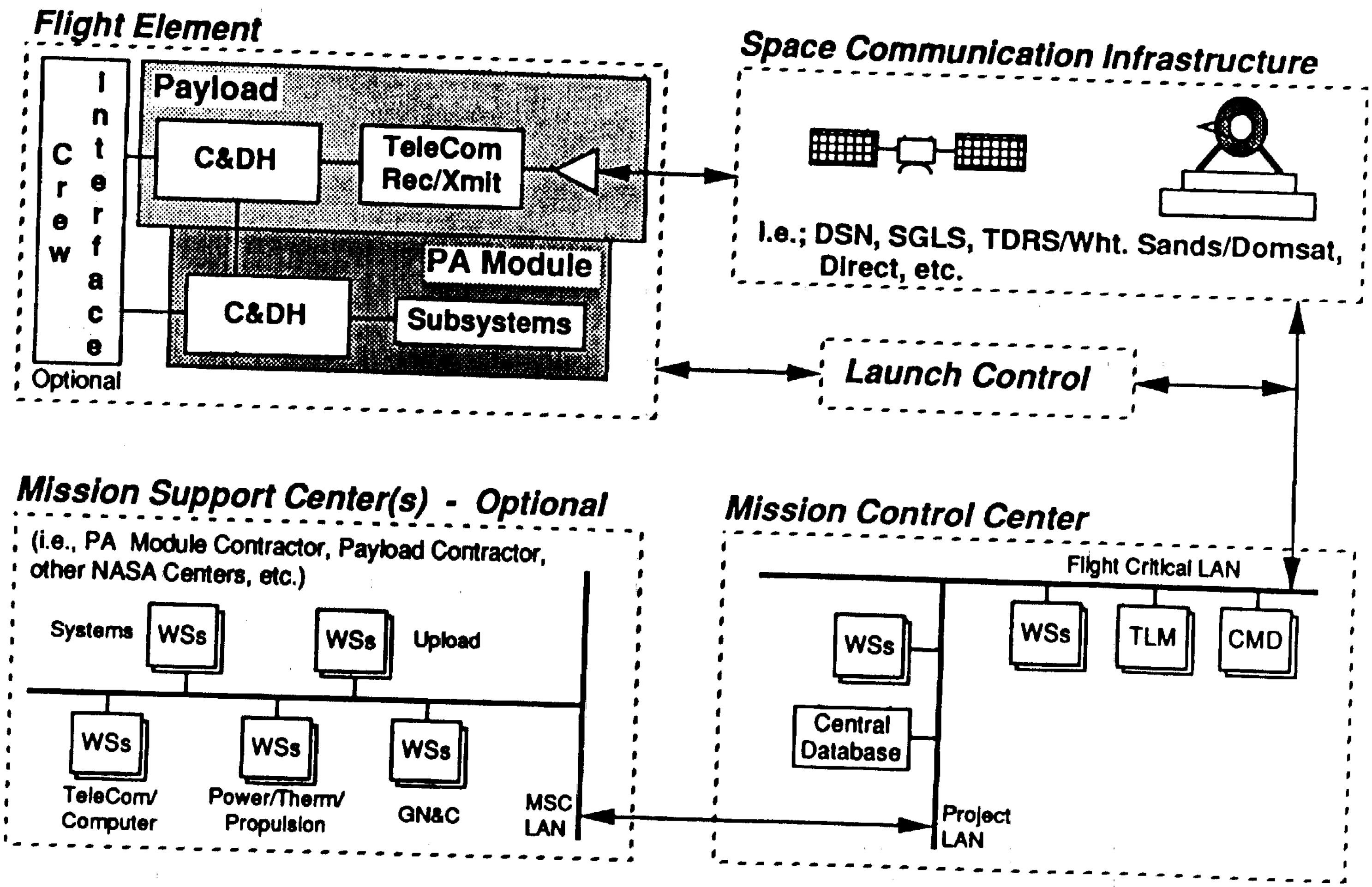
Flight/Mission Operations Program Phases

Program Phase	*Activities*
Flight/Mission Operations Development/Planning	• Support PA Module Requirements Analysis • Define Functional Cababilities and Ops Interfaces • Determine Functional Allocations of Ops Req'ts • Develop Flight Rules/Procedures/Constraints • Define Telemetry/Command Data Elements • Develop Detailed Sequence Definitions • Perform Mission Timeline/Flight Ops Event Analysis • Develop/Modify Flight Operations S/W
Flight/Mission Operations Integration	• S/W Integration into Data/Comm Systems • Support of PA Module Ground Testing (synergistic use of H/W, S/W and Personnel) • Contingency Planning and Sequence Generation • End-to-End Flight Ops Infrastructure Testing • Flight Team and Crew Training/Test • Pre-Mission Simulations
Flight/Mission Operations	• Real-Time Data Monitoring • Non Real-Time Data Analysis/Trending/Prediction • Sequence/Command Generation • Flight S/W Maintenance • Comm Systems I/F Coordination • On-Board Flight Crew Activities

PA Module Flight Operations

MSFC

Flight/Mission Operations Elements/Infrastructure



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PA Module Flight Operations

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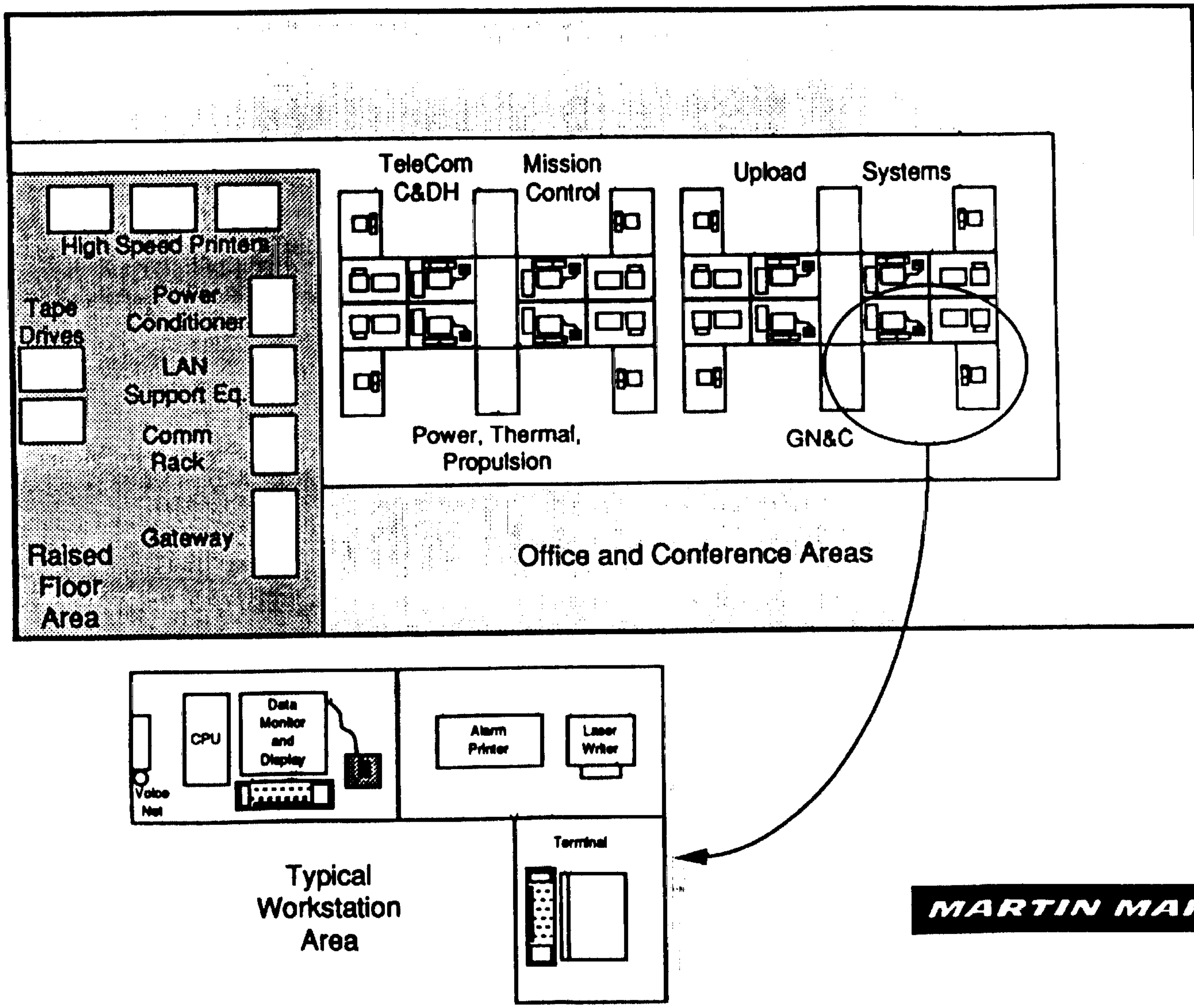
Example PA Module Mission Support Center Configuration



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Programmatics

P/A Module

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PA Module Development Program Overview

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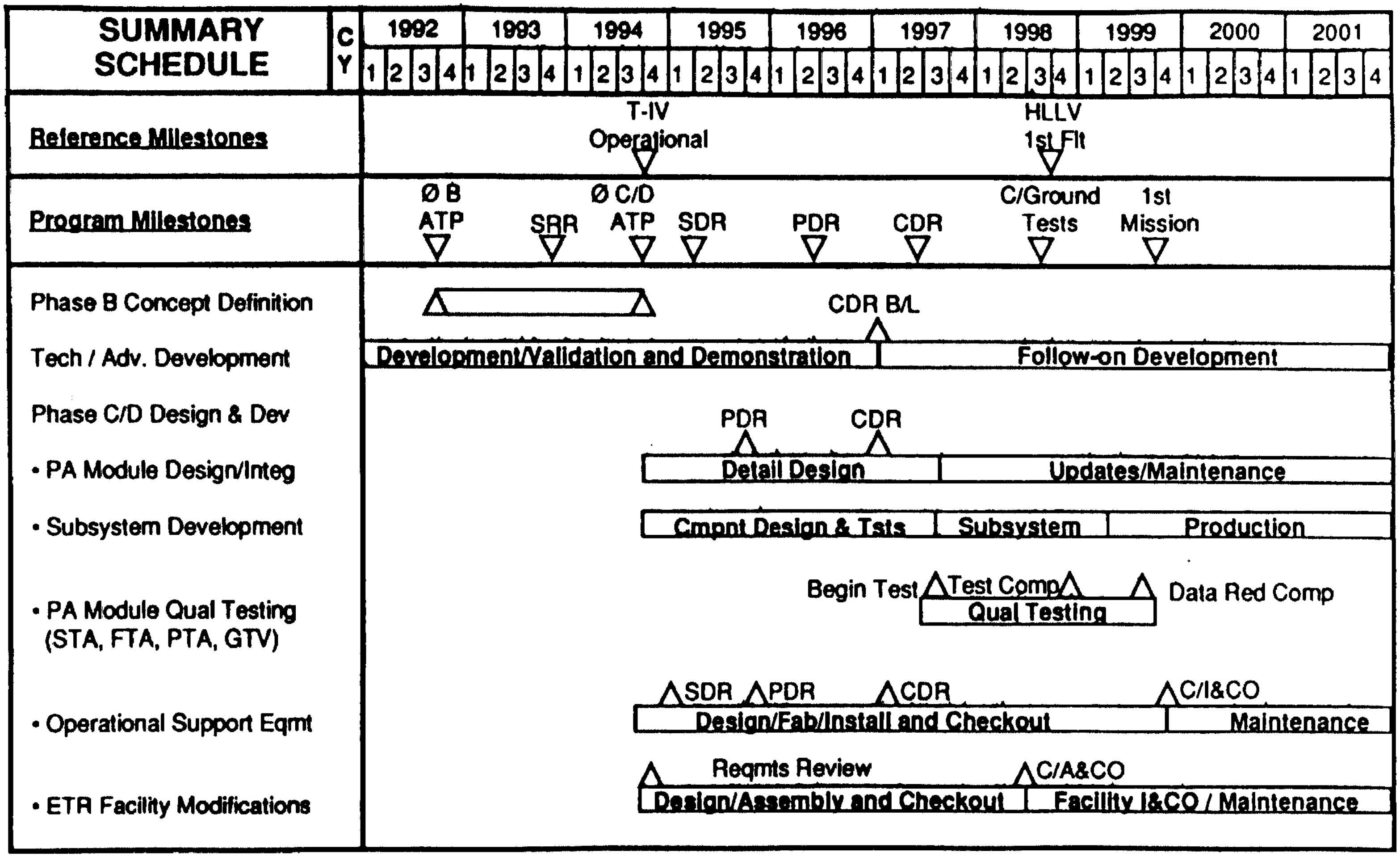

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Propulsion Avionics Module Technology

MSFC

Approach:

- Identify Applicable Technologies for PA Module
- Evaluate Technologies for Readiness and Benefits
- Establish Development Plan for Technology Implementation and Integration

Results:

- No Enabling Technology Identified for the PA Module
- Identified High Priority *Enhancing* Technologies Directly Related to Performance, Weight and Cost
  - Developed Technology Roadmap
  - Established Technology Readiness Level
  - Developed Benefits of Enhancing Technologies

No Technology/Advanced Development Required to Support Development of PA Module (Growth for Mars Missions Will be Needed)

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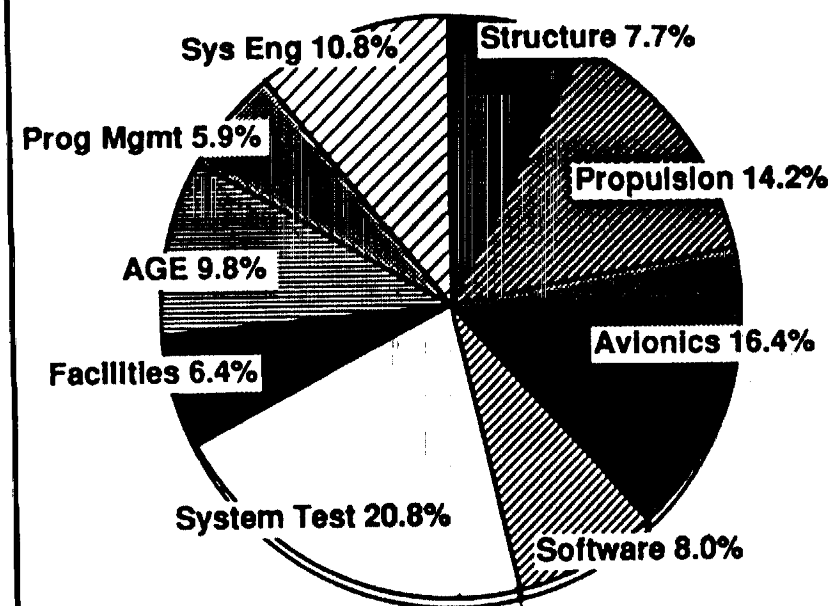
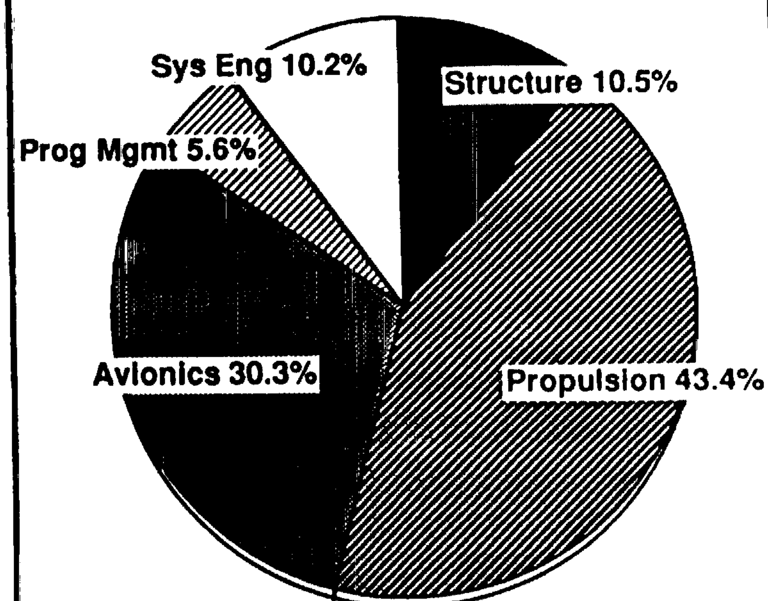
PA Module Cost Summary

PA Module FSD Cost = $225 M



PA Module TFU Cost = $70 M



Propulsion Avionics Module Developed,Tested, and Ready for Production for Approximately $300 M (40% of TLI Stage Costs)

Summary & Conclusions

P/A Module

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PA Module Study Summary

Propulsion Avionics Module Study (TD10) is Complete

- Groundrules and Assumptions Defined and Approved by MSFC
- Design Reference Missions Developed and Approved by MSFC
- Preliminary System and Subsystem Design Requirements Document (A-Level Specification) Prepared and Submitted
- Technology/Advanced Development Analysis Completed
- Initial Candidate Concepts Defined and Characterized
- Concept Downselect Completed and Final Concept Design Detailed
- Programmatics (Cost and Schedule) Analysis Completed
- Ground and Space Operations Analysis Completed

Work Continues on PA Module Benefits Assessment in the Upper Stage Requirements and Concepts Study (TD13)

Technical Directive 11

Cryogenic Lander Study (FLO)

Agenda - 19 March

MSFC

- Groundrules/Assumptions
- Configuration Selection
- Detailed Analysis
  - Performance
  - Cargo Handling
- Mission Functional Analysis
- Systems Risk Assessment
- Mission Abort Analysis

Sid Earley

John Hodge

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Groundrules and Assumptions

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TLI Stage Interface

- Post TLI Payload Capability Is 76 t
  - Current Baseline Is 93 t
- The Structure Between the TLI Stage and the Lander Will Be Carried in the Lander Mass Properties
- TLI Stage Will Not Provide Power, Communications and Other Functions for the Lander after the TLI Burn Is Completed

Element Design

- The Return Stage Will Have the Capability of Bringing 200 kg of Cargo Back to Earth
- The Lander Will Have the Capability of Delivering at Least 27.5 t (25 t + 10% margin) of Cargo to the Lunar Surface
  - Current Baseline Is 31 t
- The Lander Will Have the Capability of Delivering at Least 5.0 t to the Lunar Surface on the Piloted Mission
- The Lander Mass Estimates Will Include a 20% Dry Mass Margin
- The Lander Mass Estimates Will Include a 1% Total Propellant FPR
- Crew Module Mass Is 9.2 t (including radiation shielding & consumables)
  - Current Baseline Is 6 t
- The Lander Must Maintain a 1.0 m Minimum Clearence

Groundrules and Assumptions (continued)

- **Total Lander ΔV Budget is 5480 m/s**

Event	ΔV m/s
Trans Lunar Mid-Course Corrections	30
Lunar Orbit Insertion (@ 185 km)	892
Lunar De-Orbit	20
Lunar Descent	1830

Event	ΔV m/s
Lunar Ascent	1810
LLO Circularization	20
Trans-Earth Injection	848
Trans-Earth Mid-Course Corrections	30

- **Descent & Ascent System Isp's**

Event	Isp (sec)
Lunar Orbit Insertion/Ascent/Trans-Earth Injection	444
Lunar Descent	440

- **Mission**

Trans-Lunar/Trans-Earth Transfer Time	4 days/4 days
Lunar Stay Time	45 days
Lunar Surface Propellant Boiloff (1.5% LOX + 7% LH_2)	2.5% per month
Engine Type/Number of Engines (Thrust = 16,300 lbs)	RL10A-3/4
Trapped and Residual Propellants (of total)	1.5%

Four Lander Configurations Considered

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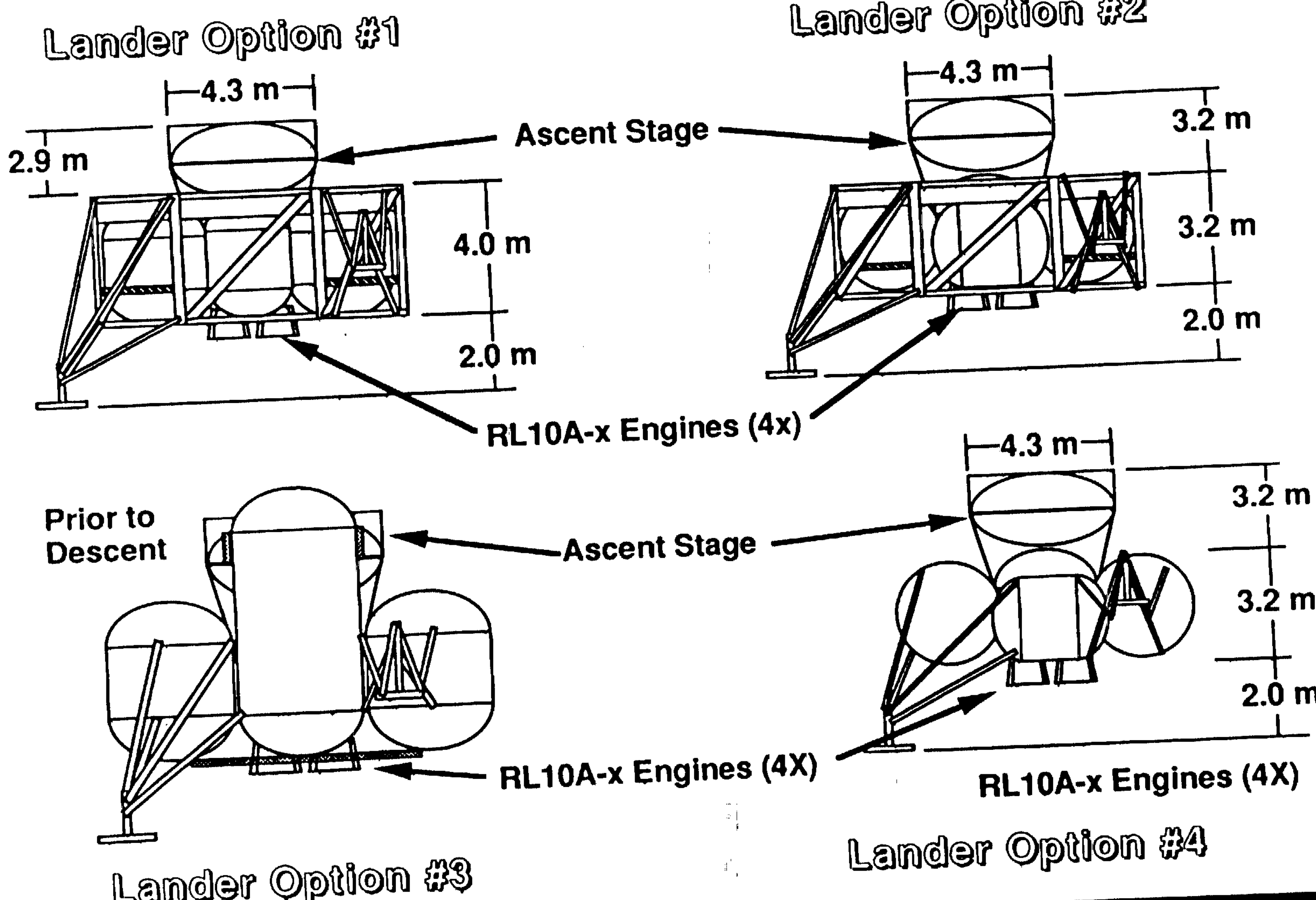

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Comparison of the Lander Configurations

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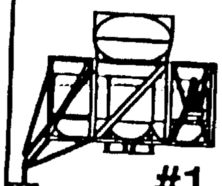
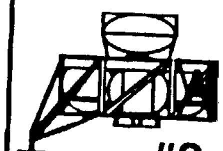
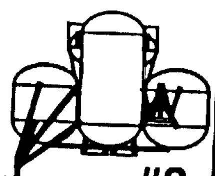
Option	Advantages	Disadvantages	Mass Reduc.	Mass Fraction
 #1	• Close To Surface P/L Platform • Packages in 33 ft Dia. • Few Strct Mod's For Cargo Mission • Conventional Tank Mounting	• Large # of Tanks • Large Surface Area / Volume Ratio	0.0%	0.848
 #2	• Close To Surface P/L Platform • Packages in 33 ft Dia. • Few Strct Mod's For Cargo Mission • Fewer # of Tanks	• Additional Baffles & Acquisition Device Work for Tanks • Non Conventional Tank Mounting	7.7%	0.855
 #3	• Lower Structural Mass • Packages in 33 ft Dia. • Fewer # of Tanks	• Increased Thermal Leak From Tank Attach Structure • No Infinite Plane Cargo Deck • Complex Mechanisms	1.6%	0.849
 #4	• Moderately Lower Structural Mass • Packages in 33 ft Dia. • Fewer # of Tanks	• High Thermal Leak From Tank Attach Structure • Complex Ascent Adaptor • Non-Conventional Tank Mounting	6.9%	0.855

Recommendation: Option #1 as Baseline & Option #2 as Alternate

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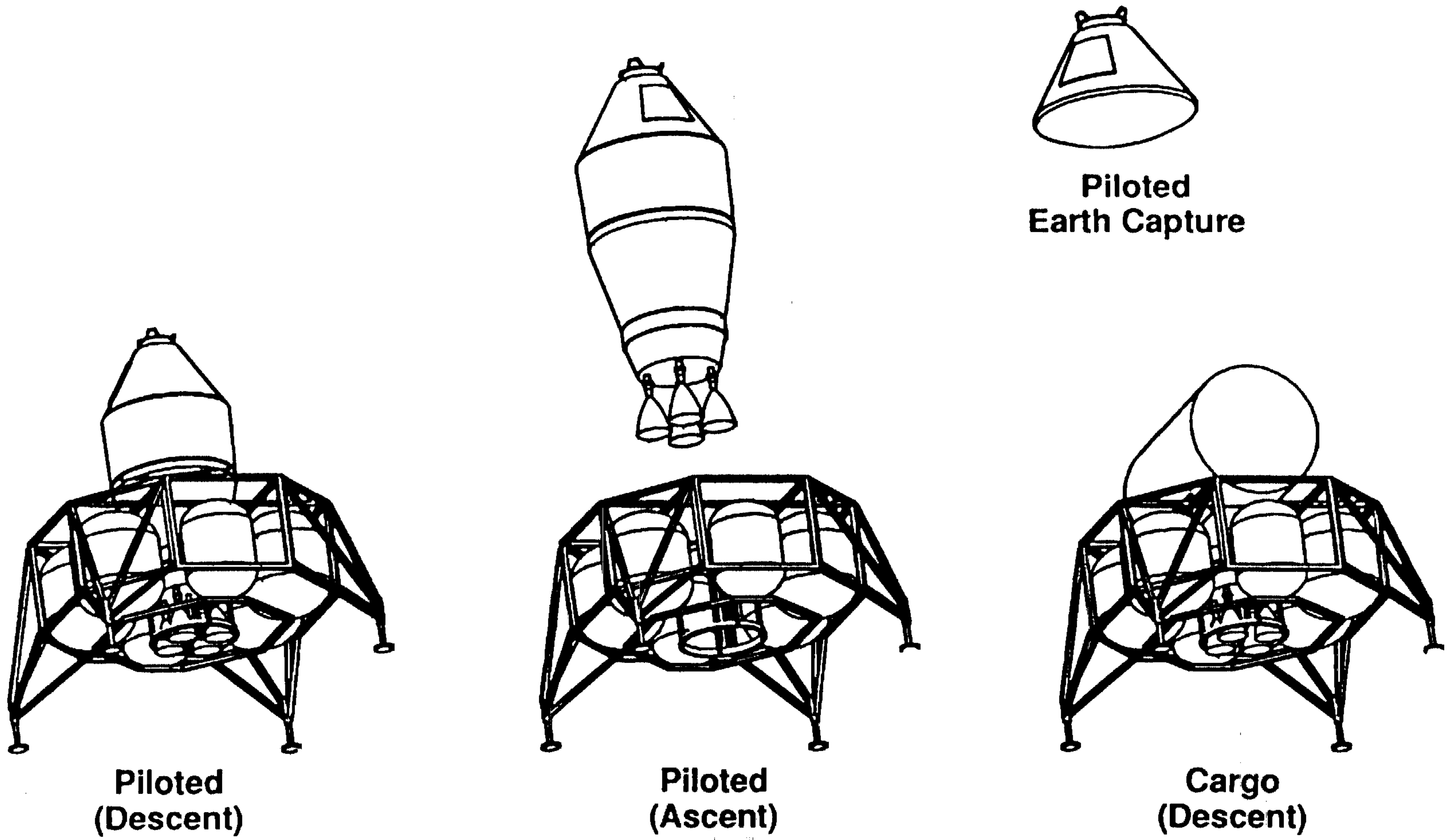
Recommended Configuration

Piloted & Cargo Vehicles with a Common Lander



Configuration Details (Piloted)

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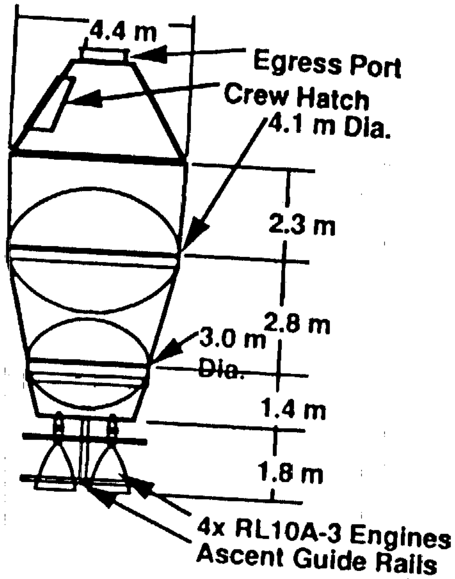

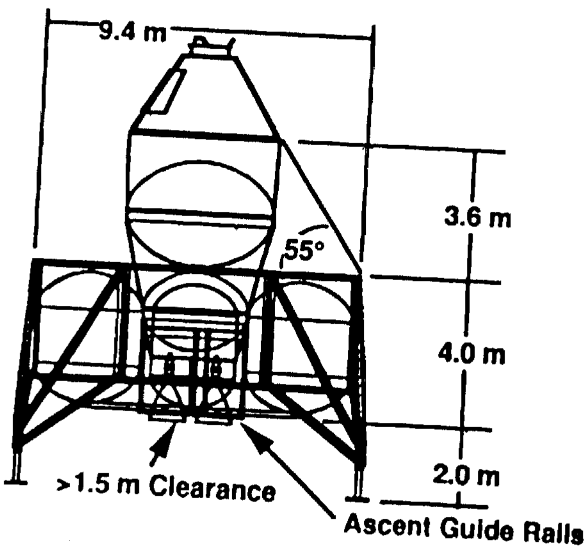

Subsystem Element	Piloted Ascent (kg) *	Piloted Lander (kg) *
Structures & Mechanisms	1359.13	3720.27
Tanks		
Main Propulsion Tanks	657.47	1200.04
RCS Tanks	68.03	90.70
Thermal	134.68	388.01
Propulsion		
Engines	553.29	0.00
RCS	122.45	0.00
Feed System	136.05	142.86
Avionics		
GN&C	290.25	0.00
Power	458.05	0.00
Command & Control	246.94	0.00
Cabling	54.88	5.78
Total	4081.21	5547.66
Growth	816.24	1109.53
Total Dry Mass	4897.46	6657.20
Main Propellant	11642.20	35774.50
RCS Propellant	172.00	1011.00
Total Mass	16711.66	43442.70

172

* 5.5 t Cargo & 9.2 t Cab Not Included

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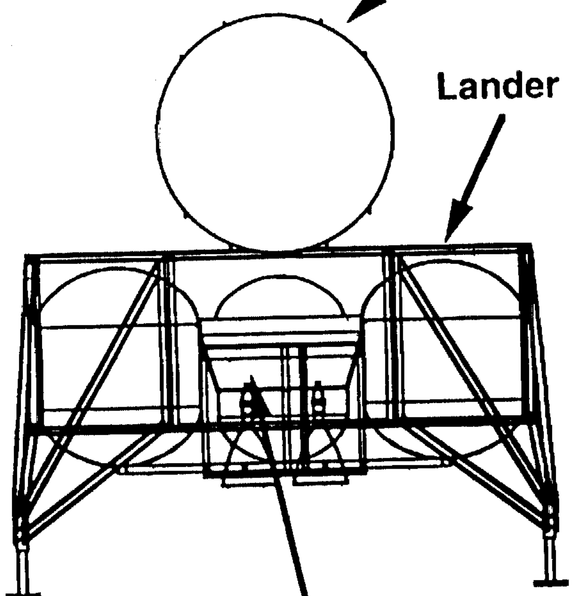
Configuration Details (Cargo)



Subsystem Element	Cargo Ascent (kg)	Cargo Lander (kg) *
Structures & Mechanisms	273.47	4657.01
Tanks		
Main Propulsion Tanks	0.00	1200.04
RCS Tanks	68.03	90.70
Thermal	0.00	388.01
Propulsion		
Engines	553.29	0.00
RCS	122.45	0.00
Feed System	45.35	142.86
Avionics		
GN&C	290.25	0.00
Power	458.05	0.00
Command & Control	246.94	0.00
Cabling	54.88	5.78
Total	2112.70	6647.66
Growth	422.54	1329.53
Total Dry Mass	2535.24	7977.20
Main Propellant	0.00	35774.50
RCS Propellant	172.00	1011.00
Total Mass	2707.24	44762.70

* 27.6 t Cargo Not Included

Configuration Analysis - CG Location

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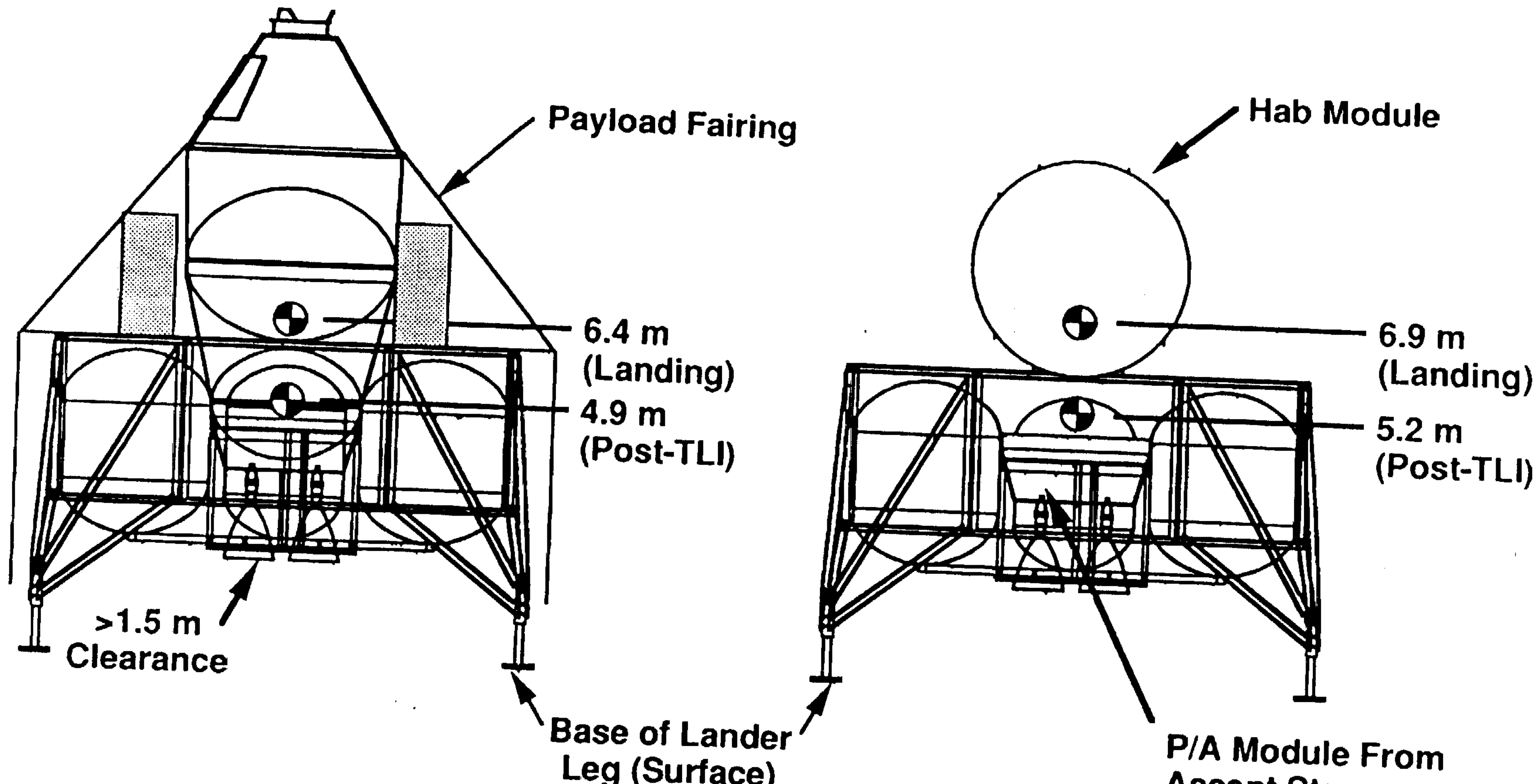
Piloted Mission

Cargo Mission

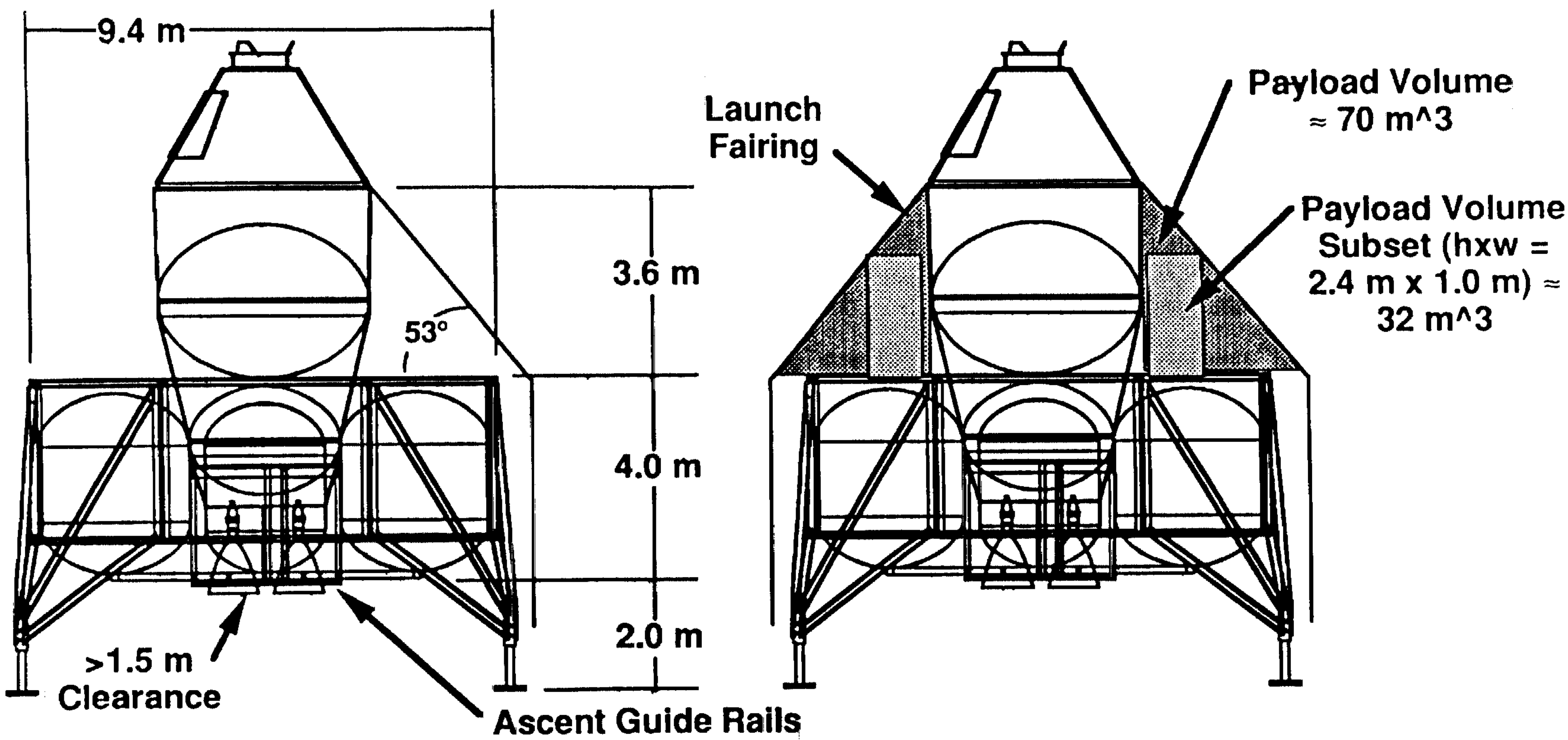


- All CG Locations Are From Base of Landing Leg (Surface)
- Landing Leg Pad Diagonal Diameter 13.95 meters

Configuration Analysis

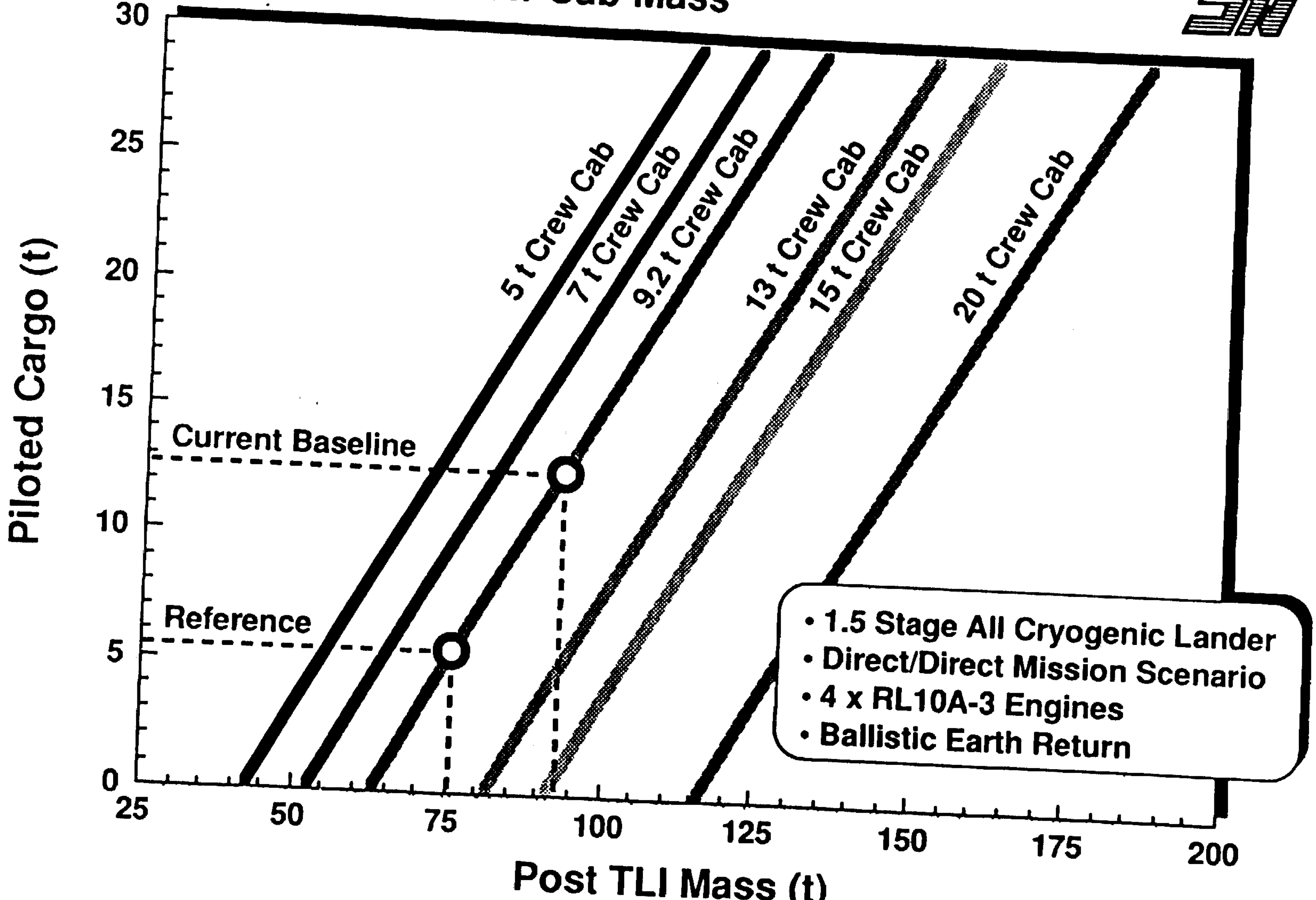
MSFC

Piloted Mission Payload Capabilities



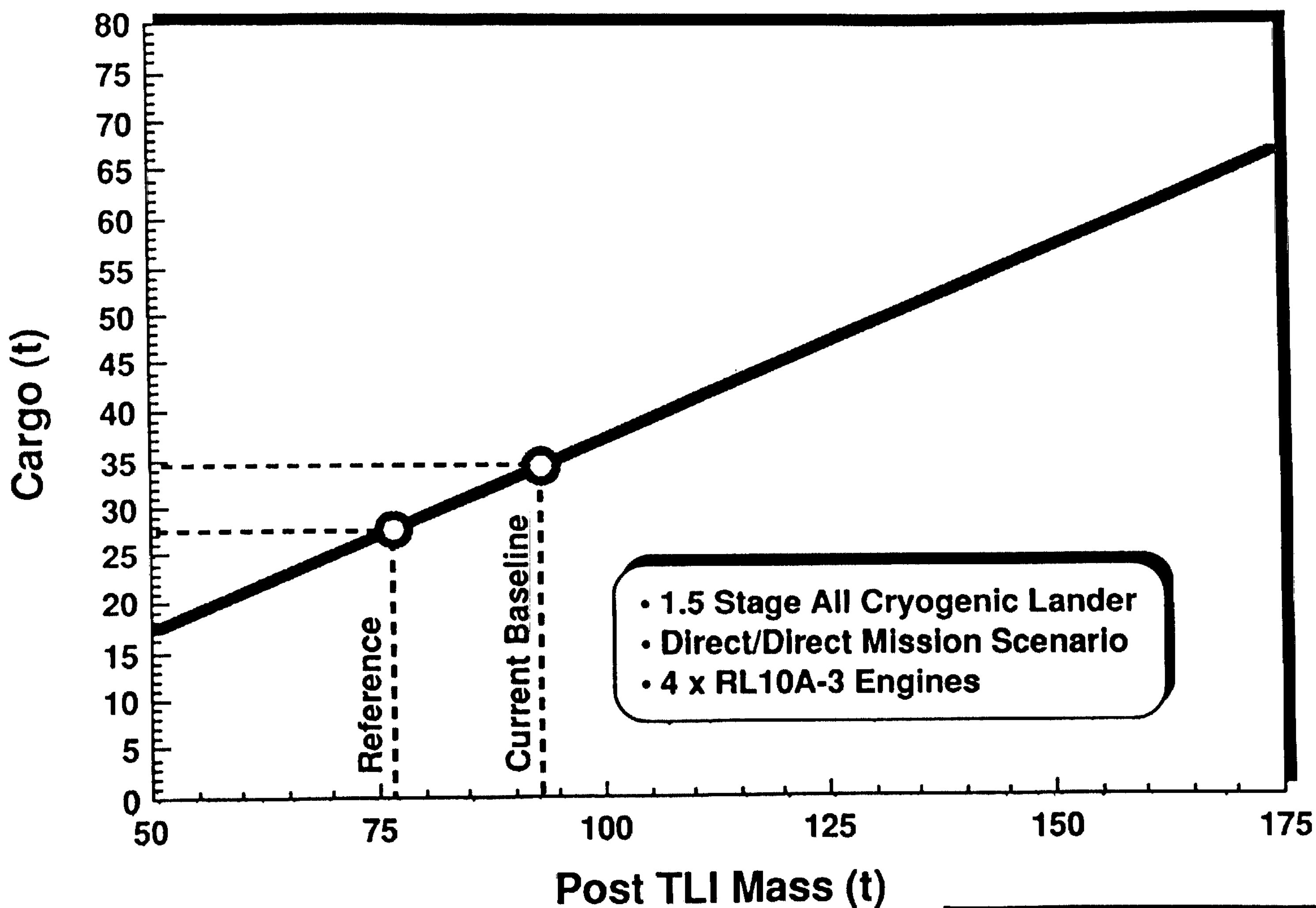
Performance Analysis

Piloted Cargo vs. TLI & Crew Cab Mass



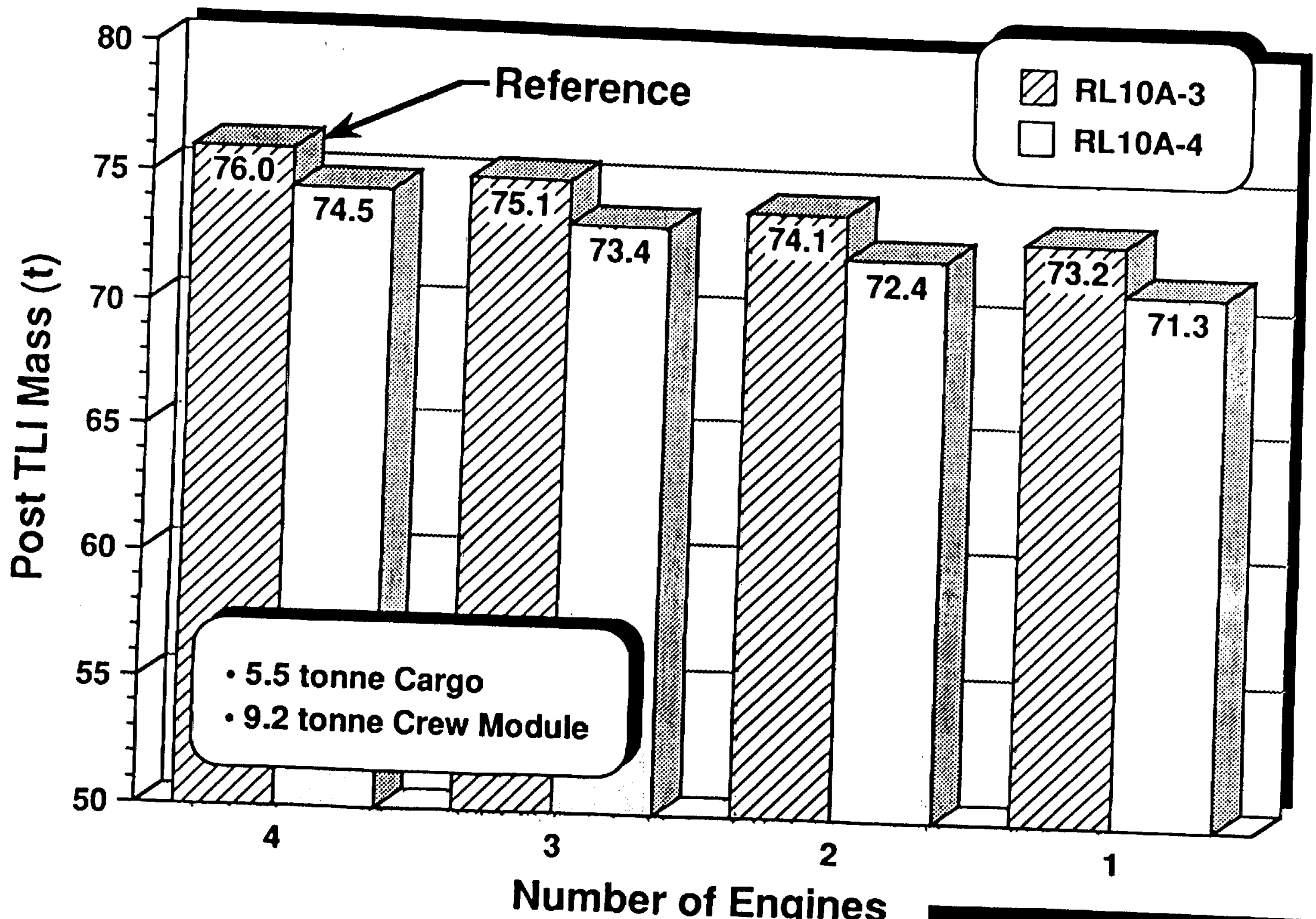
Performance Analysis

Lunar Surface Cargo Only vs. Post TLI Mass



Performance Analysis

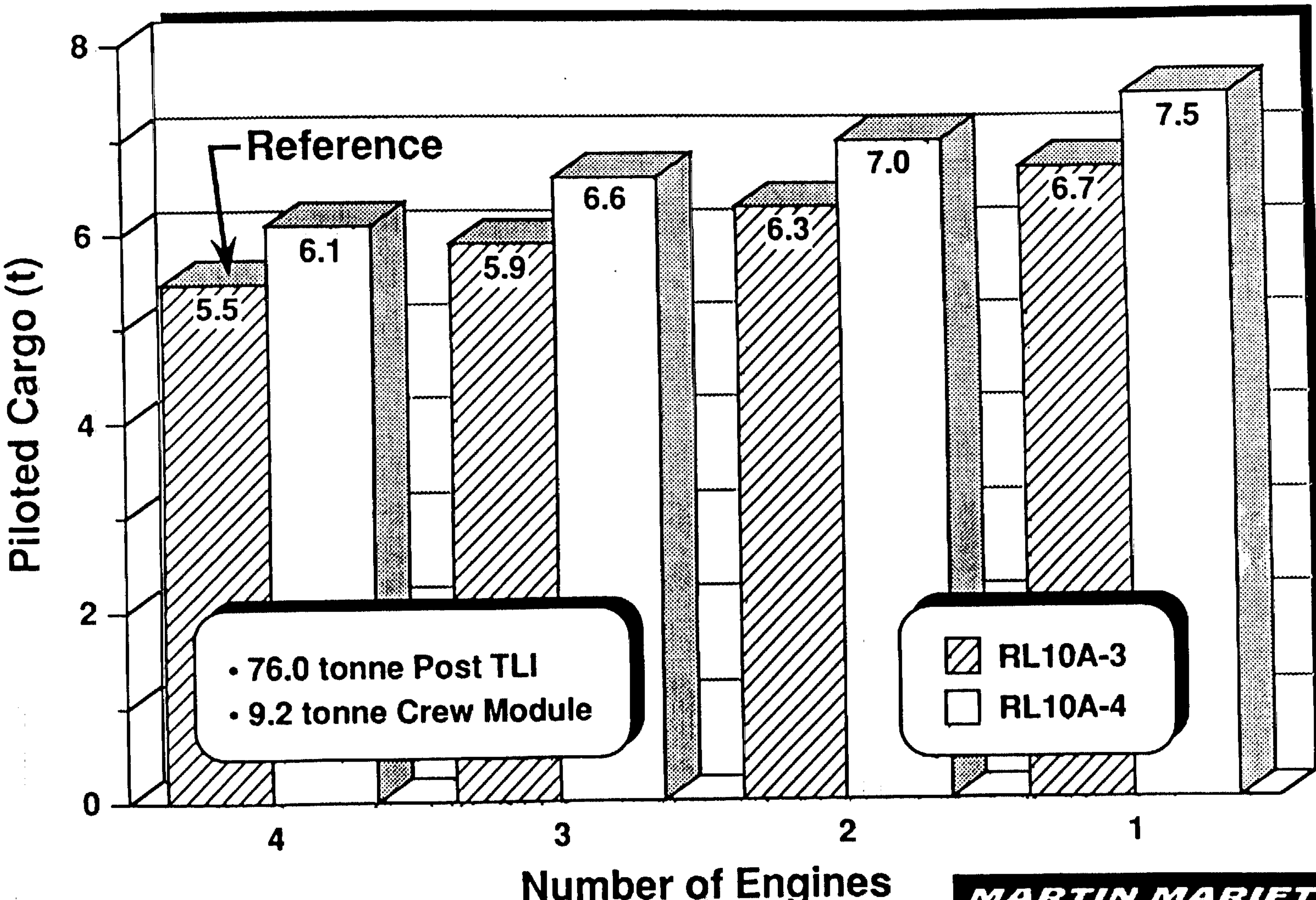
Piloted Engine Parametrics I



Performance Analysis

MSFC

Piloted Engine Parametrics II



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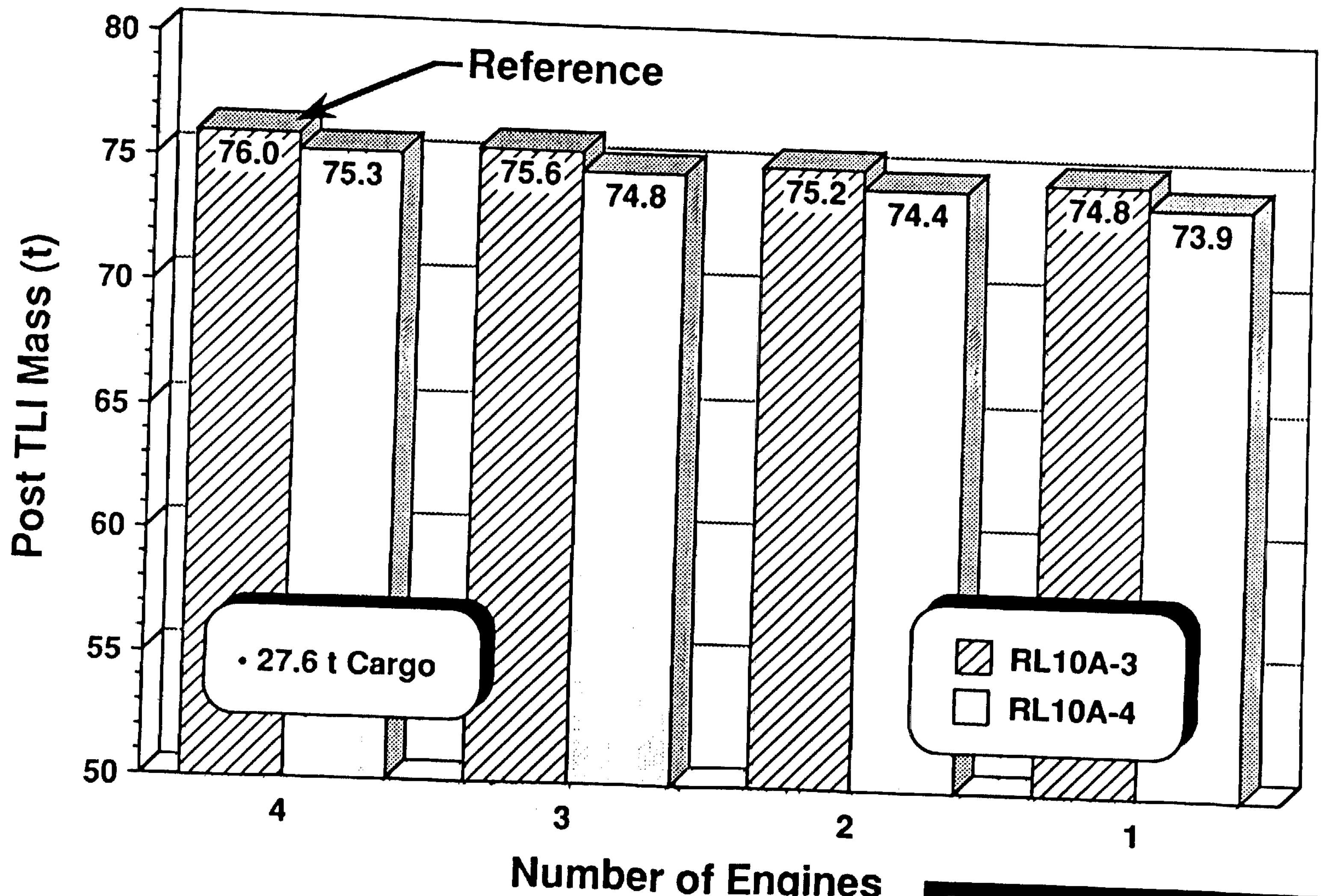
Performance Analysis

Cargo Engine Parametrics I

MSFC



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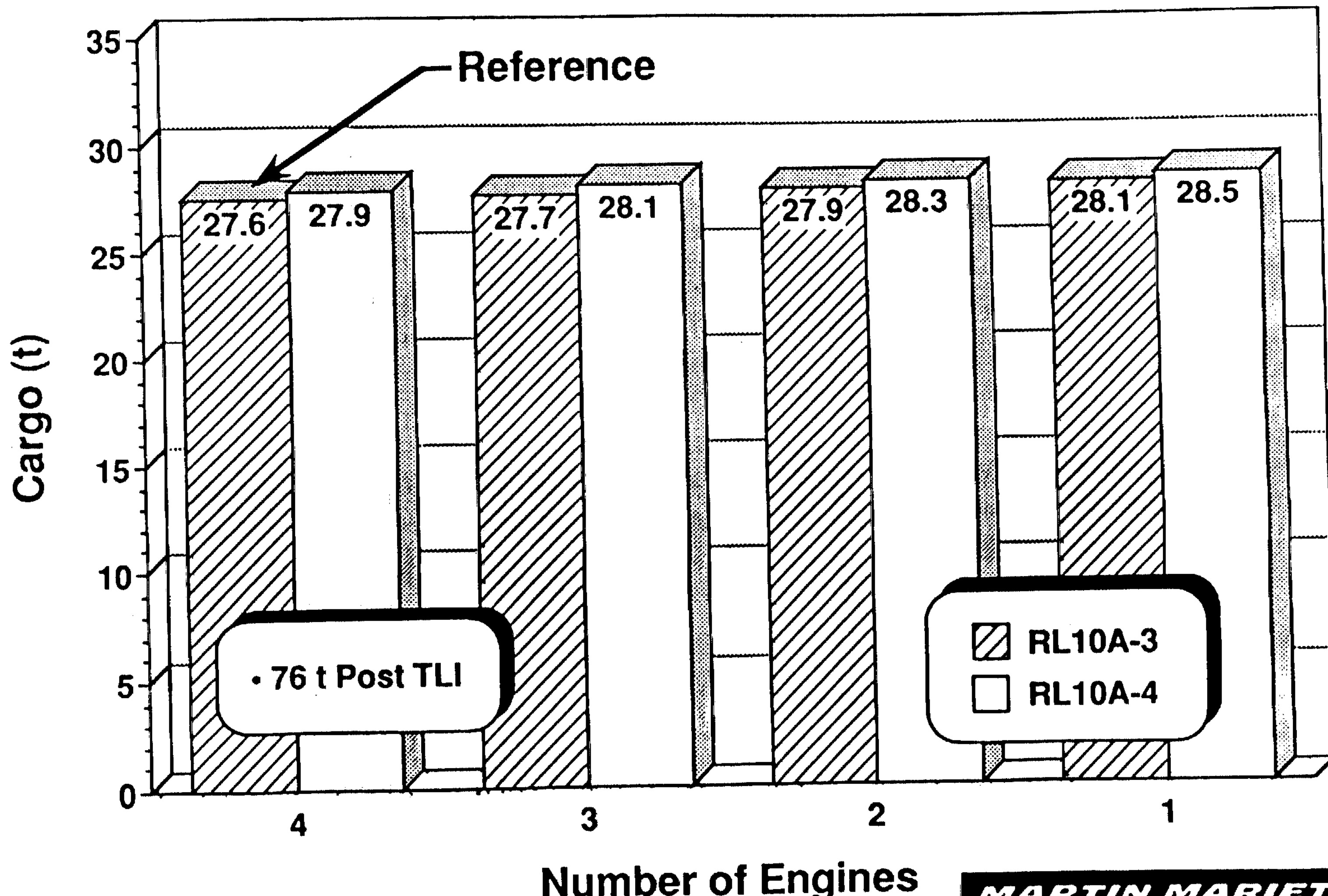
Performance Analysis

Cargo Engine Parametrics II

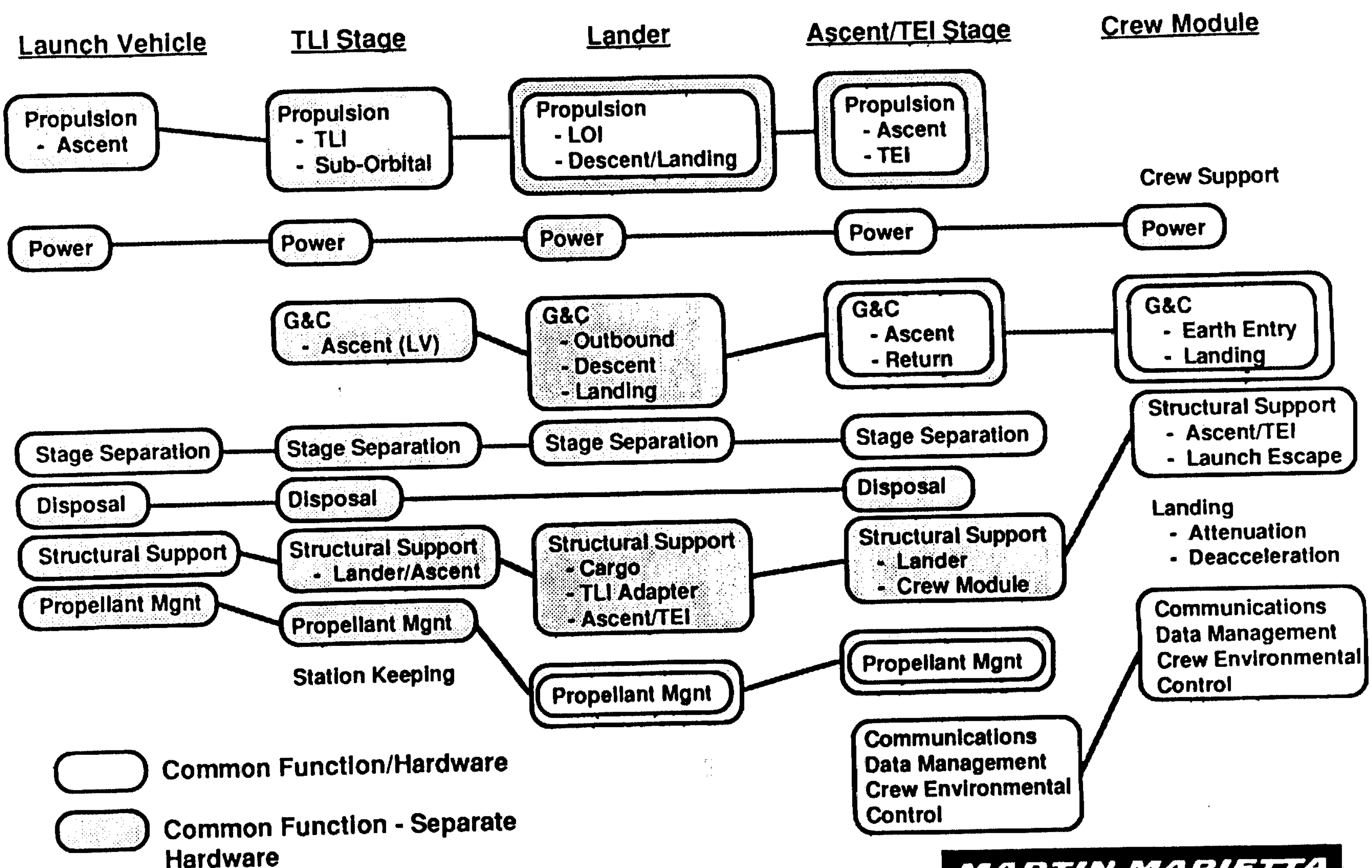


Agenda - 19 March

MSFC

- Groundrules/Assumptions — Sid Earley
- Configuration Selection
- Detailed Analysis
  - Performance
  - Cargo Handling
- Mission Functional Analysis — John Hodge
- Systems Risk Assessment
- Mission Abort Analysis

Mission Function Commonality Assessment



System Risk Assessment

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- Cryogenic Propellant Management (Ascent)

Risk	Mitigation
• Boiloff Of Critical Return Propellant - Lunar Day/Night/Day (7% H_2, 1.5% O_2 per month)	• Reduce Tanks' View To Direct Radiation - Center of Vehicle - Combination Vapor Cooled and Debris Shield • Separate Ascent and Descent Tanks - Heavier Insulation Possible on Ascent Tanks
• Pressure Build Up In Tanks - Frozen Vents - Large Temperature Increases On Tank Surfaces	• Backup Cryo Management Systems - Redundant Pressure Relief - Redundant Vapor Cooled Shield Tubing • Tankage Configuration Reduces Visibility to Heating Source
• Liquid Acquisition - Problem Similar With Storable - LH_2 & LO_2 Difficult To Handle	• Acquisition Devises Ensure Vapor-Free Liquid. - Tank Head Idle - Paramagnetic

System Risk Assessment

MSFC

- Single Propulsion System

Advantages

- Reduction In Engine Quantities
- Performance Gains with Higher Cryo Isp
- Reduction In Overall System Height
- Operational Confidence Due to LOI and Descent Burns.

Risks

- Disconnects With Lander Tankage Vulnerable to Leakage
- Increased Plumbing Complexities
- Higher Potential For Engine Damage During Initial Ascent
- Potential Damage Or Contamination During Final Descent
- Lander/Ascent Vehicle Clearance
  - Engine Gimbaling
  - Lander Deformation At Landing
  - Non-Vertical Ascent
- Release Failure of:
  - "Hold-Downs"
  - Fluid Disconnects
  - Electrical Disconnects
- System Restart Following Extended Surface Stay

System Risk Assessment

• RL-10A3 Cryogenic Engine

Risk	Concerns/Mitigation
• System Restart Following Extended Surface Stay - Lunar Day/Night/Day - Up to Three Starts Prior to Ascent - Temperature Differential Across Engines	• Longest Period Between RL-10 Burns Has Been 24 Hours (In-Space) - Titan/Centaur Operations: - Ten Minutes Between First and Second Burns - Several Hours Between Second and Third Burns - Tested To - 290°R With Successful Restart • Temperature Differentials Create Start Lags - Centaur Specification = 700 ms Δ - Colder Engine Slower To Start - Controllability Impacted - Thermal Control Proven (Passive/Active) - Thermal Control Systems - Centaur Roll Providing Uniform Heating (In-Flight Option Only)

Abort Analysis - Summary & Issues

- Abort Scenarios and Options Were Developed for Each Phase of the Mission:
  Pre-TLI to Lunar Landing to Earth Reentry

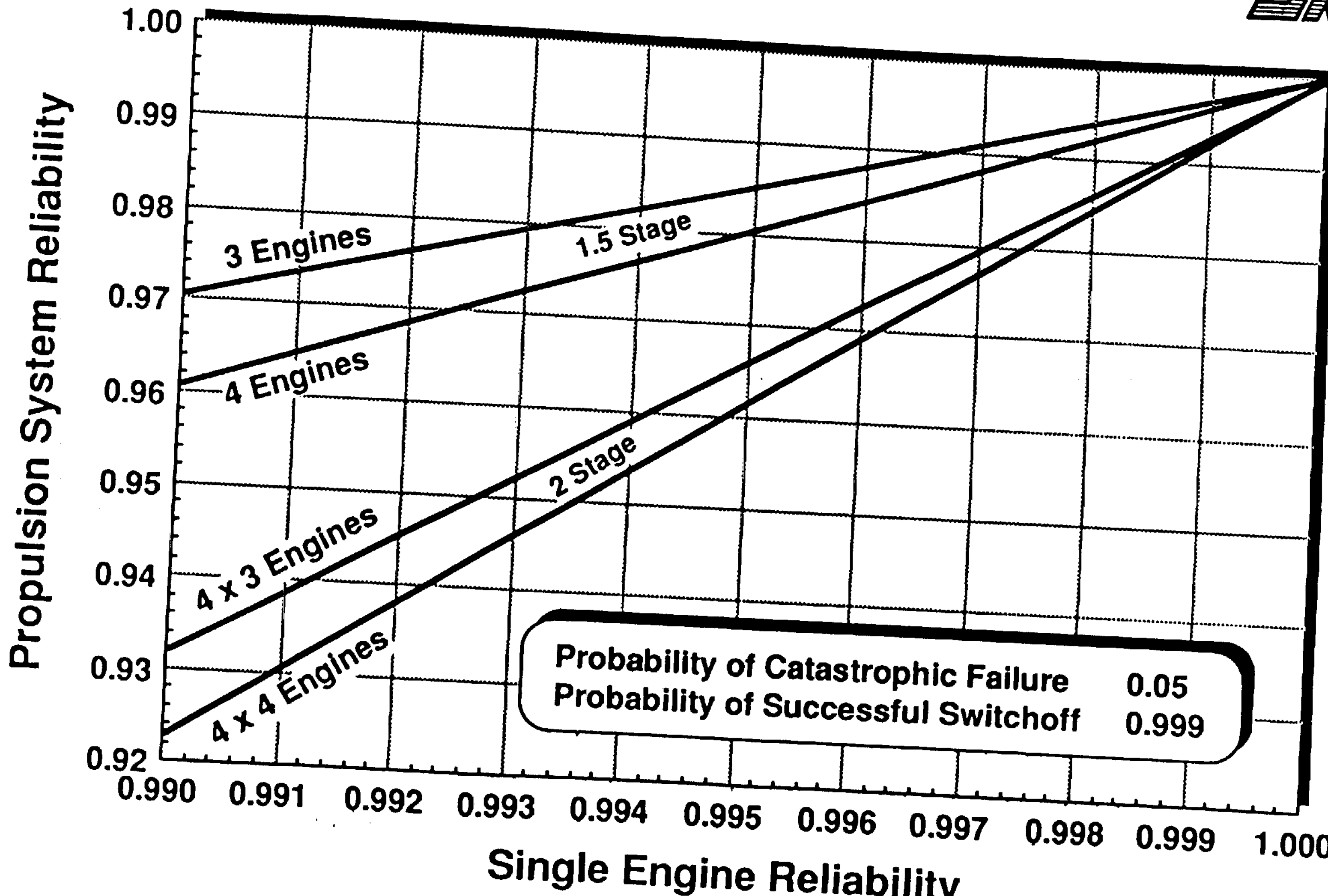
- Abort During Lunar Descent Is a Major Discriminator Between the 1.5 Stage and 2 Stage Systems in Regards to a Main Propulsion Failure
  - The 1.5 Stage Vehicle Has No Abort Option Available
  - The 2 Stage Vehicle Can Abort to LLO with the Ascent Stage
  - This Can Be Mitigated with Single Engine Out Capability

- The 1.5 Stage System Will Have a Lower Probability of a Propulsion Failure than the 2 Stage System

- Both Lander Options Cannot Tolerate an Engine Failure during the Lunar Ascent Phase of the Mission without Incorporating an Engine Out Capability
  - This Would Also Give the 1.5 Stage System Engine Out Capability During Descent on the Piloted and Cargo Missions

Propulsion System Reliability

No Engine Out



Propulsion System Reliability

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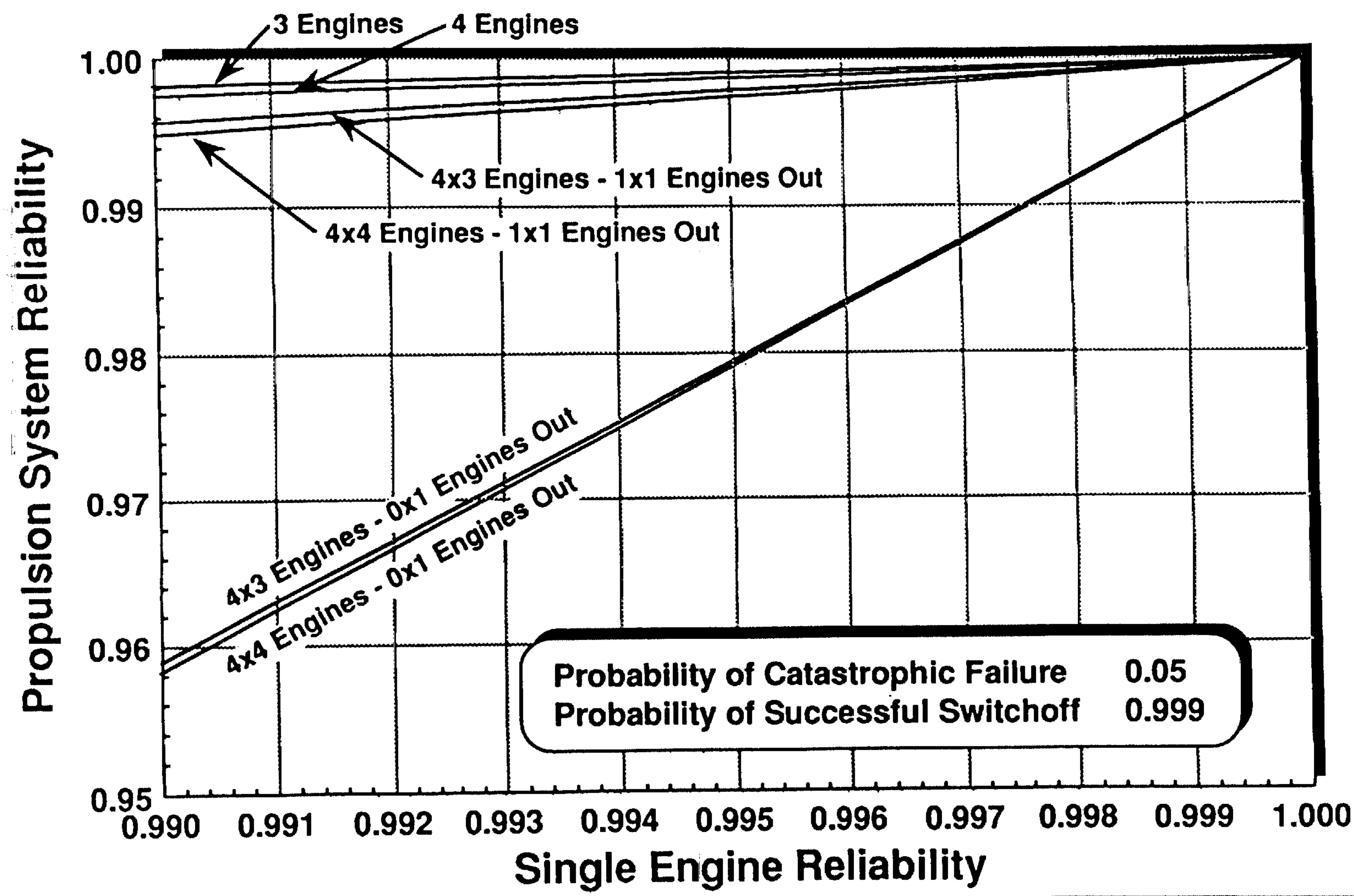
Single Engine Out



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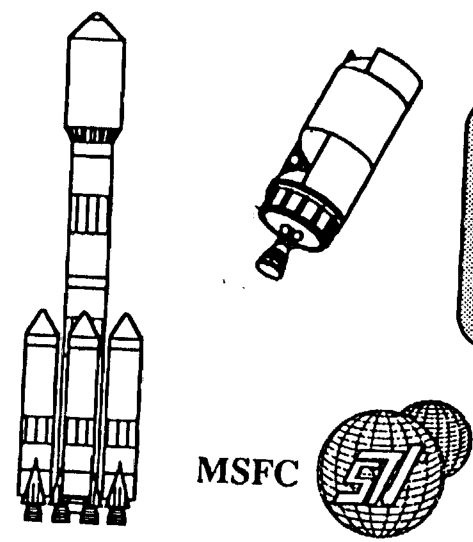
Summary

- Used a Consistent Set of Groundrules to Evaluate Configuration Options
- The Selected Reference Configuration Met All Original Performance Goals
  - 76 tonne Post TLI Mass
  - 5.0 tonne Piloted Cargo (Achieved 5.5 tonnes)
  - 27.5 tonne Cargo Only (Achieved 27.6 tonnes)
- The Selected Reference Configuration Can Meet the Current Cargo Baseline of 31 tonnes, Given a 93 tonne Post TLI Mass
- The CG Locations of the Reference Configuration Are Comparable to the Two Stage Cryogenic/Storable Baseline
- The Reference Configuration Provides Good Payload Stowage in the Piloted Mission & Its Fairing Nose Angle Is Sufficiently Steep
- Some of the Risks Associated with the Reference Configuration Have Been Recognized and the Solutions to Mitigate Them Have Been Identified

Technical Directive 12
Upper Stage Requirements and Concepts Study

Technical Directive 13
Phase II, Upper Stage Requirements and Concepts Study

Technical Directive 14
FLO TLI Study



TLI Stage Study Status

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Agenda

• Study Goals/Objectives	John Hodge
• TLI Stage Definition - Current Configuration Definitions - Key Mission Requirements/Groundrules/Assumptions - Issues/Concerns	Sid Earley
• Stage Functionality - Function/TLI Stage Identification/Allocation - TLI Stage Interfaces & Subsystem Relationships	John Cuseo
• Subsystems Definition - Structures/Tankage - Avioncis - VHM - Propulsion - RCS	Jim McKinnis
• Summary/Conclusions - Summary - Issues/Concerns	John Hodge

TLI Stage Study Status

TLI Stage Definition

Sidney M. Earley
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TLI Stage Study Plans

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SRR I (Sept 92)

Element Level Requirements
- Functional Analysis
- Analysis/Derivation
- Allocations

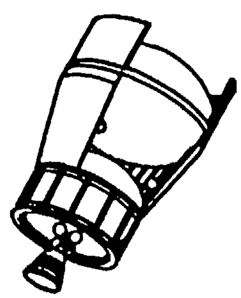
Element/Element Interface Relationships

Conceptual Stage Configurations

Subsystem Layouts w/Defined Functional Relationships

Support &Ops Concepts

Supporting Technology Plan



SRR II (Feb 93)

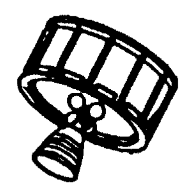
Subsystem Requirements
- Functional Analysis
- Analysis/Derivation
- Allocations

Element/Element IRD. Internal Interface Relationships

Detailed Stage Configurations
- Vibration/Stress
- Materials

Conceptual Subsystem Configurations w/Defined Component Relationships

Support & Ops Requirements



SDR (July 93)

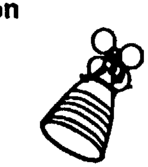
Component Requirements
- Functional Analysis
- Analysis/Derivation
- Allocations

Element/Element ICD. Internal IRD

Pre-Engineering Level Stage Configurations

Detailed Subsystem Configurations

Support & Ops Element Definition



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TLI Stage Study Overview

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STV Study Objectives

Define Space Transportation Elements Capable of Meeting NASA's and DOD's Near and Long Term Needs Beginning in 1999 and Continuing Through the Completion of the SEI Missions.

STV Special Studies Task #12

Support the Development of High Energy Upper Stage Systems, Capable of Meeting the Needs of a Changing Space Transportation Environment.

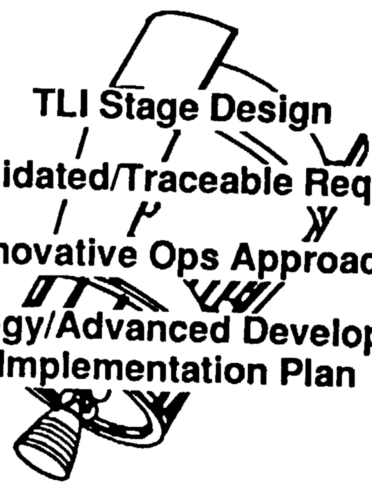
Goals - Civil/DoD

Goals - SEI/FLO

- Participant in First Lunar Outpost Study
  - Design
  - Requirements Definition/Analysis
- Define Aggressive Development Program
- Act as Integrator Between LV & Payload

Products

TLI Stage Design
Validated/Traceable Req'ts
Innovative Ops Approach
Technology/Advanced Development
Implementation Plan



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C-2

NLS Derived Heavy Lift Launch Vehicle

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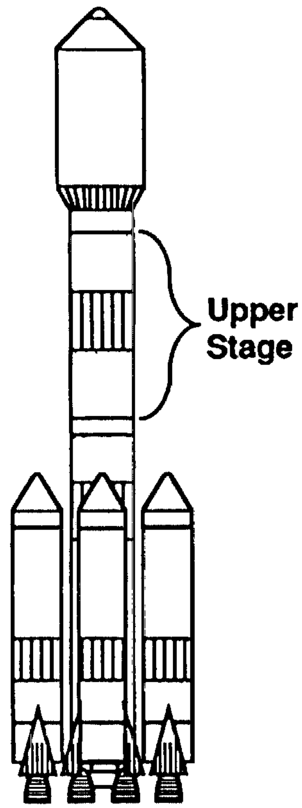

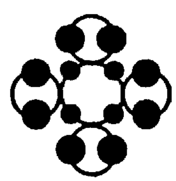

Characteristic	Booster	Core
Number	4	1
Inert Mass	75.5 t	88.8 t
Propellant Mass	998 t	766 t
Propellant Type	LOX/RP	LOX/LH2
Engine Type/#	F-1A/2	STME/4
Length	44.5 m	52.1 m
Diameter	6.74 m	8.41 m
# of Engines Out	0	0

Structure	Aluminum 2219
Usable Shroud Volume	10.0 x 12.5 m
Shroud Mass	9450 kg
Maximum G's	4.0
Maximum Q	43.0 kPa (900 psf)

Saturn V Derived Heavy Lift Launch Vehicle

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NG 2nd STAGE
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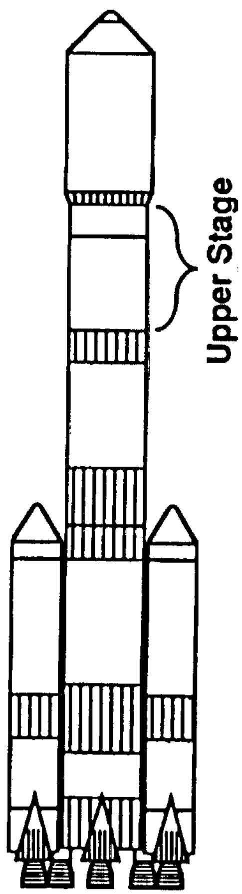
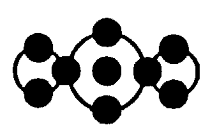


Characteristic	Booster	1st Stage	2nd Stage
Number	2	1	1
Inert Mass	75.7 t	209 t	60.8 t
Propellant Mass	998 t	2721 t	635 t
Propellant Type	LOX/RP	LOX/RP	LOX/LH2
Engine Type/#	F-1A/2	F-1A/5	J-2S/6
Length	~52 m	48.8 m	31.4 m
Diameter	6.7 m	10.0 m	10.0 m
# of Engines Out	0	0	0



Structure	Aluminum 2219
Usable Shroud Volume	10.0 x 18.3 m
Shroud Mass	12807 kg
Maximum G's	4.0
Maximum Q	43.0 kPa (900 psf)

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NLS HLLV Upper Stage Configuration

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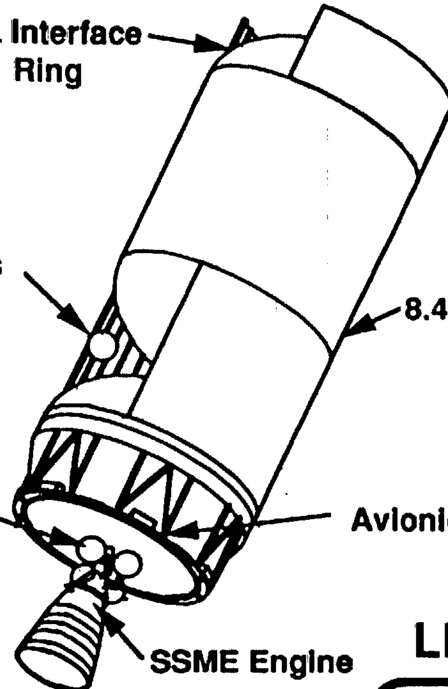
Stage Length = 24.9 m



TLI Stage Mass Prop.

Component	Mass (kg)
Stage Dry	21,250
Contingency (20%)	4,250
Dry Mass	25,500
Propellant	304,500
RCS Prop	590
Total Stage	330,590
Eff. Mass Fract.	0.909
Payload	95t

Total Dry Mass Delta 18148 kg

LEO Stage Mass Prop.

Component	Mass (kg)
Stage Dry	36,373
Contingency (20%)	7,275
Dry Mass	43,648
Propellant	304,500
RCS Prop	590
Total Stage	348,738
Eff. Mass Fract.	0.862
Payload	227t

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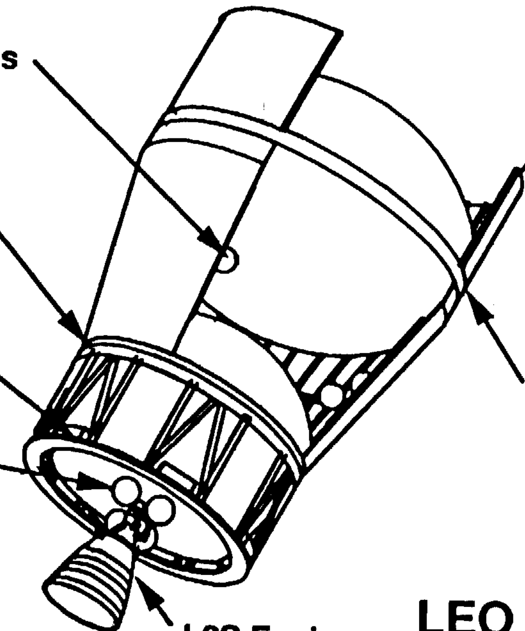
Saturn V HLLV Upper Stage Configuration

MSFC

Stage Length = 15.7 m



TLI Stage Mass Prop.

Component	Mass (kg)
Stage Dry	13,963
Contingency (20%)	2,793
Dry Mass	16,756
Propellant	137,025
RCS Prop	590
Total Stage	154,371
Eff. Mass Fract.	0.878
Payload	102t

Total Dry Mass Delta 2280 kg

LEO Stage Mass Prop.

Component	Mass (kg)
Stage Dry	15,863
Contingency (20%)	3,173
Dry Mass	19,036
Propellant	137,025
RCS Prop	590
Total Stage	156,651
Eff. Mass Fract.	0.865
Payload	245t

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RS920611-02B

Key Mission Requirements & Groundrules

MSFC

- MSFC Provided the Two HLLV Options (*NLS Derived and Saturn V Derived*)
- 93 tonne Post-TLI Payload Capability
- The TLI Stage Is a Free Standing, Load Carrying Structure
- Liquid Oxygen and Liquid Hydrogen Are the Propellants
- 20% Dry Mass Contingency on the Upper Stage (*FLORG = 10%*)
- The Upper Stage Uses Existing Hardware Where Possible
- A Single Avionics Suite Shall Be Capable of Performing All DRMs and Provide Guidance and Control to the Heavy Lift Launch Vehicle (HLLV)
- 3 Hour Mission Time (*From Lift-Off to TLI Stage/Lander Separation*) → GROUP CONCURRANCE

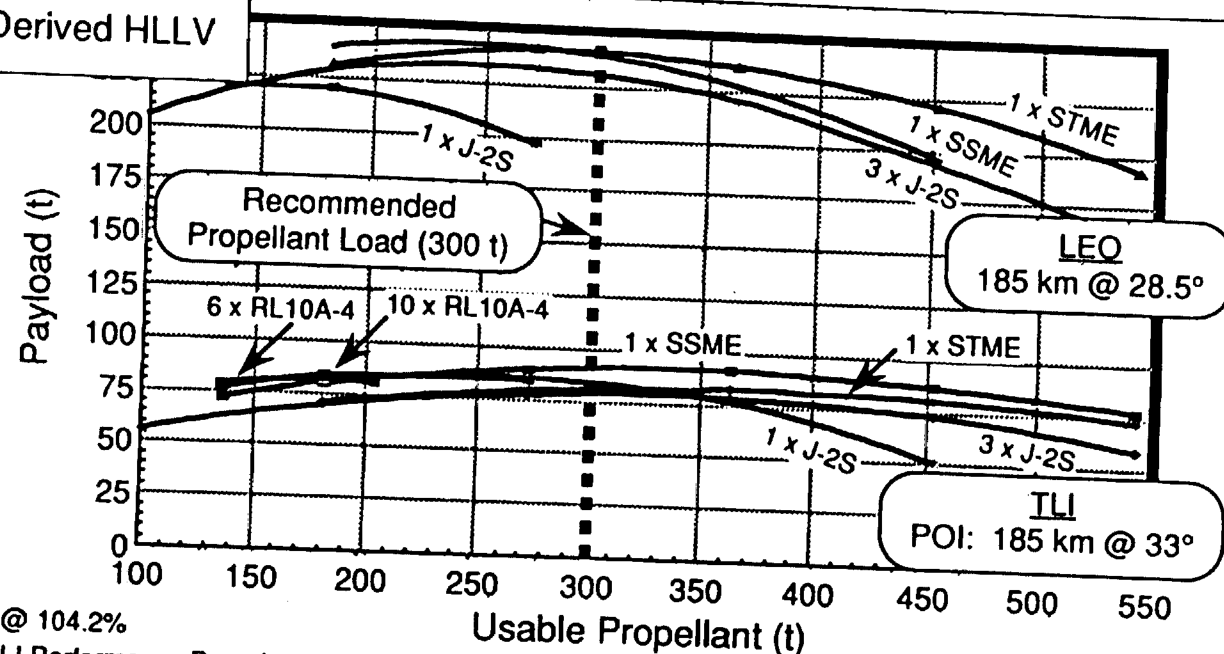
Contradiction in the Detailed Assumptions Occurs Between Operations (2 Orbits) and HLLV (8 hours) Sections

Requirements Impacts and Influences

MSFC

93 tonne Post-TLI Payload Capability

NLS Derived HLLV



Notes:
- SSME's @ 104.2%
- LEO & TLI Performance Based on Different MF's

Launch Vehicle & TLI Stage	TLI Capability (tonnes)
NLS Derived	95
Saturn Derived	102

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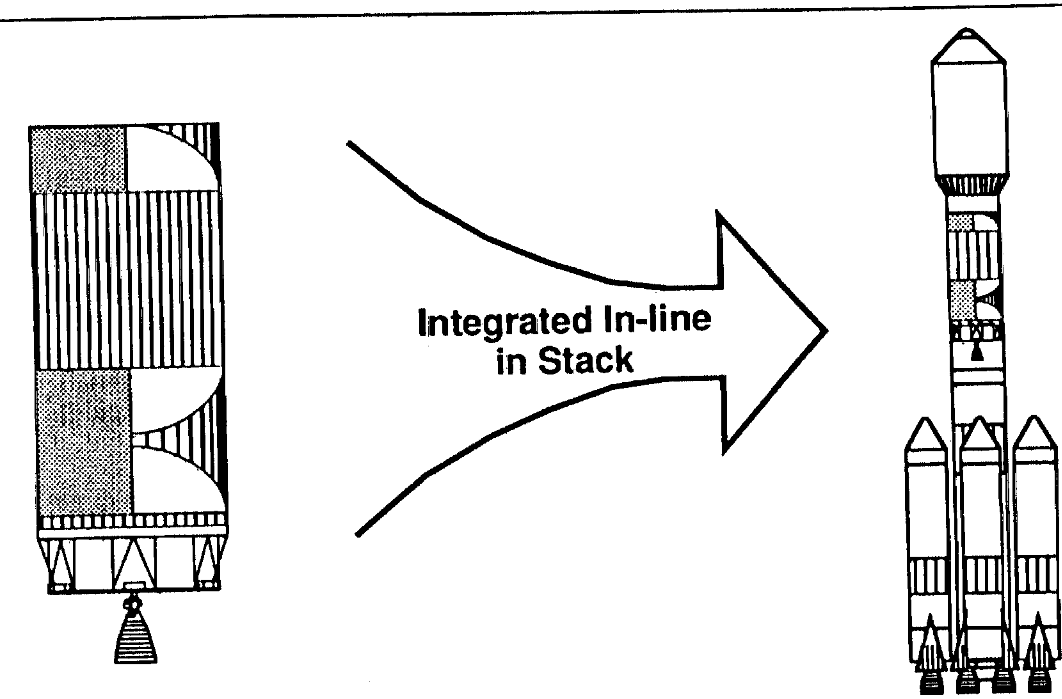
Requirements Impacts and Influences

The TLI Stage is a Free Standing, Load Carrying Structure



- The TLI Stage For Both HLLV's (NLS & Sat V) is designed as a Self Supporting Structure That Will Accommodate Launch Loads as an In-line Segment of The Launch Vehicle / Payload System

Requirements Impacts and Influences

MSFC

Liquid Oxygen and Liquid Hydrogen Are the Propellants

Engine Option	Type	Thrust (klbs)	Isp (sec)	Potential Upper Stage Size
RL10A-3	Cryo	16.5	444.4	S, M, L
RL10A-4		20.8	448 - 452	S, M, L
RL10B-2		22	456 - 468	S, M, L
RS-44		16 - 20	~480	S, M, L
IME		20 - 200	465 - 475	S, M, L
J-2S		265	436	L
SSME		470 (100%)	452.9	L
STME		650	428.5	L
NERVA Derived	Nuclear	25 - 75	870 - 925	L
Particle Bed		20 - 200	900 - 1000	S, M, L
Thermionic		0.2 - 1	~850	S, M
AJ10-118	Storable	9.6	319	S
OMS		6	320	S
XLR-132		3.7 - 15	340 - 347	S

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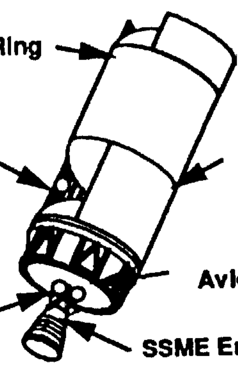
Requirements Impacts and Influences

20% Dry Mass Contingency on the Upper Stage (FLORG = 10%)

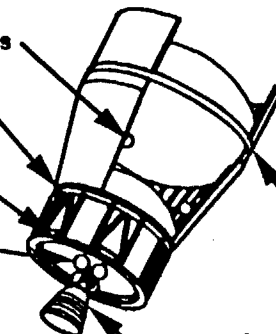
Stage Length = 24.9 m



TLI Stage Mass Prop.

Component	Mass (kg)
Stage Dry	21,250
Contingency (20%)	4,250
Dry Mass	25,500
Propellant	304,500
RCS Prop	590
Total Stage	330,590
Eff. Mass Fract.	0.909
Payload	95,000

Stage Length = 15.7 m



TLI Stage Mass Prop.

Component	Mass (kg)
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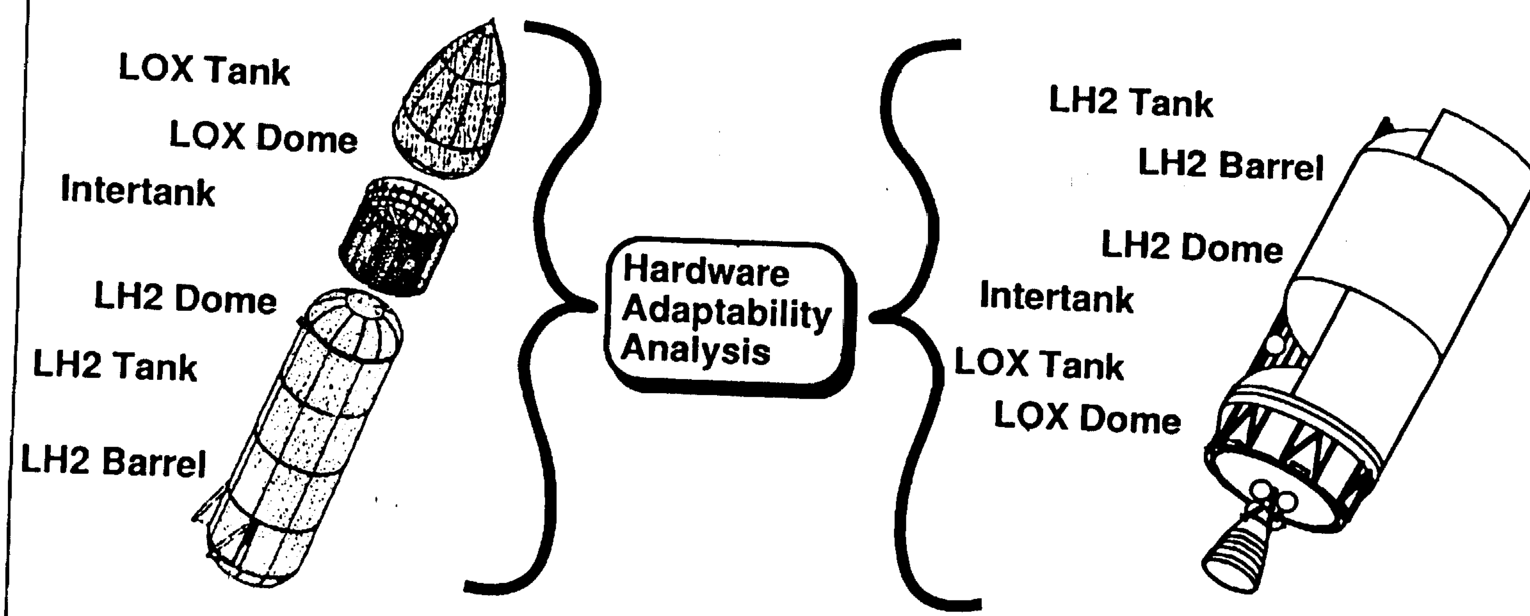
Contingency (20%)	2,793
Dry Mass	16,756
Propellant	137,025
RCS Prop	590
Total Stage	154,371
Eff. Mass Fract.	0.878
Payload	102,000

Requirements Impacts and Influences

MSFC

The Upper Stage Uses Existing Hardware Where Possible



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Requirements Impacts and Influences

A Single Avionics Suite Shall Be Capable of Performing All DRMs and Provide Guidance and Control to the HLLV

Mission	Avionics Mass Penalty (kg)
Space Station Freedom (*Driver*)	0
Low Earth Orbit	106
Sun-Synchronous Orbit (*from ETR*)	53
Molniya (12 hr.) Orbit	69
Geostationary Orbit	71
Trans-Lunar Injection	65
Trans-Mars Injection	65
Interplanetary Injection	65

Represents ~0.06% Post-TLI Payload Penalty

Requirements Impacts and Influences

3 Hour Mission Time
(from Lift-Off to TLI Stage/Lander Separation)

3 Hour Trans-Lunar Mission

			Boiloff (%)		
	LH2 Tank	LOX Tank	LH2	LOX	Overall
Insulation	SOFI	Bare Aluminum	9.7	2.3	3.4
	SOFI + White Coating	White Coating Only	7.1	1.6	2.5
	SOFI + 5 Layers of MLI	5 Layers of MLI	1.2	0.3	0.3

8 Hour Trans-Lunar Mission

			Boiloff (%)		
	LH2 Tank	LOX Tank	LH2	LOX	Overall
Insulation	SOFI	Bare Aluminum	25.8	4.0	7.4
	SOFI + White Coating	White Coating Only	18.9	4.4	6.6
	SOFI + 5 Layers of MLI	5 Layers of MLI	3.2	0.8	0.8

TLI Stage Summary and Issues

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- Both of Our Upper Stage Reference Designs Meet the 93 tonne FLO TLI Requirement
  - NLS Derived HLLV Upper Stage = 95 tonnes
  - Saturn V Derived HLLV Upper Stage = 102 tonnes

- RL10's Were Considered for Upper Stage Main Propulsion
  - *Did Not Meet* the 93 tonne FLO TLI Requirement from the NLS Derived HLLV
  - *Did Meet* the 93 tonne FLO TLI Requirement from the Saturn V Derived HLLV
  - Relatively Large Number of Engines Are Required (at least 5) for TLI
  - Can Provide Commonality with the Lander Element, Especially in a P/A Module Approach
  - Not an Attractive Propulsion Option for LEO Missions on Either HLLV

- Post-TLI Payload Requirement Could Grow from 93 tonnes, Posing a Potential Problem for the NLS Derived HLLV

TLI Stage Study Status

Stage Functionality

John Cuseo
(303) 971-7896
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TLI Stage Function/Requirements Definition

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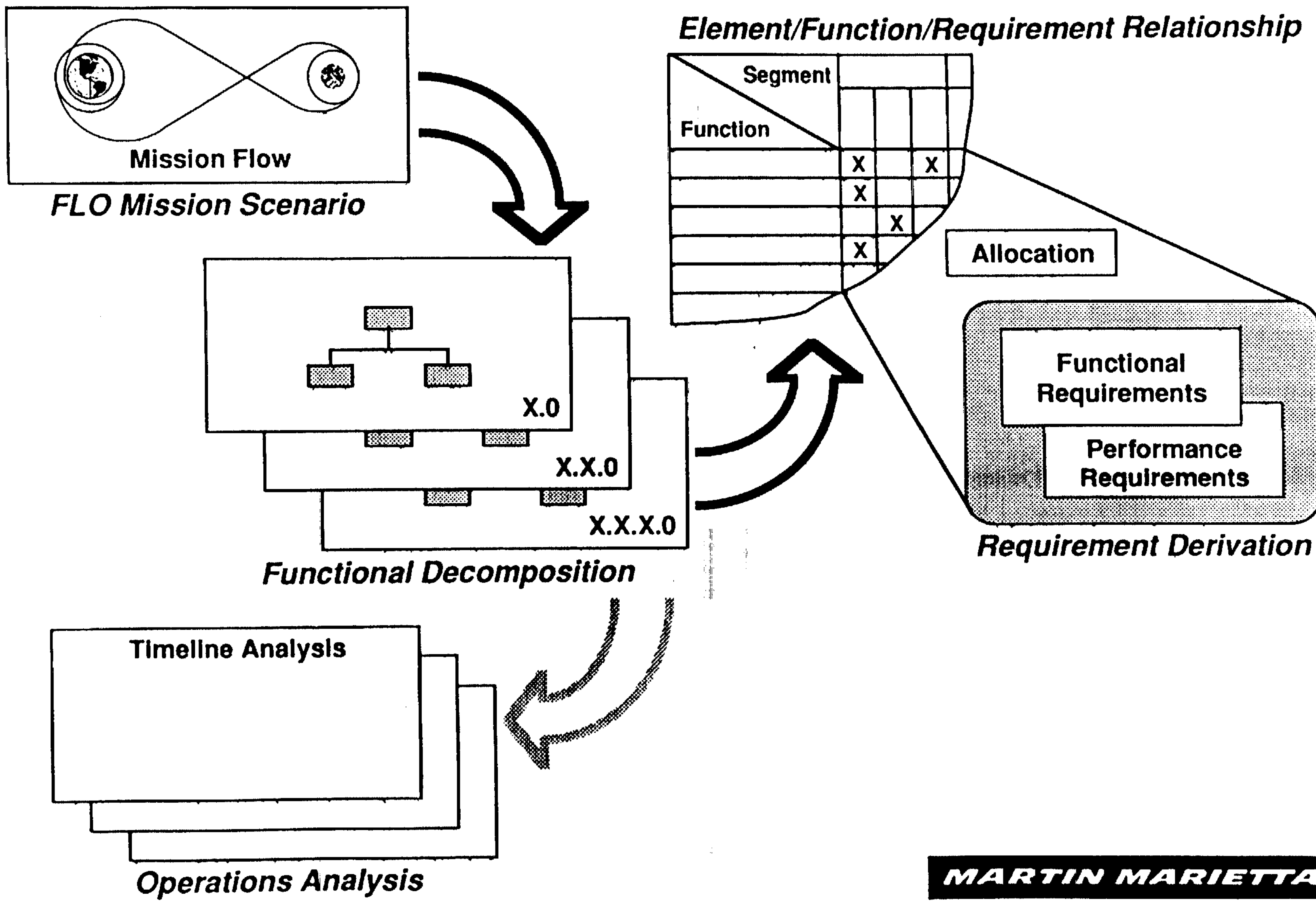

FLO TLI Stage Functional Decomposition

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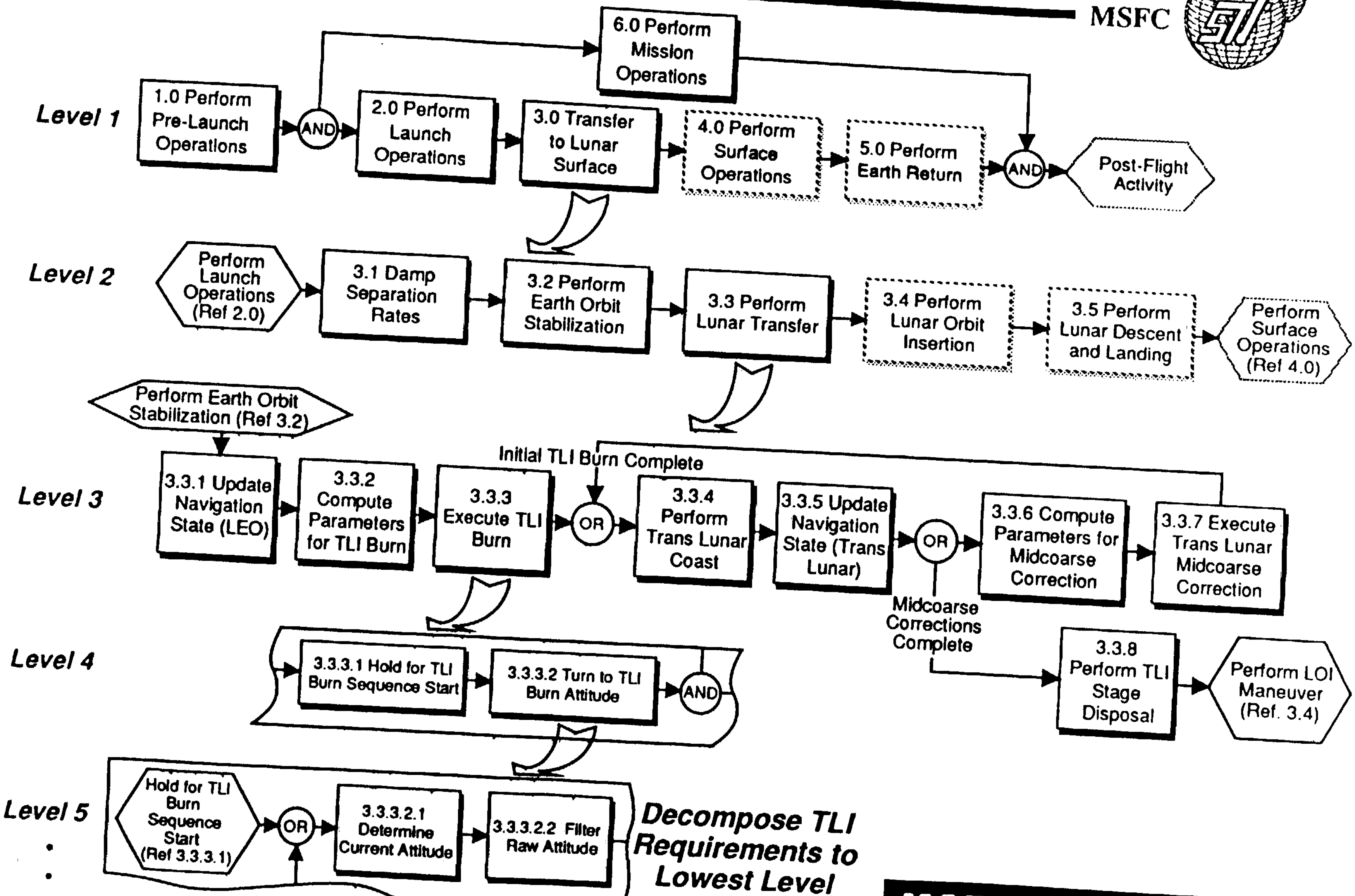

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Element/Function/Req't Relationship

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Segment/Element Allocation / Functions	ETS			Space Tran			
	HLLV	Shroud	TLI Stage	Launch Esc	Lander	Crew Module	Log Carrier
⋮							
3.3.3.1 Hold for TLI Sequence Start			√		√	√	√
3.3.3.1.1 Maintain Pre-TLI Attitude			√				
3.3.3.1.2 Verify PreTLI Health/Status			√		√	√	√
3.3.3.1.3 Refine Burn Duration Calc			√				

FLORG Reference

Paragraph: 5.2.1.2.1 TLI Stage Element
Number: 418

The TLI stage element shall provide the capability for attitude correction prior to TLI burn. (B. Pattison 03/03/92)

Performance Analysis and Requirements

Function Number: 3.3.3.1.1
Function Name: Maintain Pre-TLI Attitude

<u>Performance Requirements</u>

3.3.3.1.1.a Attitude Accuracy Prior to TLI Burn
Responsibility: S. Earley, MMAG

3.3.3.1.1.b Rotational Acceleration (control authority) Required for Pre-TLI Attitude Control
Responsibility: J. Cuseo, MMAG

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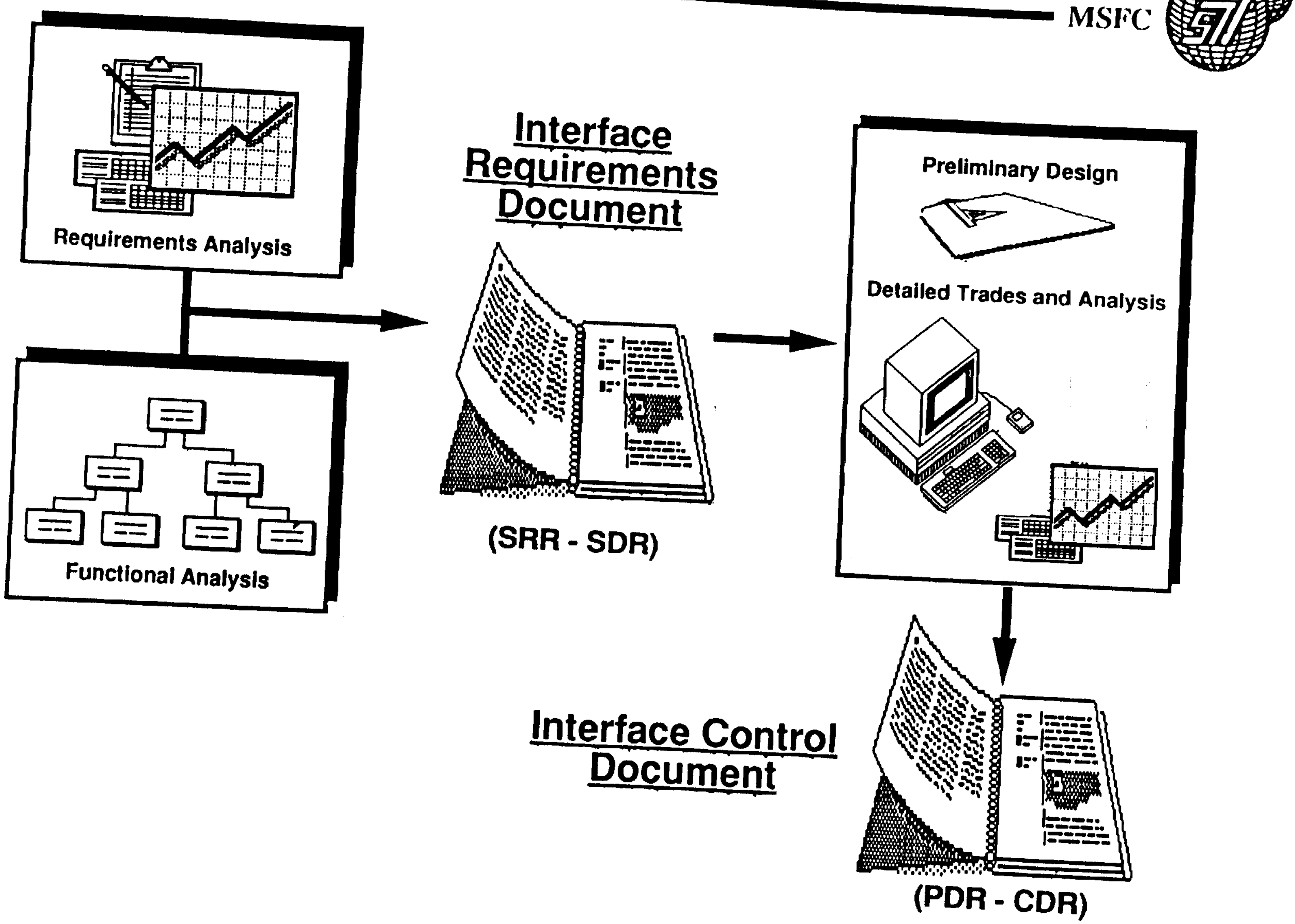
STV/TLI Interface Analysis Flow



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Identify External Interface Elements & Types

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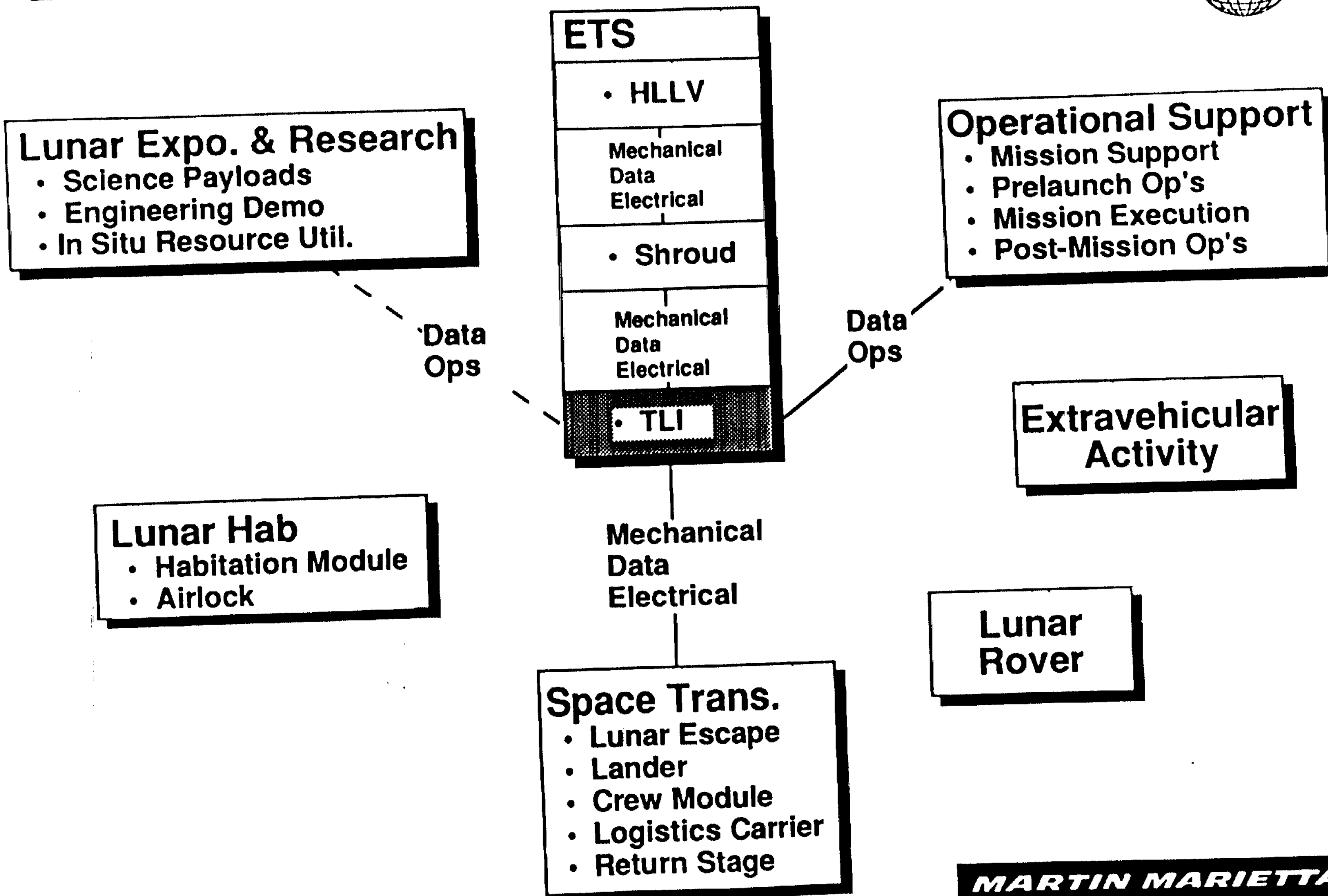

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Function\Segment\Interface Traceability

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Function	Allocation \ Elements	HLLV	Shrd	TLI St	L Esc	Lndr	Crw M	Log C
2.1 Perform Boost Phase	HLLV	–	NM	NMDE	N	NP	N	N
	Shrd	NM	–	MD		P	P	P
	TLI St	NMDE	MD	–		NM	N	N
	L Esc	N			–		NM	
	Crw M	N		N	NM	NM	–	NE
	Msn E	D	D	D	D	D	DVH	
2.2 Perform Booster Sep	HLLV							

FLORG Reference

Paragraph: 5.2.1.2.1 TLI Stage Element
Number: 417

The TLI stage element shall provide the capability for ascent guidance and control of the launch vehicle during launch from Earth. (B. Pattison 03/03/92)

Performance Analysis at the TLI/HLLV Interface

TLI Stage Study Status

Subsystems Definition

Jim McKinnis
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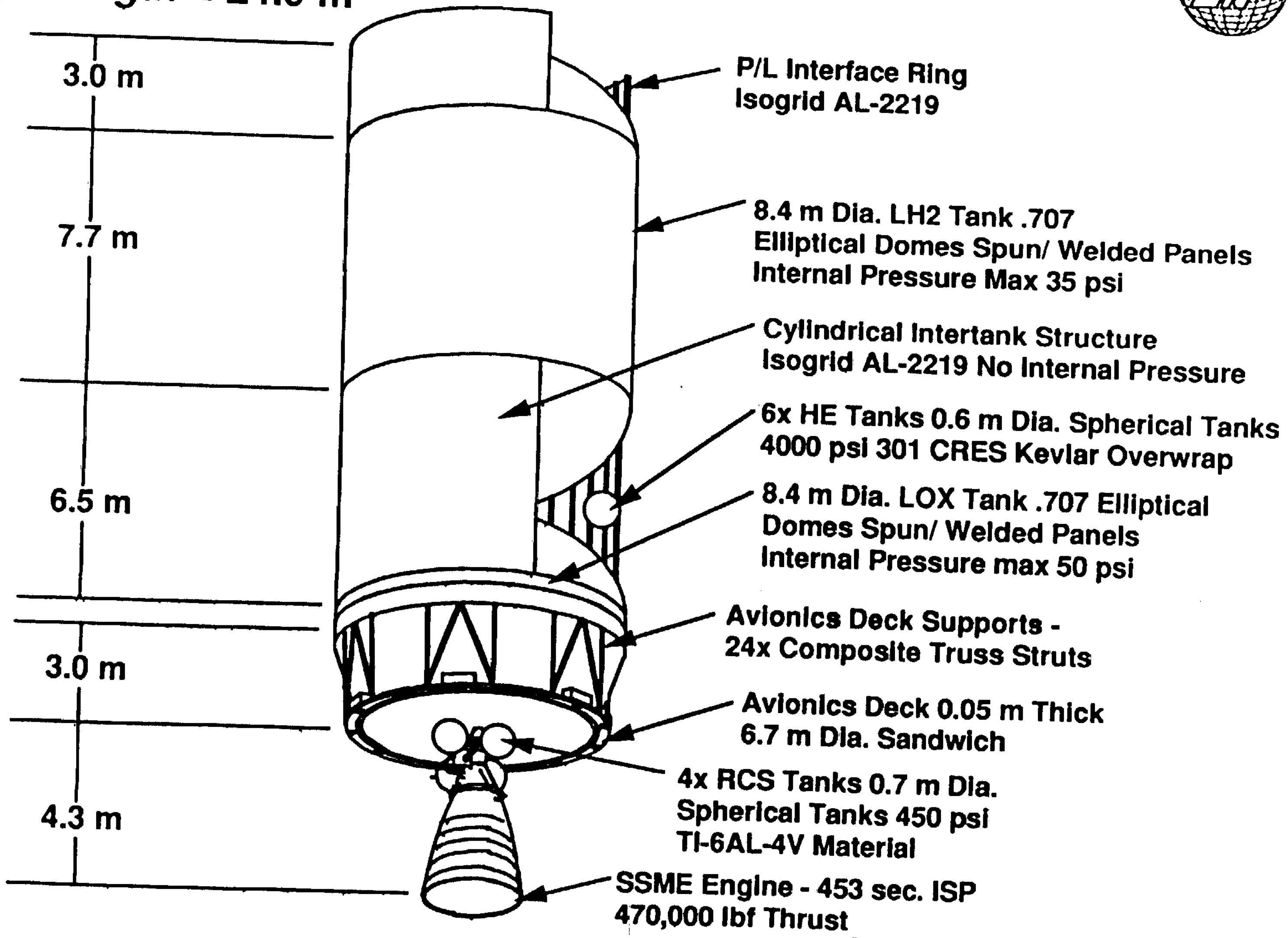
NLS HLLV Upper Stage Configuration

Stage Length = 24.9 m



Saturn V HLLV Upper Stage Configuration

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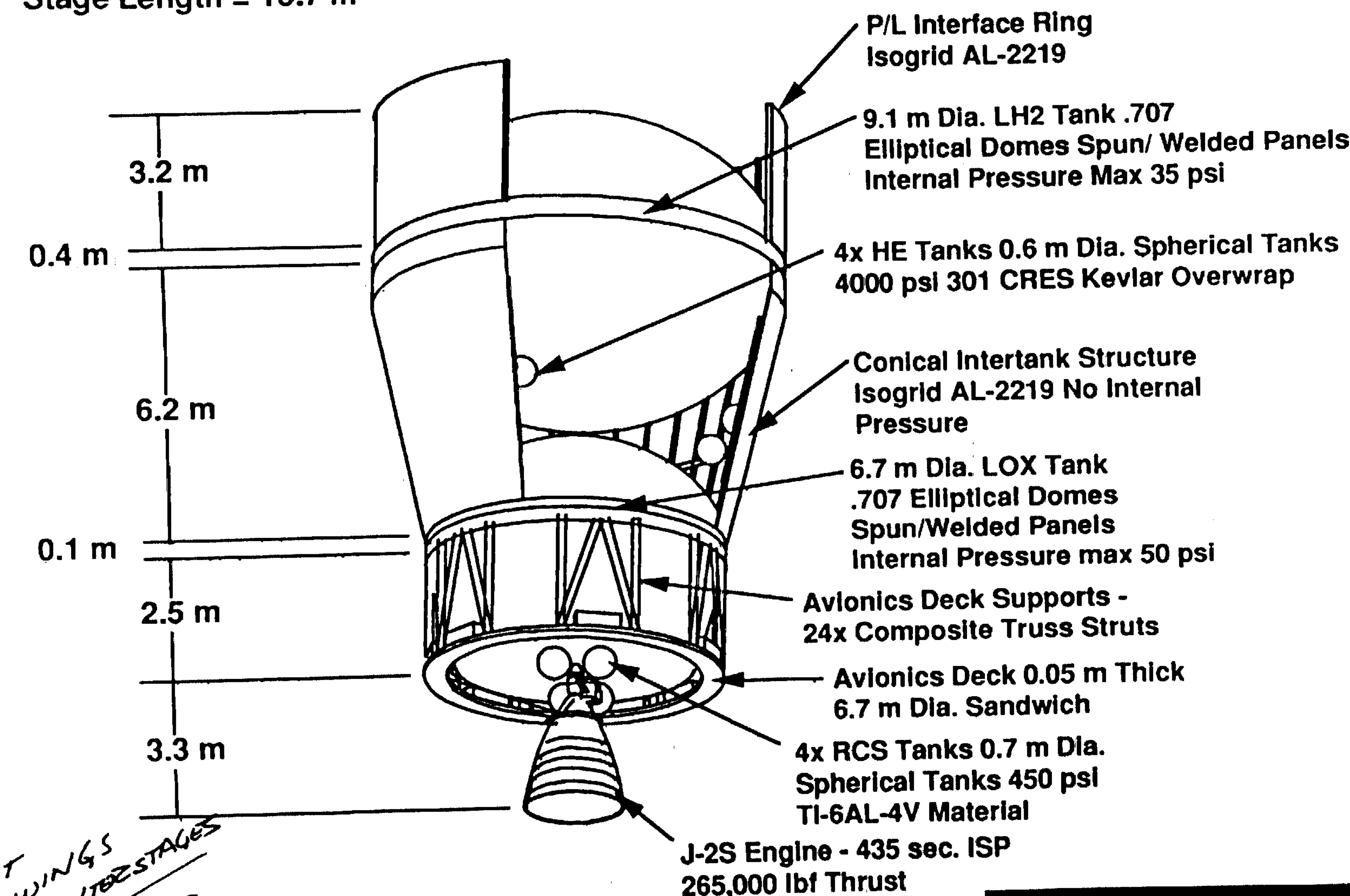
Stage Length = 15.7 m



NEED LAYOUT DRAWINGS OF INTERSTAGES

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TLI Stage / External Tank Structure Analysis

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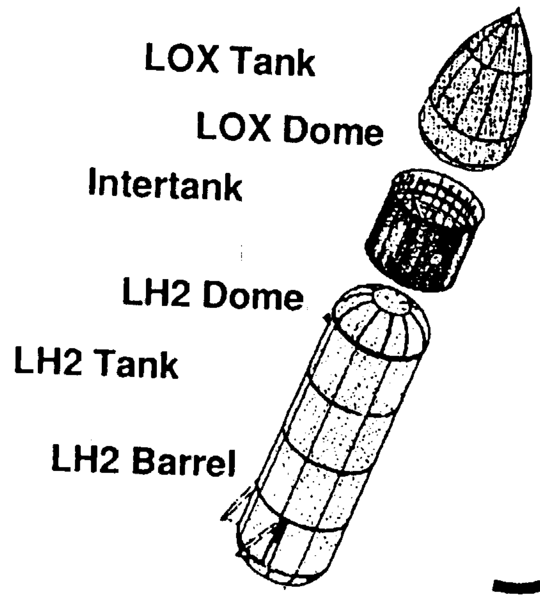
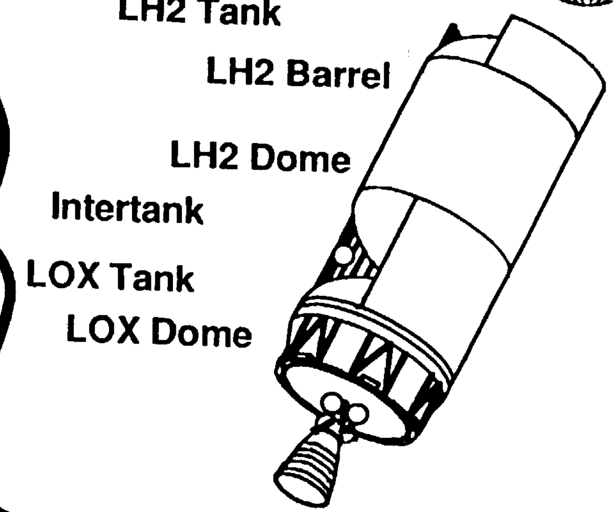


Hardware Adaptability Analysis



External Tank "ET" Hardware Integration

- Comparative Evaluation of ET Hardware Elements and Their Structural Impacts as TLI Baseline Hardware
- Assessment of Impacts to Current ET Manufacturing Process and Schedule
- Determine What Minor Modifications (if any) Need to be Implemented That Would Allow Usage or Improve Usage of Existing ET Hardware
- Derive Cost Impacts Associated with Utilization of ET Hardware and Tooling

Flat Hierarchy Bus Architecture

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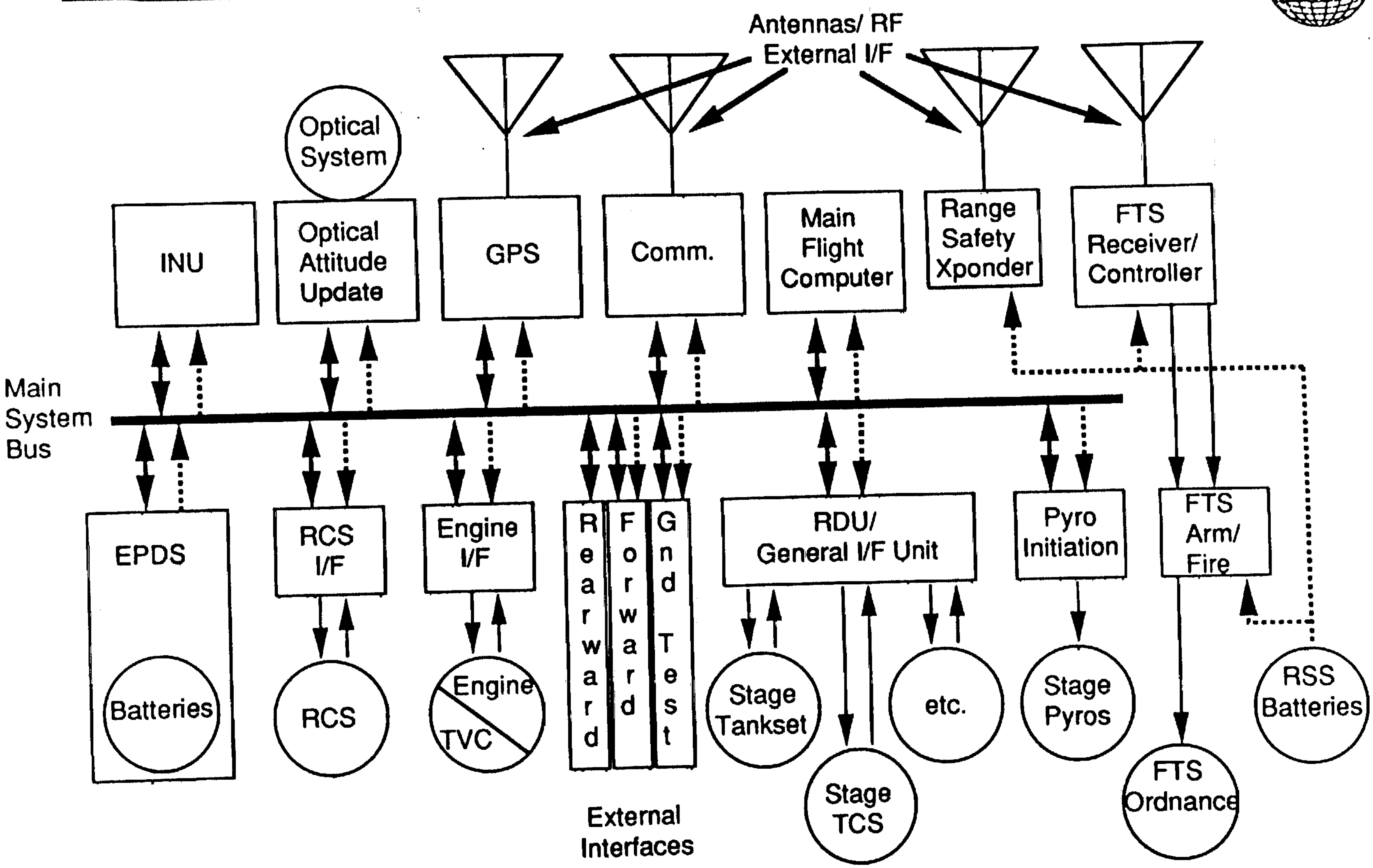

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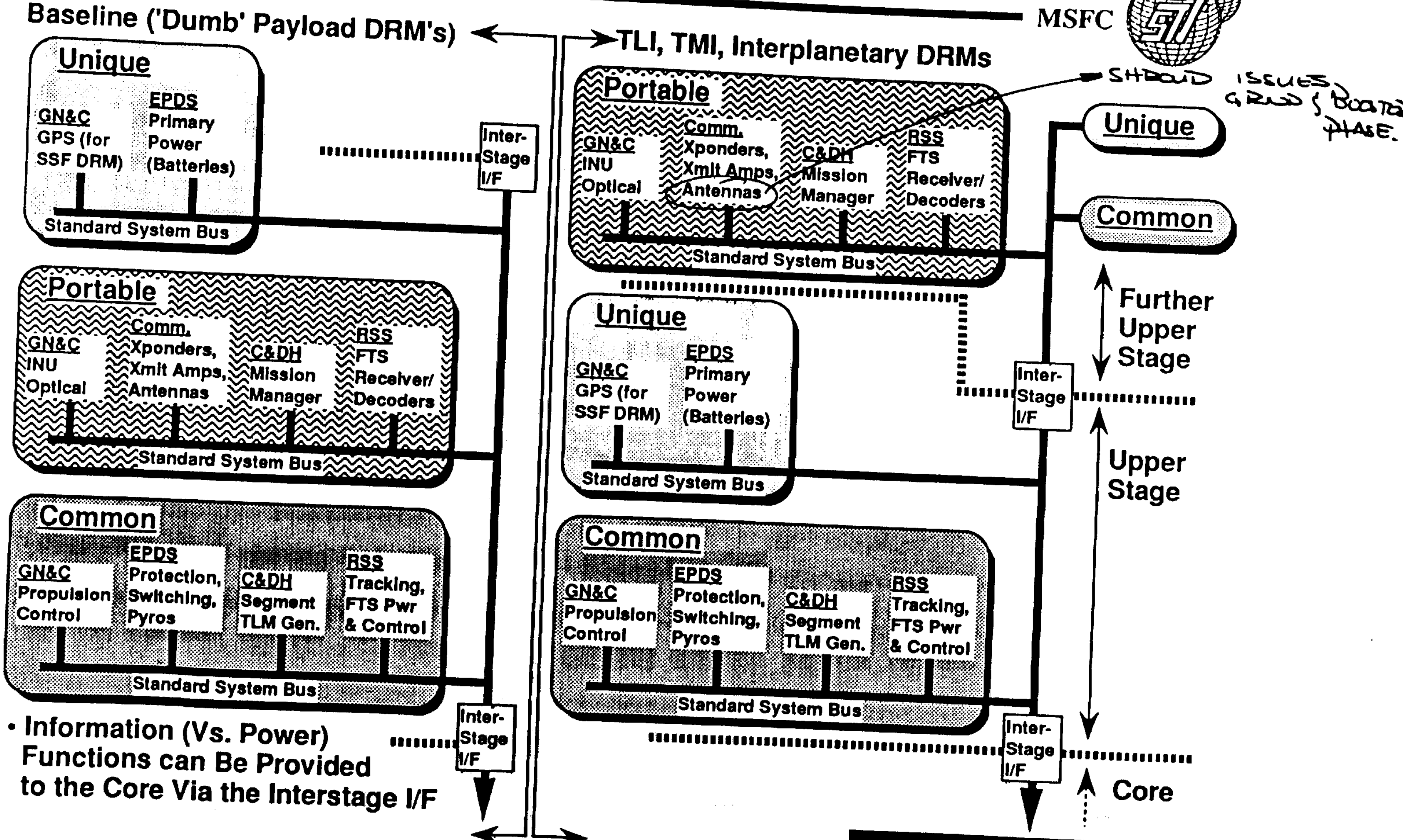
TLI Stage Avionics Configuration



• Information (Vs. Power) Functions can Be Provided to the Core Via the Interstage I/F

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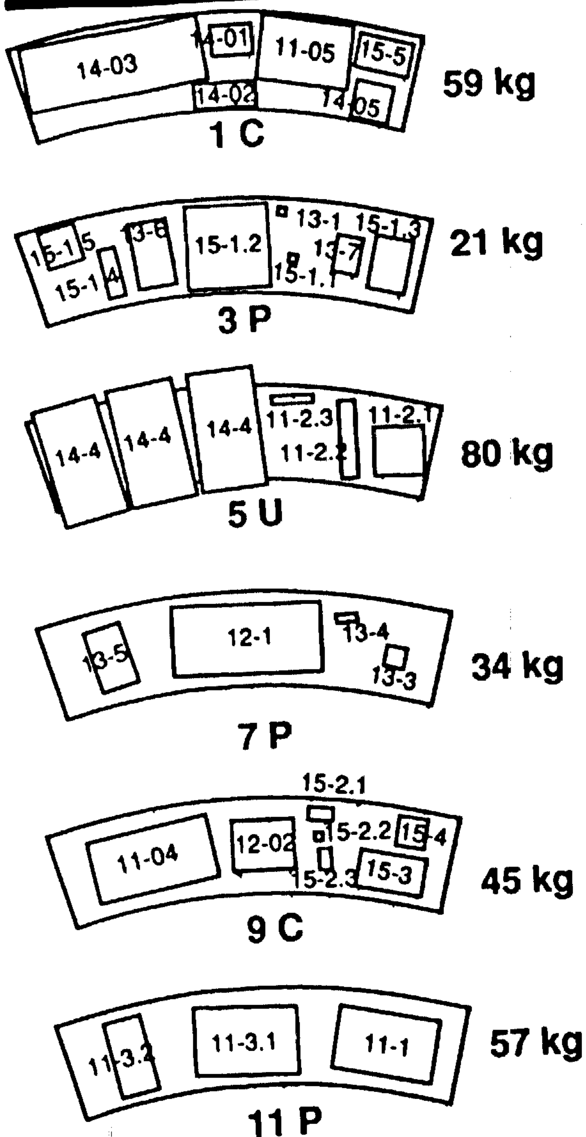
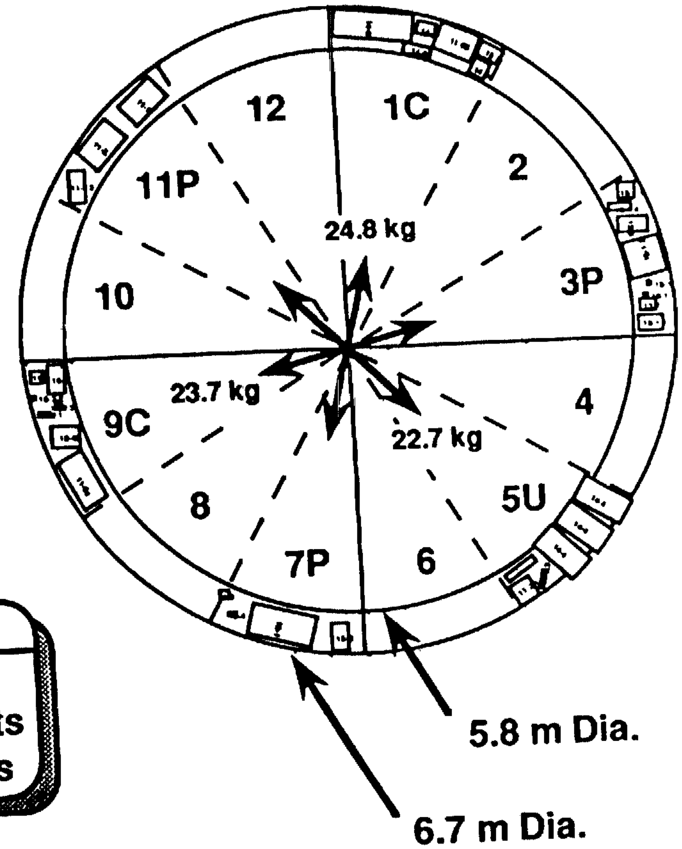
SE920805-09A

TLI Stage Avionics Deck Layout

MSFC



SUB-SYS	CODE
GN&C	11-xx
Mission Manager	12-xx
Comm.	13-xx
Power	14-xx
RSS	15-xx



Designation
Unique Elements
Common Elements
Portable Elements

Even # Segments Are Variable Support Structure

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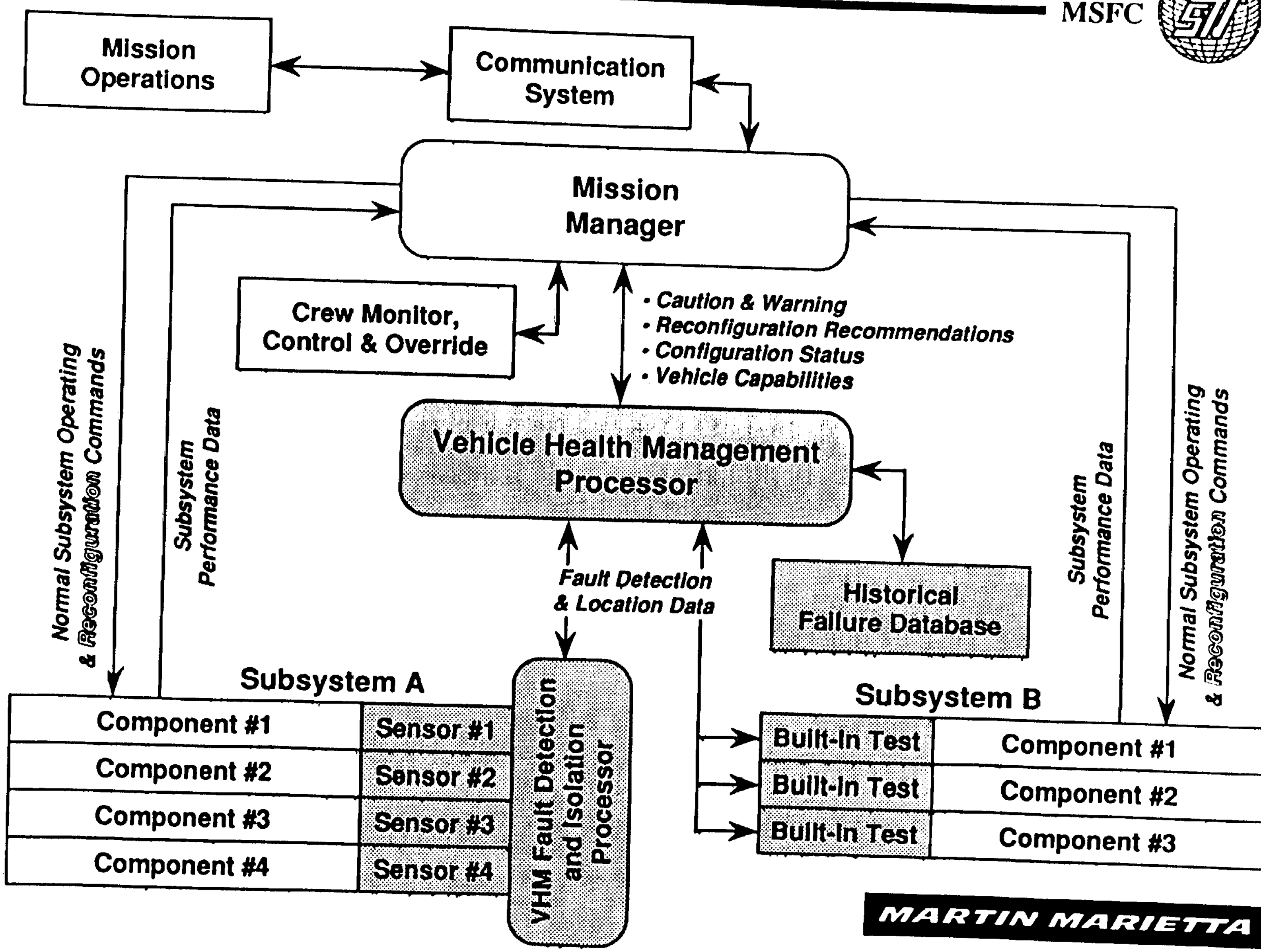
RS920618-01B

VHM Architecture Example

MSFC



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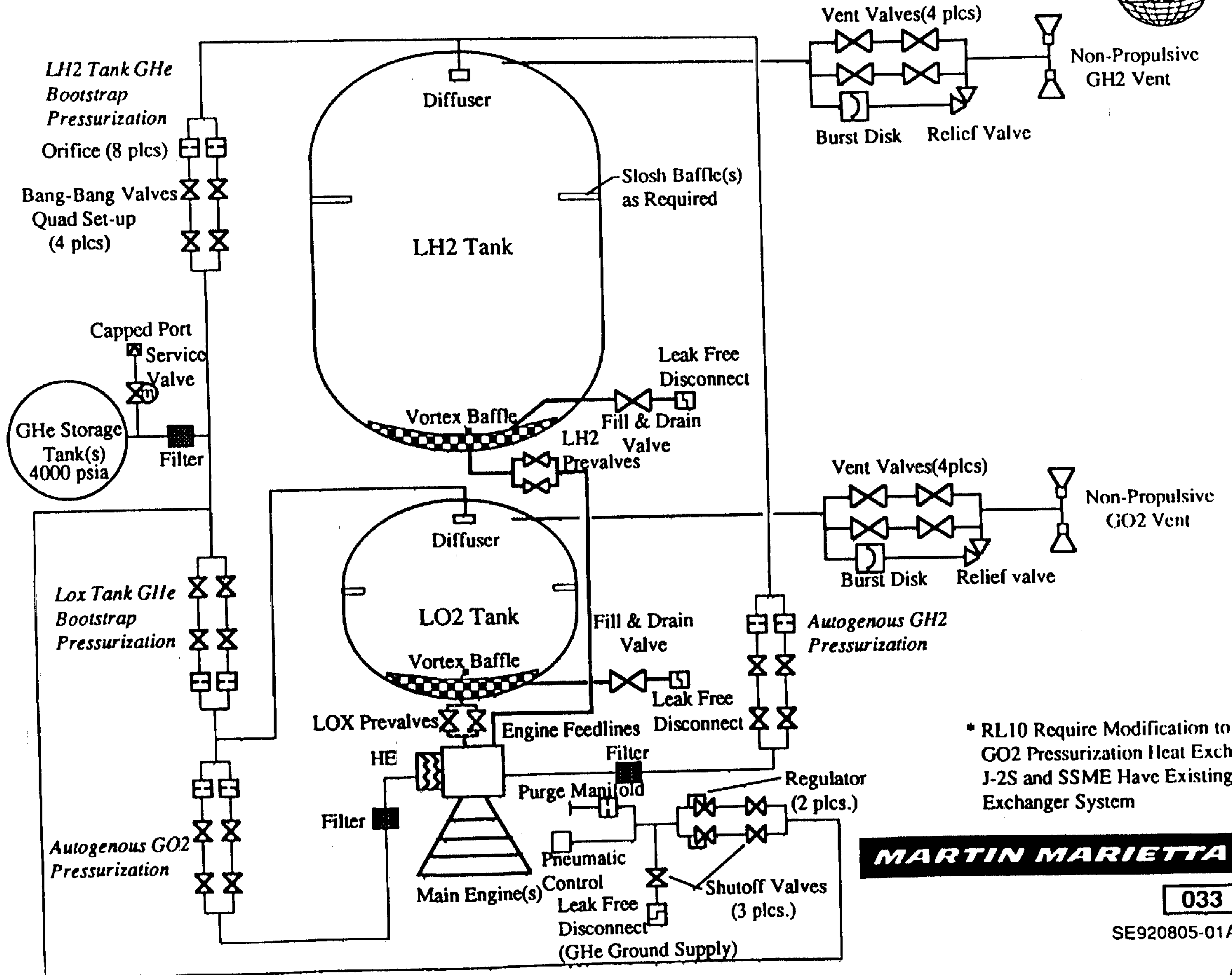
SE920806-03A

Main Propulsion Schematic

MSFC



* RL10 Require Modification to Incorporate GO2 Pressurization Heat Exchanger. Both J-2S and SSME Have Existing Heat Exchanger System

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SE920805-01A

Reaction Control System Schematic

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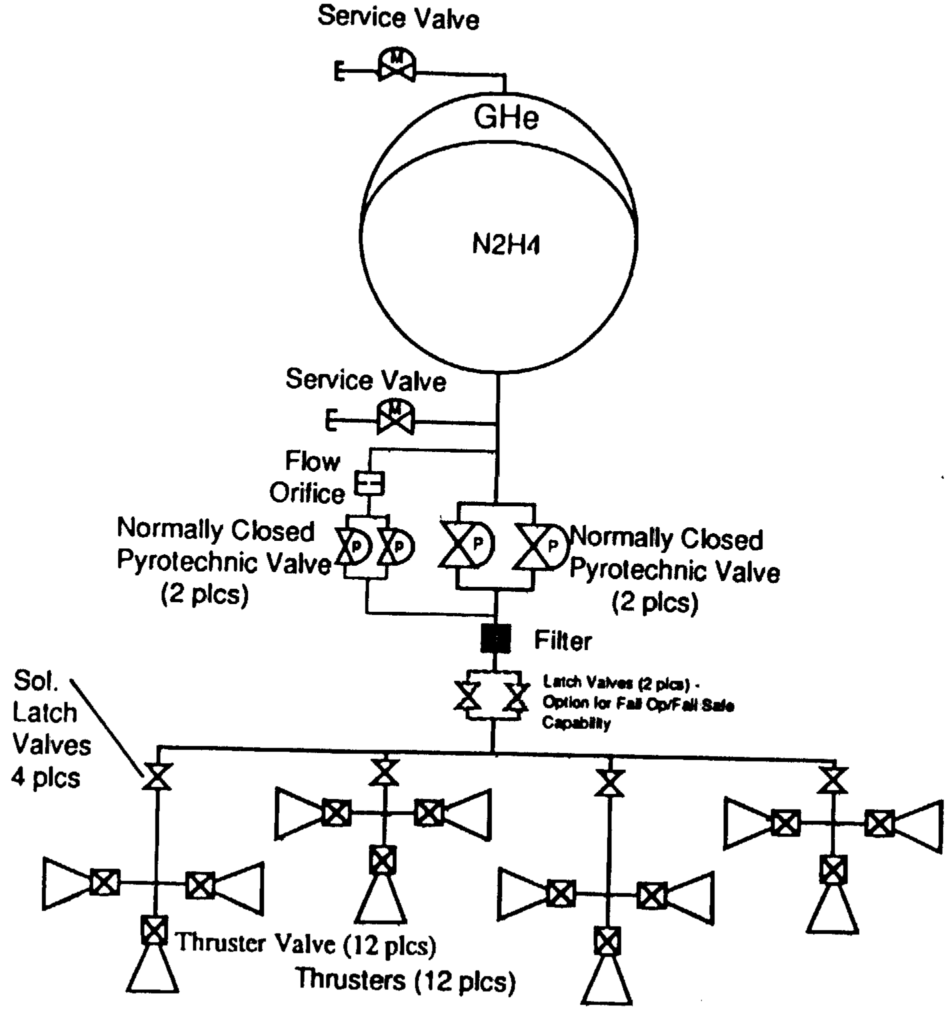

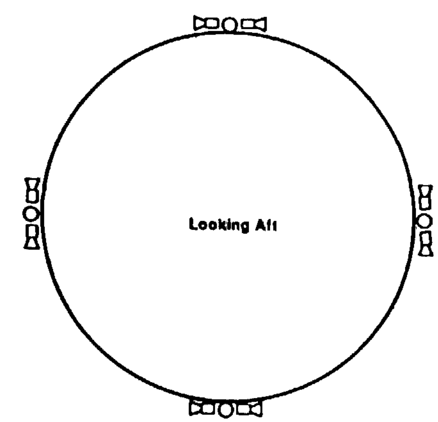

Thruster Layout

- Hydrazine Tanks Can be Added as Required
- Thruster Can be Readily Changed
- Single Fault Tolerance

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TLI Stage Study Status

Summary & Conclusions

John Hodge
(303) 977-2792
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Summary

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Programmatics

- (TLI) On Schedule To Meet All Study Objectives
  - U/S REQ'TS/CONCEPT
  - TLI Stage Lead
  - Involvement In Several Other FLO Team Efforts.
  - Establishing Upper Stage Development Consortium
    - NASA (JSC, KSC, MSFC)
    - Industry (Systems, Propulsion, Structures)

- P/A MODULE
  - DESIGN INTERGRATED INTO SEI/FLO CONFIG
  - SENSITIVITIES INPUTING INTO ONGOING NASA STUDIES

Technical

- Conceptual TLI Stage Configurations
  - NLS
  - Saturn V
- Design Validation
  - Satisfy FLORG Requirements
  - Traceable to FLORG Requirements
- Subsystem Layouts
  - VHM Integrated Across All Subsystems
- External TLI Interface Relationships Defined FOR TLI STAGE

- DETAILED P/A MODULE

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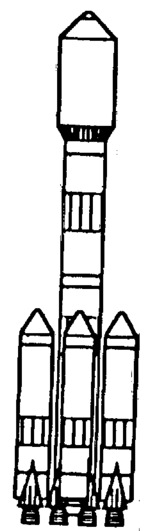
Issues/Concerns

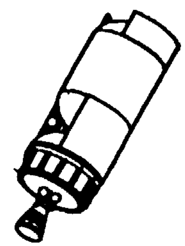
Programmatics

- Changing Business/Political Environments
  - Contractor Funding Profiles
  - Integration Of Foreign Hardware/Technologies
- Reevaluated Business Strategies
  - Government
  - Industry

Technical

- Decision On Launch Vehicle
  - NLS
  - Saturn V
- Requirements Availability/Traceability
- Technology Availability





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TLI Stage Requirements Analysis

John Cuseo
(303) 971-7896
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000

SE920804-02A

Topics

- Goals/Objectives
- Requirements Development Approach
- Functional Analysis
  - Functional Definition
  - Decomposition
  - Function/Element Allocation
- Requirements Analysis
  - Function/Requirements Relationships
  - Requirements Derivation
  - Requirements/Element Allocation
- Interface Requirement Analysis
  - Approach
  - TLI/FLO Element Relationships
  - Interface Requirement Derivation
- Systems Data Management
  - Current On-Line Tools/Capabilities
  - Proposed Capabilities
- Summary/Conclusions

TLI Stage Requirements Analysis Overview

This Analysis is Being Performed as a Subtask of the Overall TLI Study (STV Technical Directive - 12), which Is Responsible for the Further Definition of the First Lunar Outpost TLI System

Requirements Analysis Description

Goals:
- Develop Approach for Requirements Definition
- Perform Functional and Performance Analysis
  - Definition, Decomposition and Flow
  - Performance Parametrics
  - Function/Performance Allocation
- Derive Interface Requirements
- Develop Relational Database (TRACED)

Products:
- Detailed Functional Flows and Dictionary
- Preliminary System (TLI Stage Element) Requirements
- Preliminary Interface Requirements
- On-Line Relational Database for Req'ts Management

FLO TLI Stage Requirements Definition

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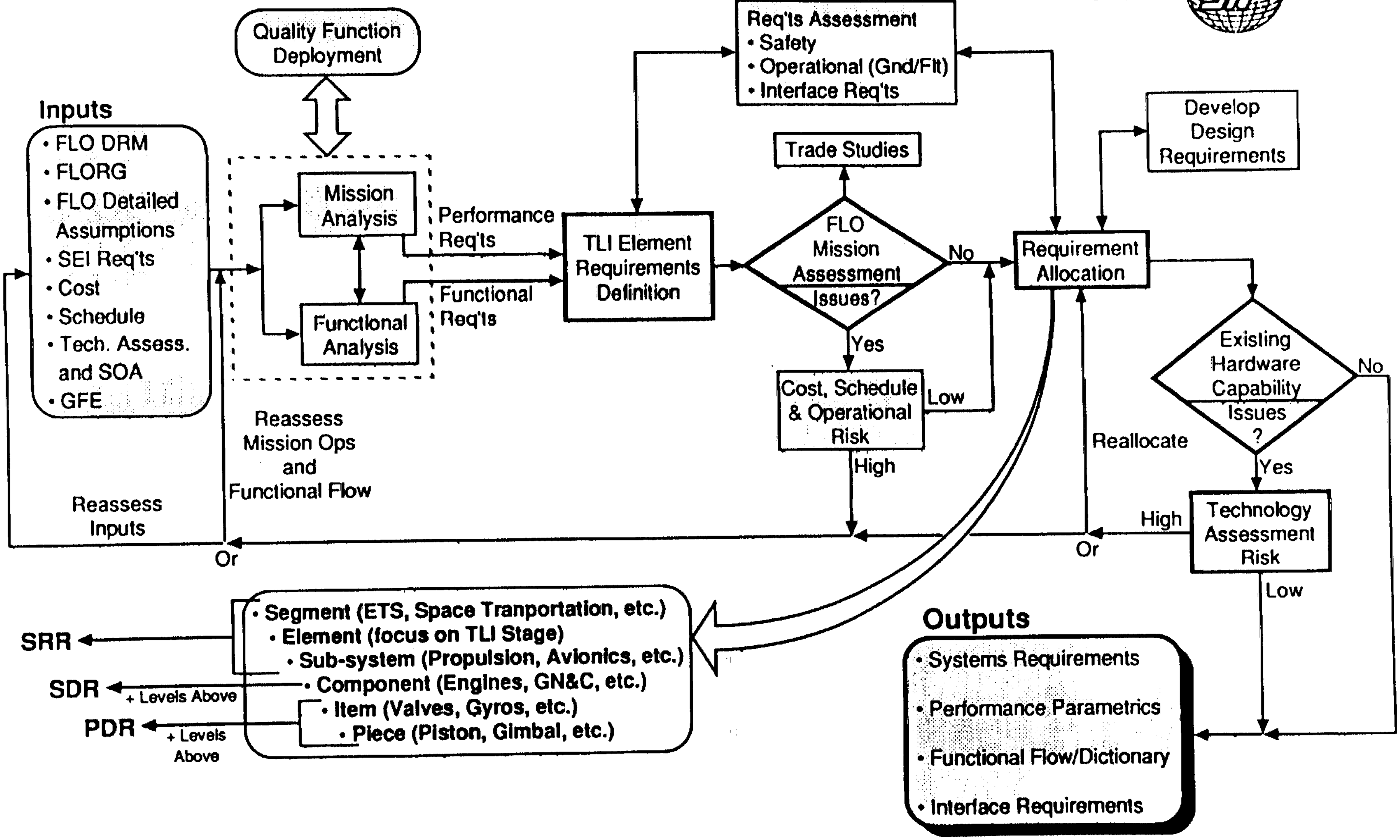

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TLI Stage Function/Requirements Definition

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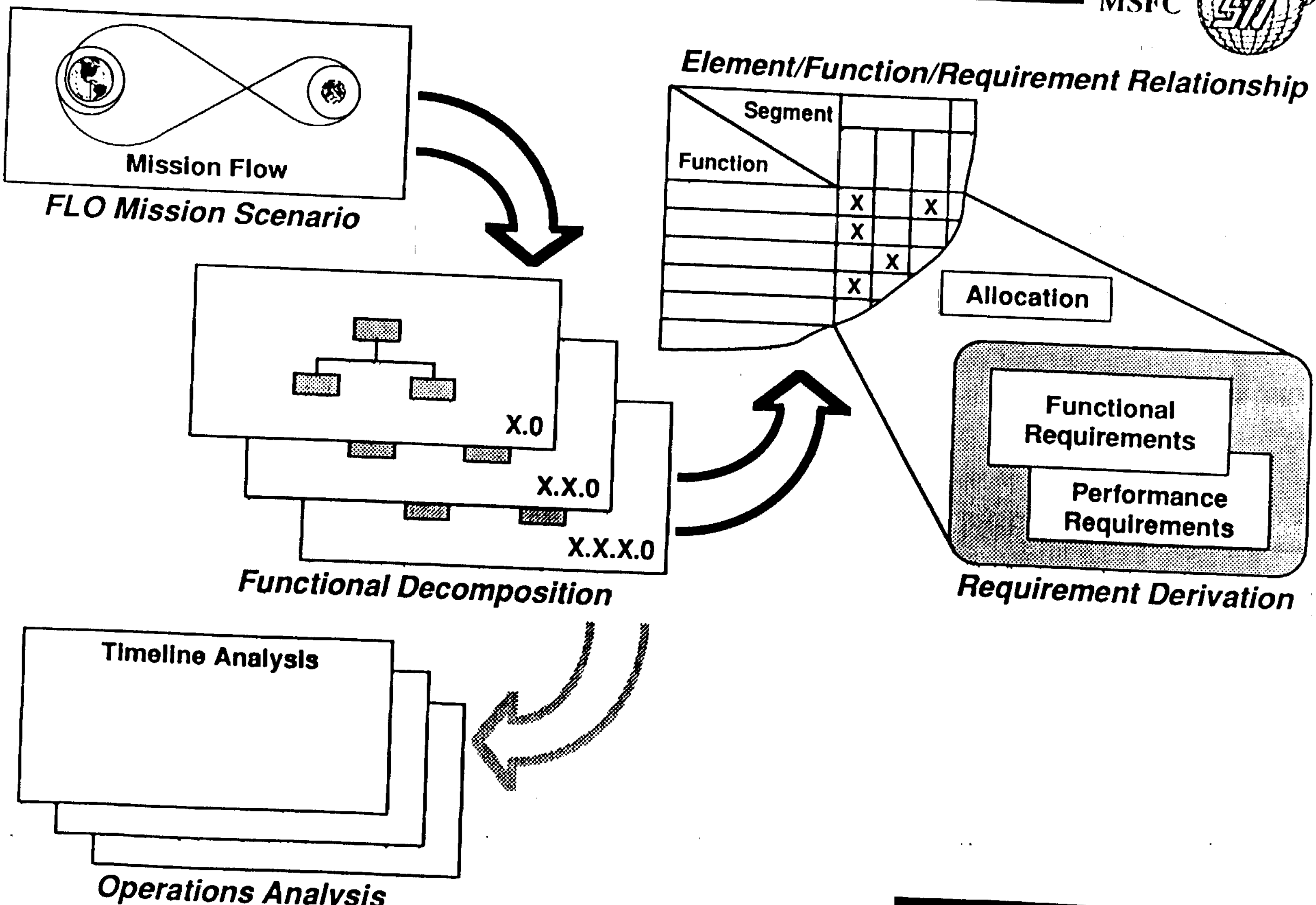

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Define Top Level FLO Functions

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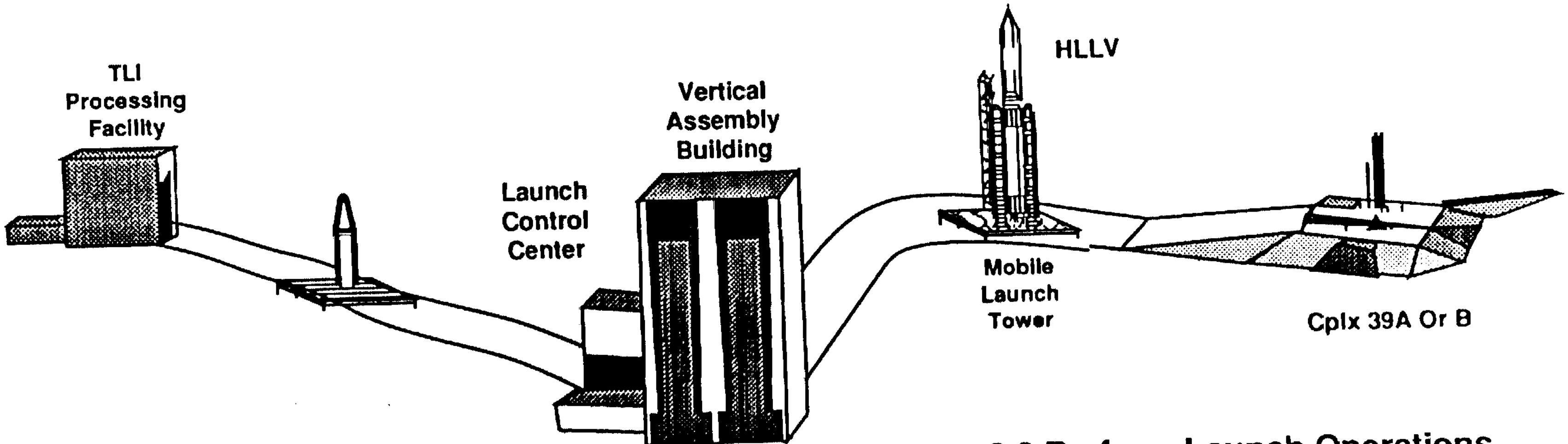

1.0 Perform Pre-launch Operations
- 1.1 Perform TLI Processing Facility Operations
- 1.2 Perform Vertical Assembly Building Operations
- 1.3 Perform Launch Pad Operations
- 1.4 Checkout Mission Operations Infrastructure

2.0 Perform Launch Operations
- 2.1 Perform Boost Phase
- 2.2 Perform Booster Separation
- 2.3 Perform Shroud Separation
- 2.4 Perform MECO and Core Separation
- 2.5 Perform Orbit Insertion and SECO
- 2.6 Perform HLLV/Cargo Separation
- 2.7 Provide Range Safety Data

Dave Lamar (Ops Sub-Team)
(713) 283-5302

MTG ~~THURSDAY~~ (13 Aug)
0800 (THURSDAY) FOR COORDINATION OF FUNCTION (TOP LEVEL)
121 RM
NOVA

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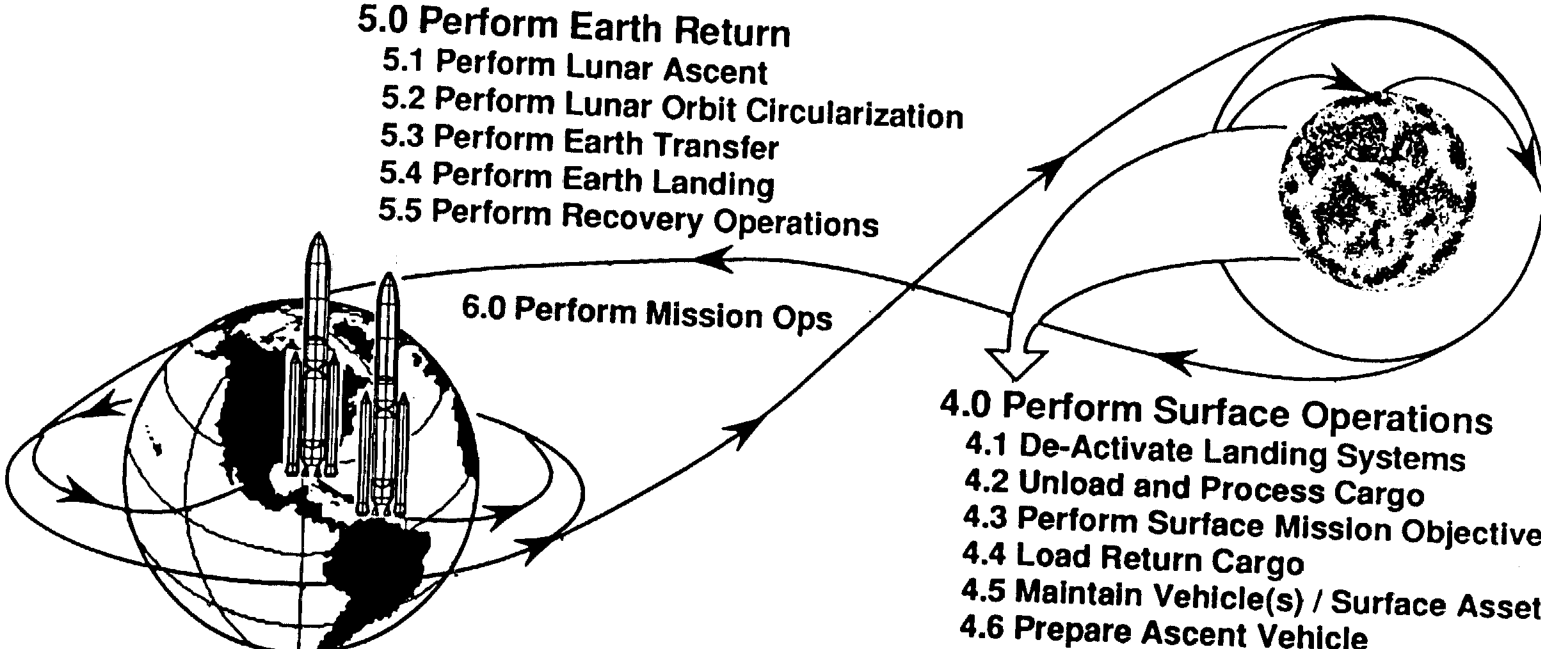
LR920707-03

Define Top Level FLO Functions (cont)

MSFC



5.0 Perform Earth Return
- 5.1 Perform Lunar Ascent
- 5.2 Perform Lunar Orbit Circularization
- 5.3 Perform Earth Transfer
- 5.4 Perform Earth Landing
- 5.5 Perform Recovery Operations

6.0 Perform Mission Ops

4.0 Perform Surface Operations
- 4.1 De-Activate Landing Systems
- 4.2 Unload and Process Cargo
- 4.3 Perform Surface Mission Objectives
- 4.4 Load Return Cargo
- 4.5 Maintain Vehicle(s) / Surface Assets
- 4.6 Prepare Ascent Vehicle

3.0 Transfer to Lunar Surface
- 3.1 Damp Separation Rates
- 3.2 Perform Earth Orbit Stabilization
- 3.3 Perform Lunar Transfer
- 3.4 Perform Lunar Orbit Insertion
- 3.5 Perfom Lunar Descent and Landing

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FLO TLI Stage Functional Decomposition

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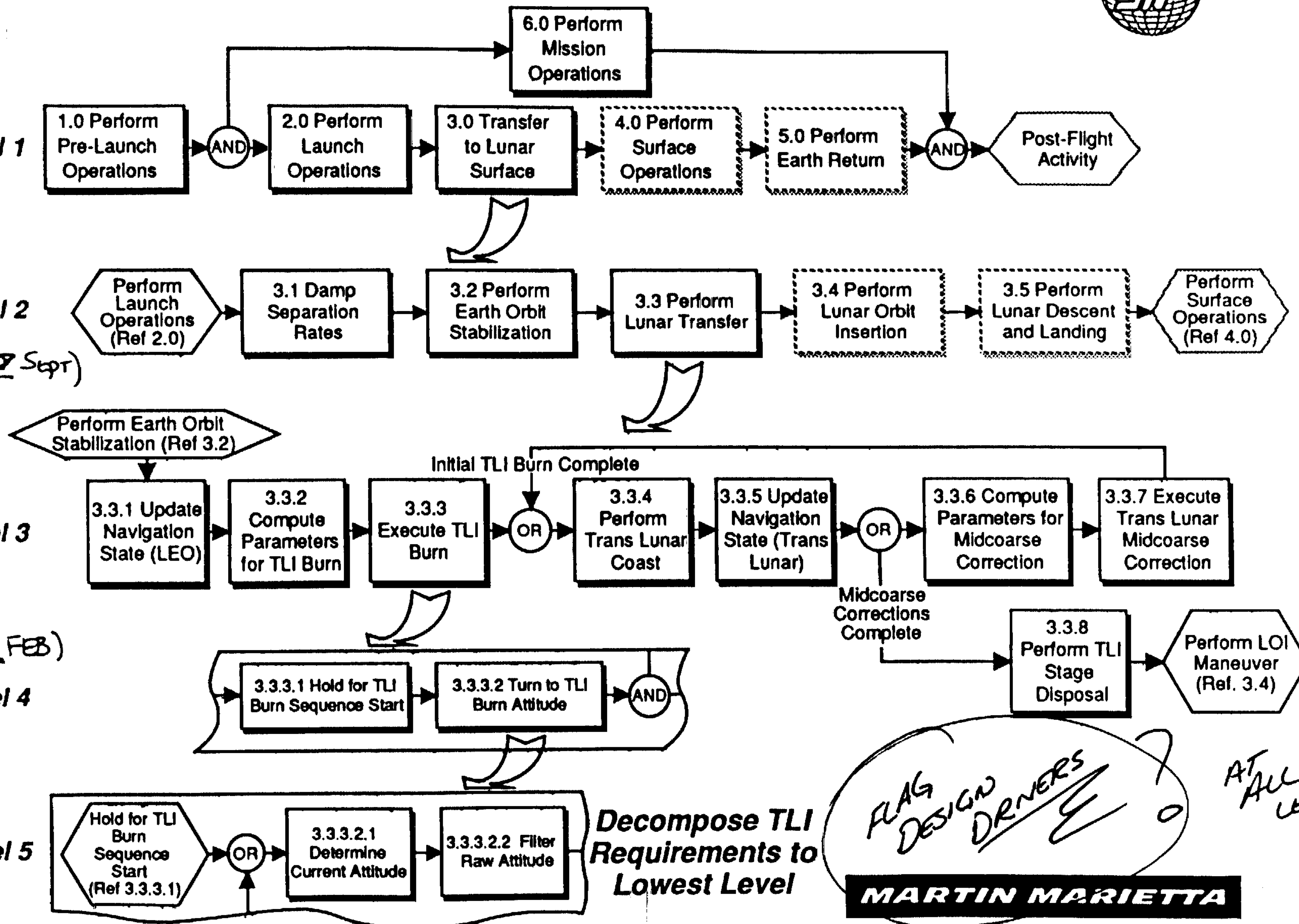
EED TO WLD FOR NLS { SAT ✓



SEE I (~~FEB~~ SEPT)

SRR II (FEB)

Decompose TLI Requirements to Lowest Level

FLAG DESIGN DRIVERS

AT ALL LEVELS

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FLO TLI Stage Functional Decomposition

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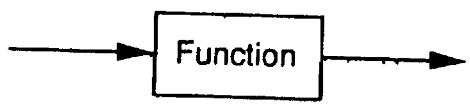
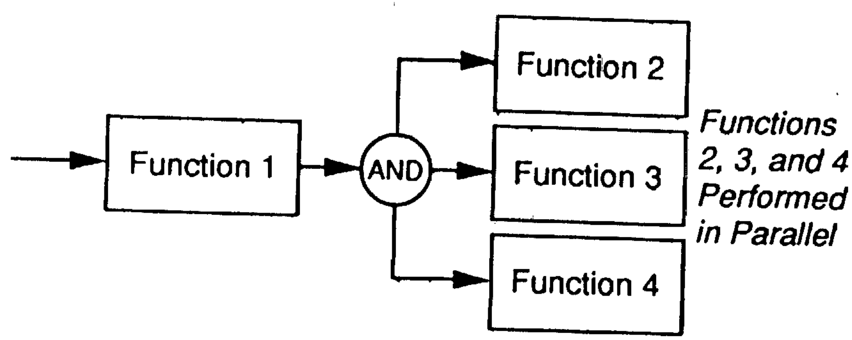
Functional Flow Format Guidelines

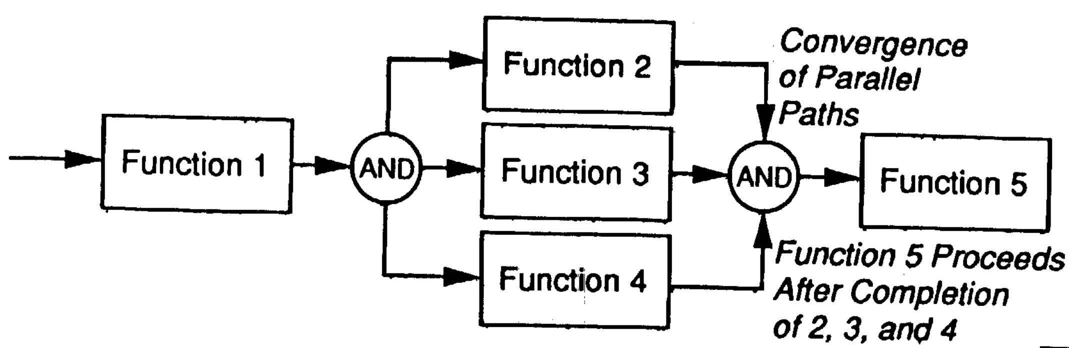
- Flow Charting Software (i.e., MacFlow™) Aids in Development of Complex Flows
- Strict Adherance to Format Guidelines Increases Flexibility, Maintainability and Expandability of Complex Functional Flows



- Function Blocks Limited to One Input and One Output Arrow with Multiple Inputs/Outputs Handled via AND Gates and OR Gates



- AND Gates with Multiple Outputs Indicate the Start of Parallel Functions



- AND Gates with Multiple Inputs Indicate the Convergence, or End, of Parallel Functional Paths

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FLO TLI Stage Functional Decomposition

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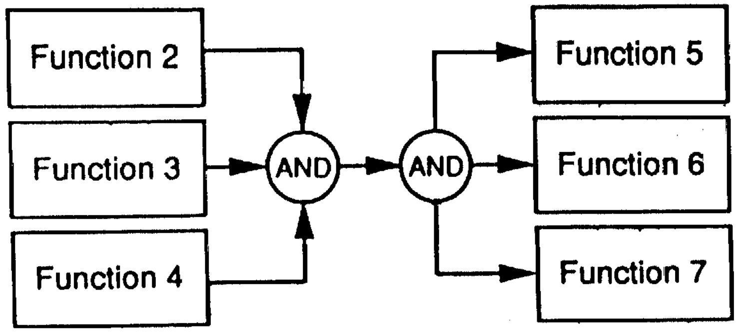
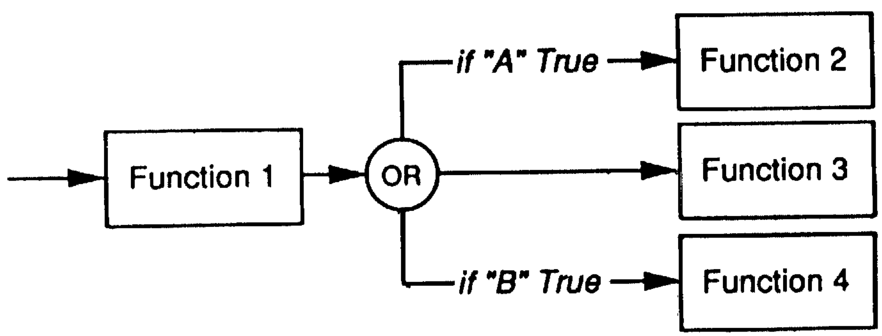
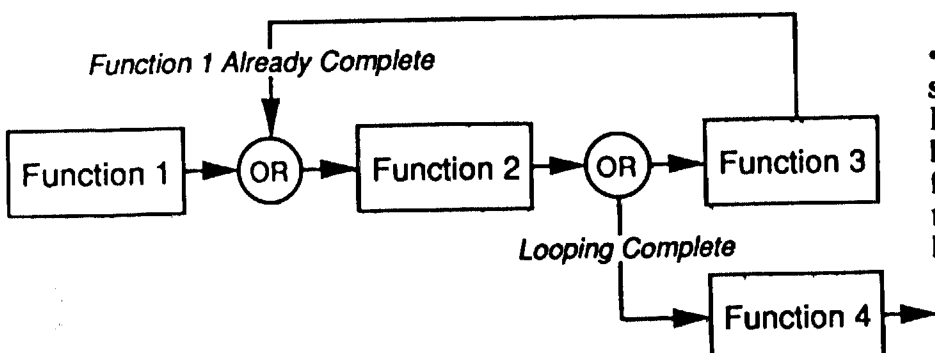
Functional Flow Format Guidelines (continued)



- Successive Parallel Paths Require Use of Two AND Gates (parallel functions 5, 6 and 7 can proceed after completion of 2, 3 and 4).



- OR Gates with Multiple Outputs Indicate Start of Diverging Functional Paths. Only One Functional Path can be Followed Based on a Conditional Statement Specified by Labeling Lines Emerging from OR Gate. One of the Emerging Lines Need Not be Labeled which Indicated That if All Other Conditional Statements Evaluate to FALSE then the Unlabeled Line is the Proper Path by Default (function 3 selected if both "A" and "B" are FALSE).



- OR gates with multiple inputs indicate a loop for specifying the repetition of selected functions. Multiple lines converging at an OR gate must be labeled to sufficiently indicate the current functional flow path. Note that repetition loops set up in this manner must incorporate another OR gate within the loop to avoid an endless cycle of repetitions.

FLO TLI Stage Functional Analysis

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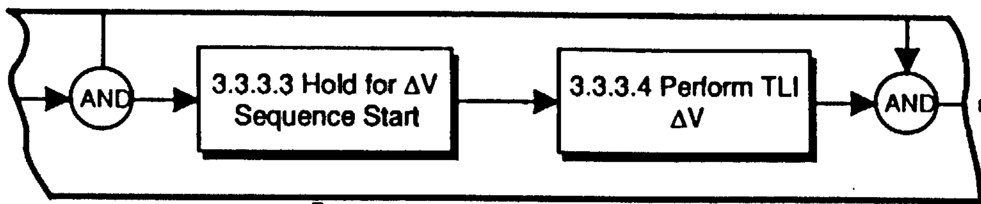
- Functional Analysis Provides Basis for Flight Operations Analysis
  - Mission Timeline and Flight Ops Event Analysis
  - Telemetry/Command Data Element Definition per Function
  - Contingency Planning



Timeline Analysis

Function: 3.3.3.4
Name: Perform TLI ΔV
Time Required: 490 seconds
Assumptions:
- J2 Engine 265,000 lbf Thrust
- 436 seconds Isp
- Total Weight 564,000 lb
- 10,500 ft/s ΔV Required

DESIGN SPECIFIC

Derivation:
$\Delta T = \Delta M/Mdot = 297{,}675/607.8 = 490$ s

Telemetry/Command Data Elements

Function: 3.3.3.4
Name: Perform TLI ΔV
Required Telemetry Elements:
Propulsion
E0951,0952: Pressurant Tank Pressures
E0021,0035: Pressurant Tank Temp
E0953-0954: Propellant Tank Pressures
E0020,0033,0036: Propellant Tank Temp

Contingency Planning

Function: 3.3.3.4
Name: Perform TLI ΔV
Anomaly ID: 0685
Anomaly Description: Loss of Propellant Tank 1 Pressure
Element Applicability: TLI Stage
Subsystem Applicability: Propulsion
Telemetry Element: E0953
Response Description

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FLO TLI Stage Functional Allocation

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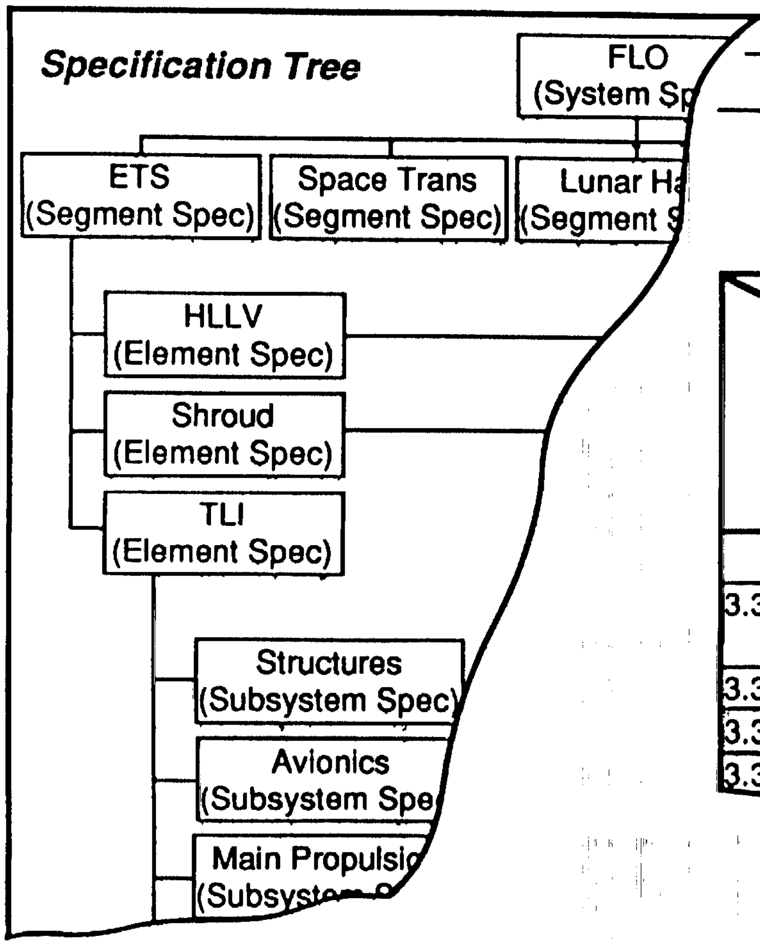
- Initial Allocation of Functions Performed After Decomposition
- Evaluate Allocation After Performance Requirements Have Been Derived
- Functional/Performance Allocation Must Be Linked to Specification Tree for Traceability



Segment/Element Allocation / Functions	ETS: HLLV	ETS: Shroud	ETS: TLI Stage	Space Trans: Launch Esc	Space Trans: Lander	Space Trans: Crew Module	Space Trans: Log Carrier	Space Trans: Return Stage
⋮								
3.3.3.1 Hold for TLI Sequence Start			√		√	√	√	
3.3.3.1.1 Maintain Pre-TLI Attitude			√					
3.3.3.1.2 Verify PreTLI Health/Status			√		√	√	√	
3.3.3.1.3 Refine Burn Duration Calc			√					

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Function/Requirement Relationship

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Functions \ Segment/Element Allocation	ETS			Space Trans				
	HLLV	Shroud	TLI Stage	Launch Esc	Lander	Crew Module	Log Carrier	Return Stage
⋮								
3.3.3.1 Hold for TLI Sequence Start			√		√	√	√	
3.3.3.1.1 Maintain Pre-TLI Attitude			√					
3.3.3.1.2 Verify PreTLI Health/Status			√		√	√	√	
3.3.3.1.3 Refine Burn Duration Calc			√					

3.X LEVEL

FLORG Reference

Paragraph: 5.2.1.2.1 TLI Stage Element
Number: 418

The TLI stage element shall provide the capability for attitude correction prior to TLI burn. (B. Pattison 03/03/92)

Performance Analysis and Requirements

Function Number: 3.3.3.1.1
Function Name: Maintain Pre-TLI Attitude

Performance Requirements

3.3.3.1.1.a Attitude Accuracy Prior to TLI Burn
Responsibility: S. Earley, MMAG

3.3.3.1.1.b Rotational Acceleration (control authority) Required for Pre-TLI Attitude Control
Responsibility: J. Cuseo, MMAG

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Function/Requirement Relationship

MSFC

Segment/Element Allocation / Functions	ETS			Space Trans				
	HLLV	Shroud	TLI Stage	Launch Esc	Lander	Crew Module	Log Carrier	Return Stage
⋮								
3.3.3.1 Hold for TLI Sequence Start			√		√	√	√	
3.3.3.1.1 Maintain Pre-TLI Attitude			√					
3.3.3.1.2 Verify PreTLI Health/Status			√		√	√	√	
3.3.3.1.3 Refine Burn Duration Calc			√					

FLORG Reference

Paragraph: TBD
Number: TBD

TBD

Performance Analysis and Requirements

Function Number: 3.3.3.1.2
Function Name: Verify Pre-TLI Health/Status

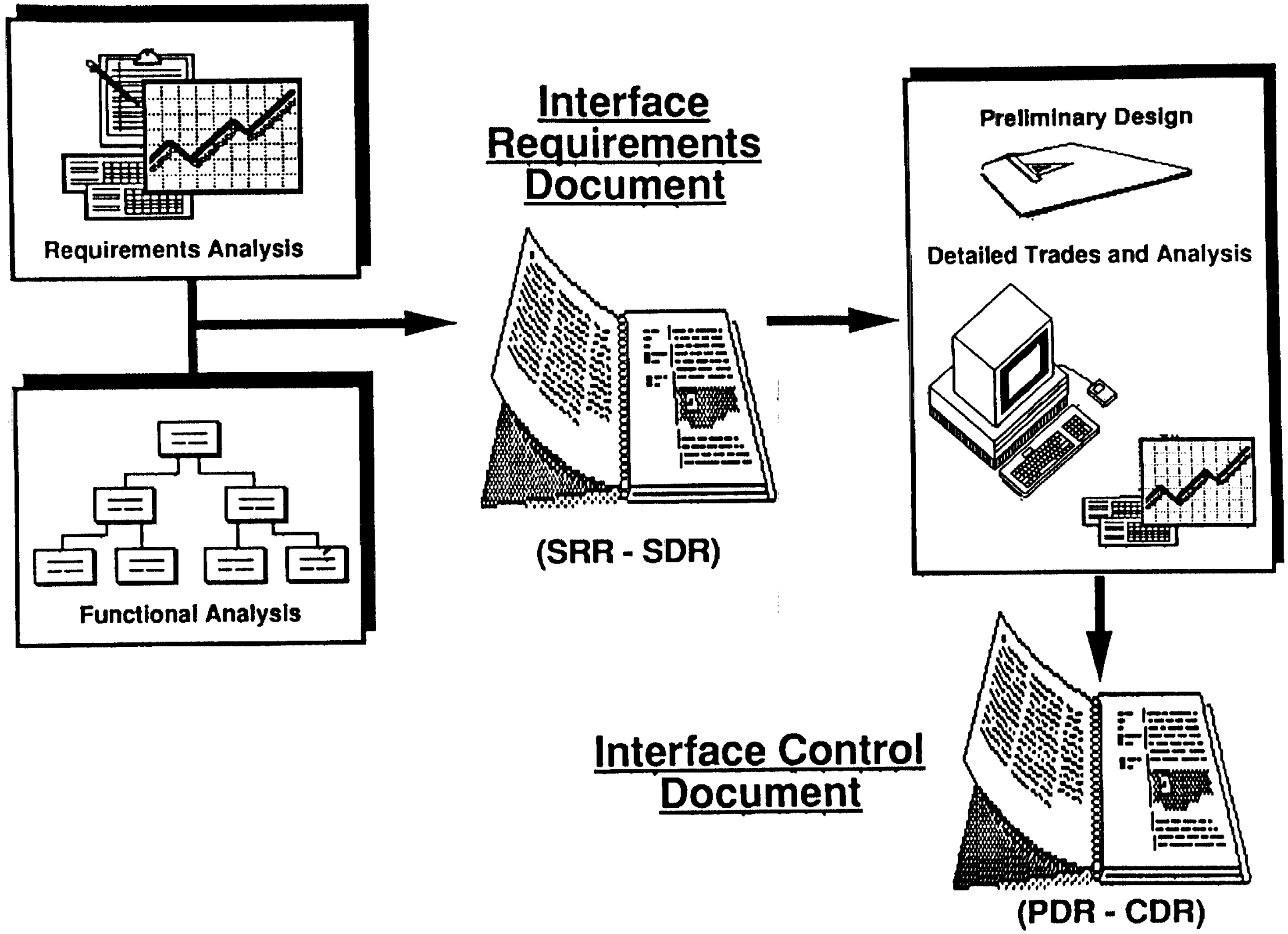
Performance Requirements

3.3.3.1.1.a Maximum Time For Detection/Reporting of All Critical Failure Modes
Responsibility: R. Welborne, MMAG

3.3.3.1.1.b Percentage of Non-Critical Failure Occurances Detected
Responsibility: R. Welborne, MMAG

Interfaces

STV/TLI Interface Analysis Flow



Interface Methodology

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- Prepare functional analysis

- Identify external and internal interfaces

- Allocate functions to interfacing entities

- Identify type of interface for each function (i. e. mechanical, electrical, fluid, data, environmental)

- Perform analysis to determine performance and interface requirements associated with each function

- Use functional, performance and interface requirements to populate initial IRD

Identify External Interface Elements & Types

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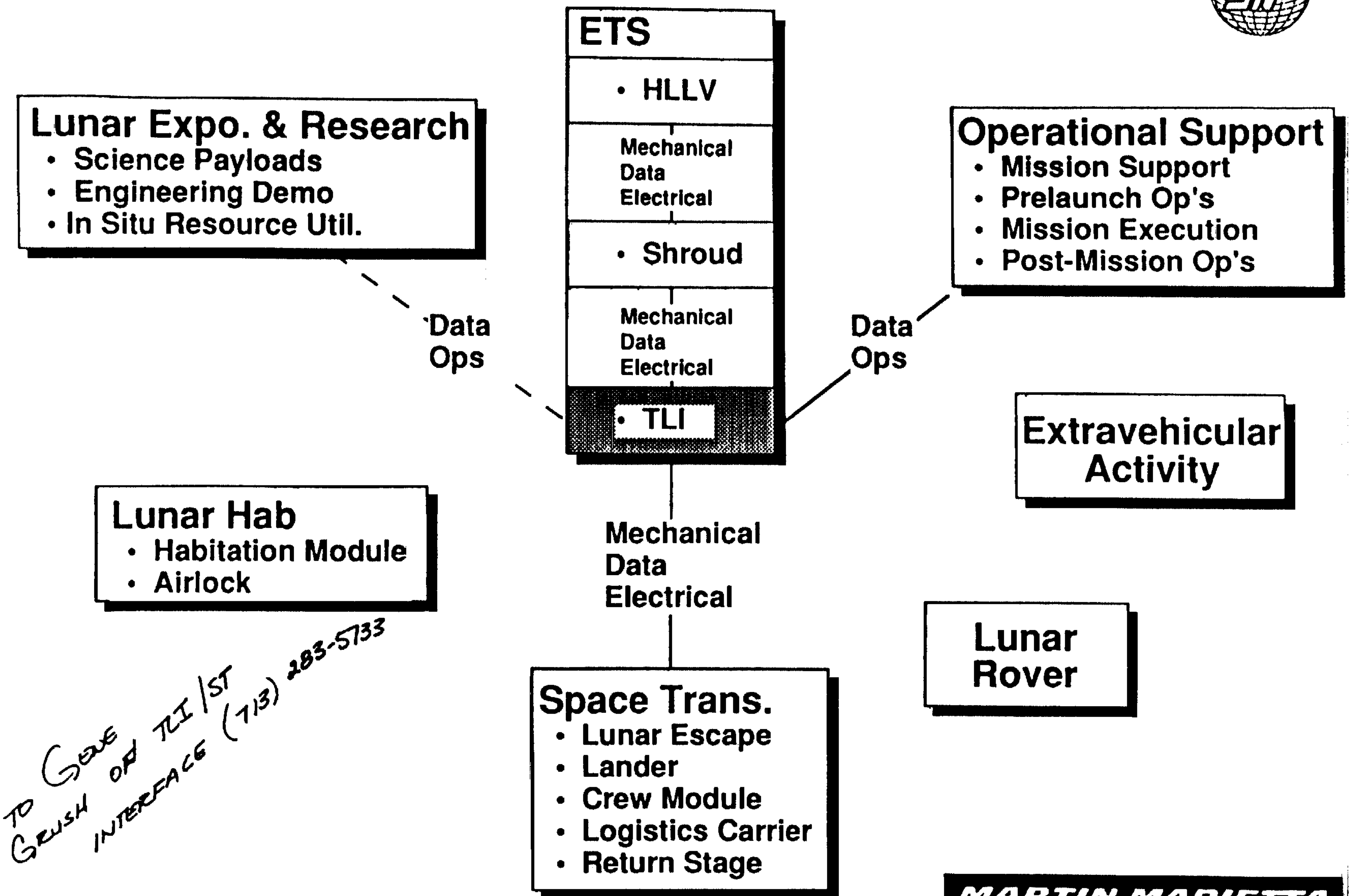

TALK TO GENE GRUSH OF TLI/ST INTERFACE (713) 283-5733

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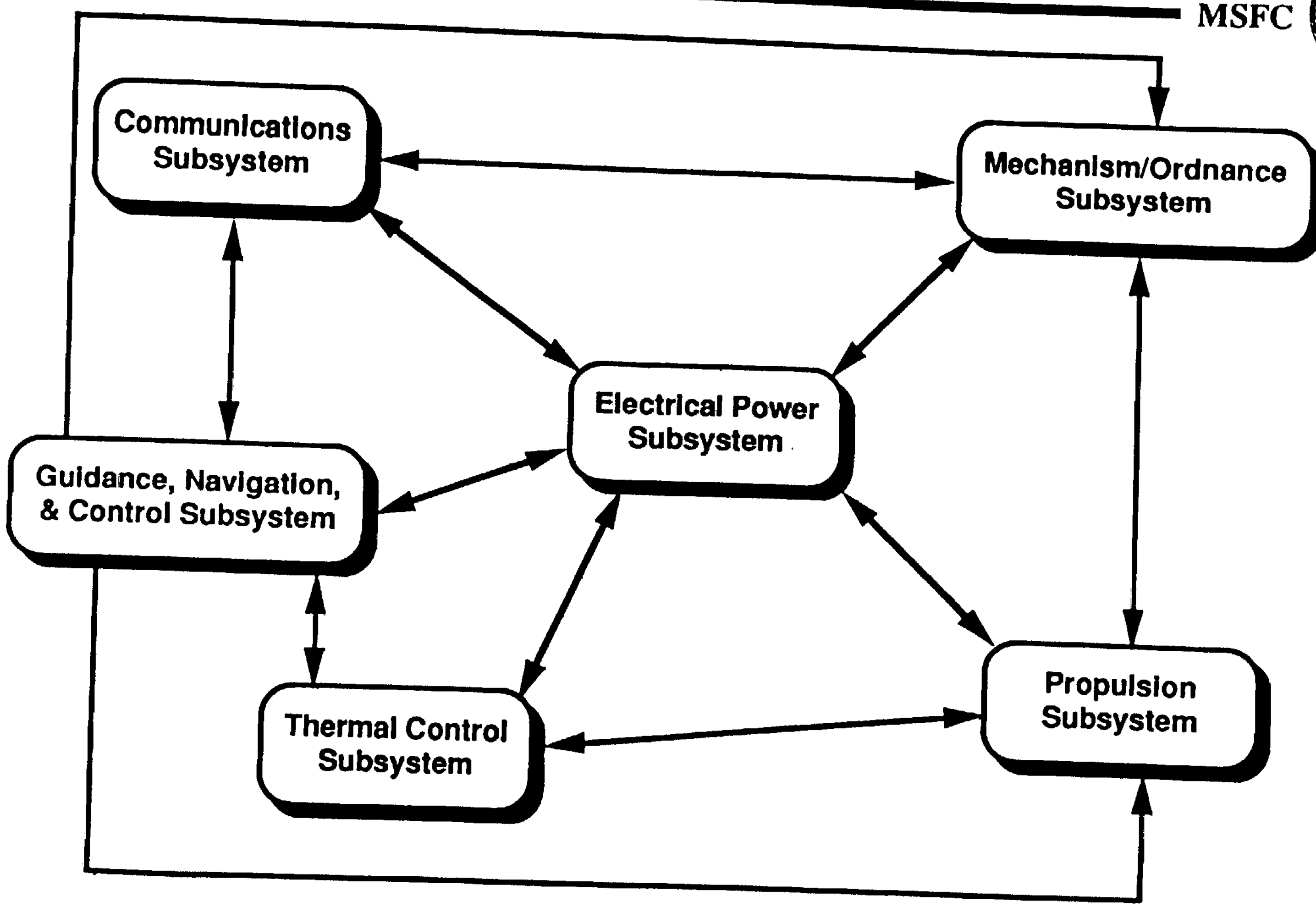
TLI Stage Internal Interfaces

MSFC



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Function\Segment\Interface Traceability

MSFC

Function	Elements / Allocation	HLLV	Shrd	TLI St	L Esc	Lndr	Crw M	Log C
2.1 Perform Boost Phase	HLLV	–	NM	NMDE	N	NP	N	N
	Shrd	NM	–	MD		P	P	P
	TLI St	NMDE	MD	–		NM	N	N
	L Esc	N			–		NM	
	Crw M	N		N	NM	NM	–	NE
	Msn E	D	D	D	D	D	DVH	
2.2 Perform Booster Sep	HLLV							

MAY NEED TO WORK AT SEGMENT LEVEL NOT ELEMENT.

FLORG Reference

Paragraph: 5.2.1.2.1 TLI Stage Element Number: 417

The TLI stage element shall provide the capability for ascent guidance and control of the launch vehicle during launch from Earth. (B. Pattison 03/03/92)

Performance Analysis at the TLI/HLLV Interface

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Systems Data Management

MEET W/DEB ON SDM
THIS WEEK (713) 283-5755

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RS920807-02A

Systems Data Management

- Relational Database Required to Support Entire Program Life Cycle
  - Automates/Assists System Engineering Functions
  - On-Line Access to All Program Requirements
  - Requirement Relationships, Traceability and Verification
  - Requirement Maintenance and Change Control
  - Documentation Automatically Generated Directly from Database
- Avoid Problems with Current Manual Methods
  - Unsatisfied, Inconsistant and Unverifiable Requirements
  - Incomplete, Error Prone Requirements Change Processing
  - Improperly Managed System Configurations

Relational Databases for TLI Stage Analysis

MSFC

- Current Capability
  - Relational Database Written in FoxBase+/Mac™
    - Functional Requirements (and Flow Blocks)
    - Performance Requirements
    - Interface Requirements
    - Mission Requirements
    - Segment Requirements
    - Element Requirements
    - Subsystem Requirements

- Proposed Capability
  - Systems Engineering Database (SEDB)
    - Developed by Martin Marietta
    - Fully Automated Systems Engineering Tool
    - In Use on Several Martin Marietta Programs
    - Implementation in Progress for STV/TLI Systems Engineering

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Current Data Management Cabability

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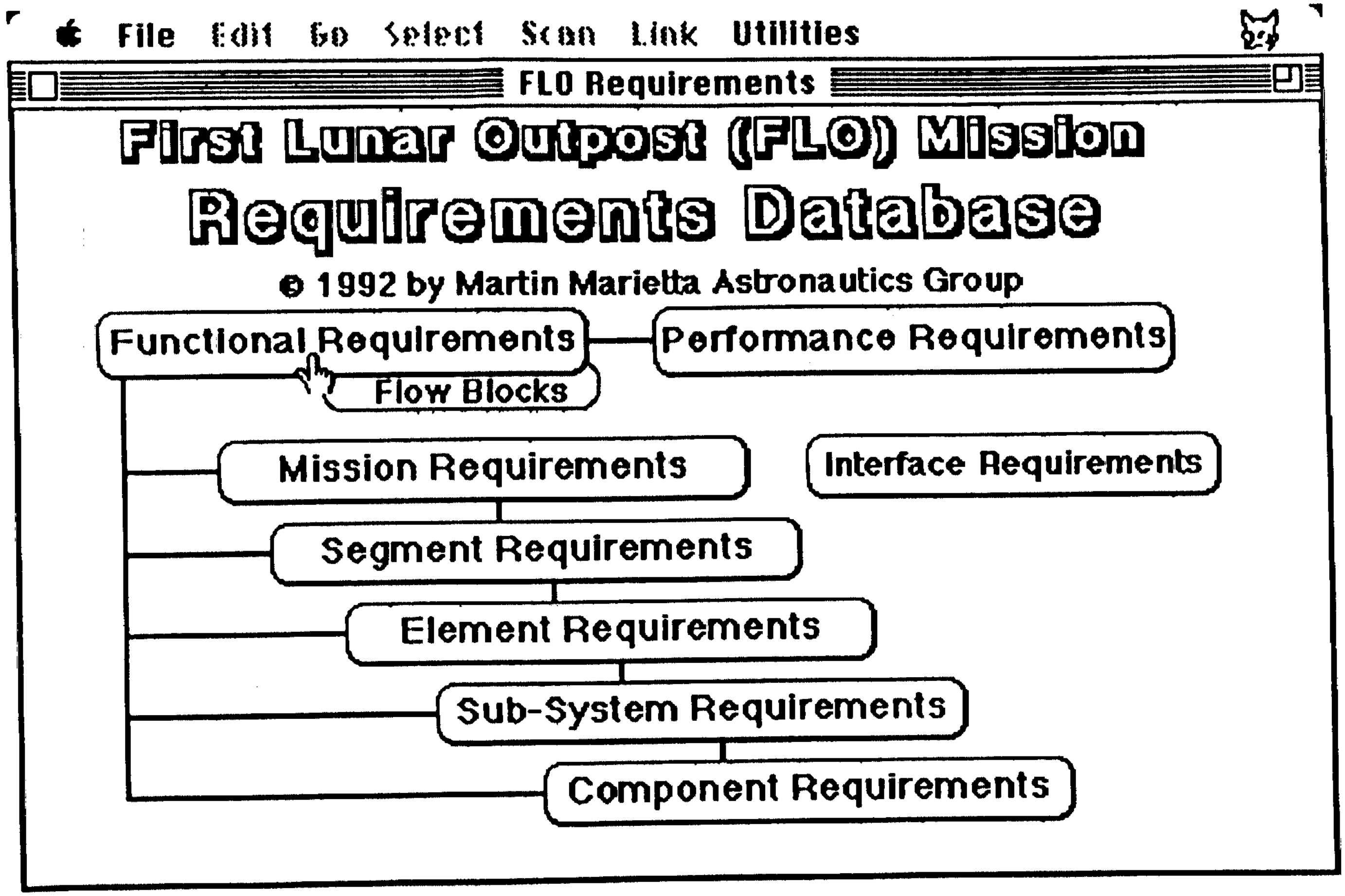
Full Relational Database for FLO TLI Stage Requirements



* Written in FoxBase+/Mac™

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Current Data Management Cabability

Functional Requirement Description/Allocation

MSFC

File Edit Go Select Scan Link Utilities

Functional Requirements

Close

Number: 3.3

Name: Perform Lunar Transfer

Last Update: 07/29/92

Source: MMAG - STV Program

Description:

This function begins after the lunar mission element have attained a stable low earth orbit. This function includes navigation states updates, TLI burn, coast, midcourse corrections, and separation of the TLI stage. This function ends with the disposal of the TLI stage (with the lunar landing element on the final trans lunar trajectory).

Functional Requirement Allocation:

- ☒ ETS — ETS Elem.
- ☒ Space Tran. — SpcTrn Elem.
- ☐ Lunar Hab — LunHab Elem.
- ☐ Lunar Rover — LunRov Elem.
- ☐ EVA — EVA Elem.
- ☐ Explr/Resrh — Ex/Res Elem.
- ☒ Ops Supp. — Ops Elem.

Forward

Back

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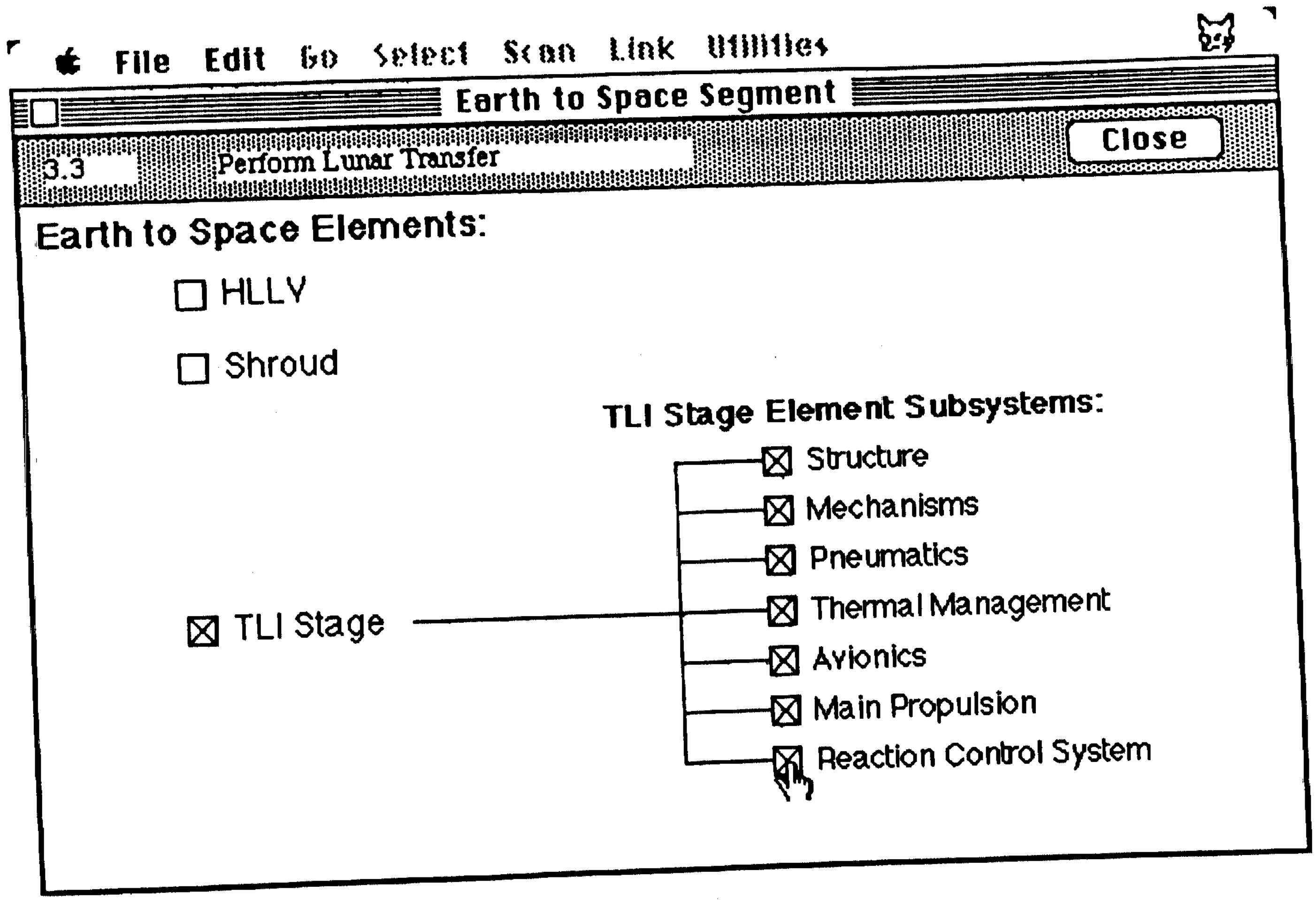
Current Data Management Cabability

MSFC

Element and Subsystem Allocation



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JCu920803-05A

Current Data Management Cabability

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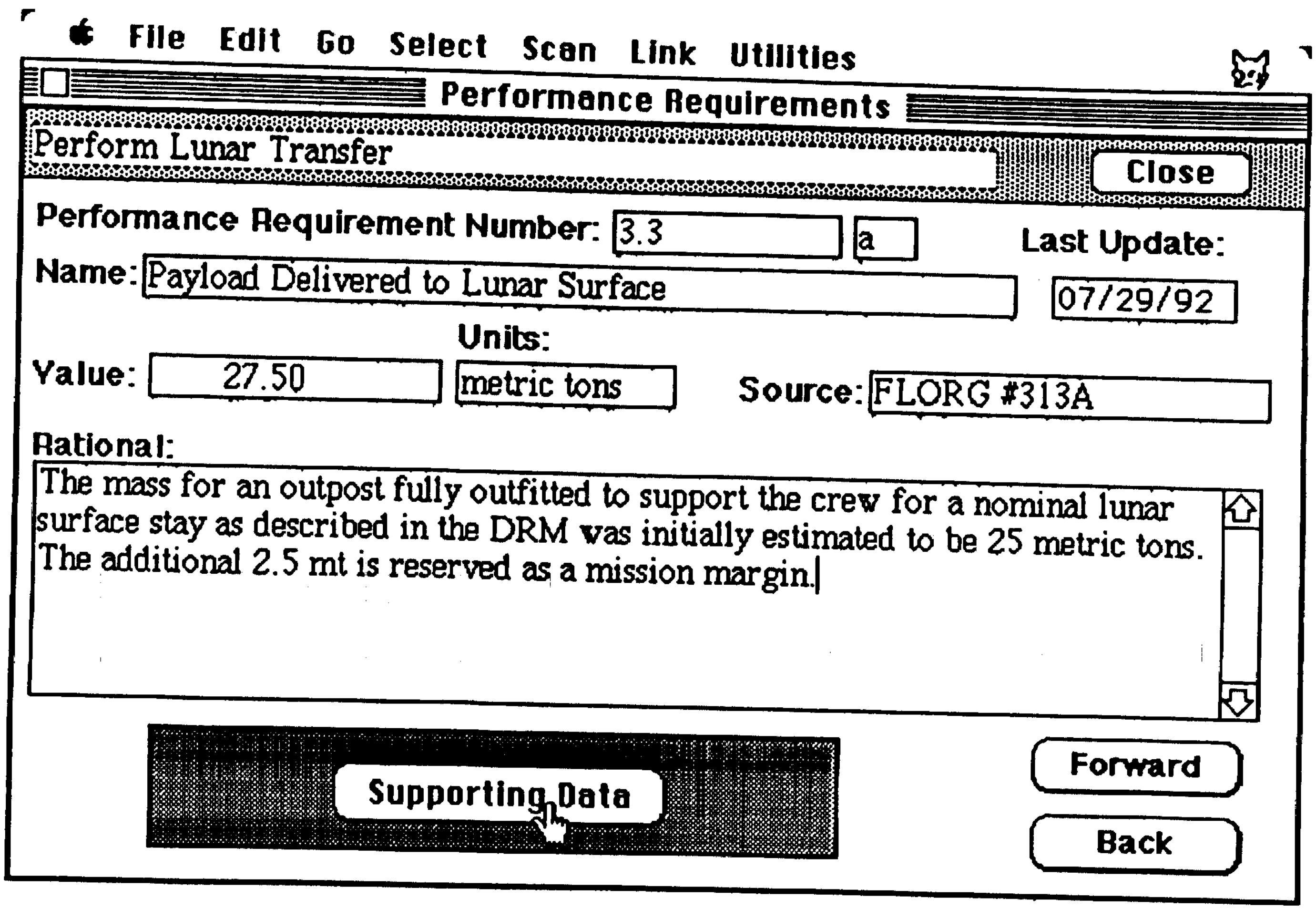
Performance Requirements Linked to Each Function



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Proposed Data Management Cabability

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Systems Engineering Database (SEDB)

- A SYSTEMS ENGINEERING TOOL
  - MACINTOSH-BASED
  - USER INTERFACE IN 4TH DIMENSION DATABASE LANGUAGE
  - DATABASE STRUCTURE IN ORACLE

- AUTOMATES/ASSISTS IN PERFORMING SYSTEMS ENGINEERING FUNCTIONS THROUGH ALL PROGRAM PHASES
  - REQUIREMENTS TRACEABILITY, VERIFICATION
  - REQUIREMENTS MAINTENANCE, CHANGE CONTROL

- MULTI-USER SIMULTANEOUS ACCESS TO PROGRAM REQUIREMENTS AND RELATED INFORMATION
  - CUSTOMER REMOTE ACCESS, SQL DIRECT QUERY CAPABILITY

Summary

MSFC

- Approach Links ("Why" and "What") FLO System Description and TLI Stage Element Functions and Requirements
  - Applicable at All Levels
- External/Internal Interface Requirements Derived From Element Functional and Performance Requirements – Traceable To FLORG
- Requirements Analysis Efforts Have Identified FLORG/TLI Stage Element Discrepancies
- Implemented Data Management System
  - Existing Database
  - Expandable With Program
  - Multi User (Internal & External)

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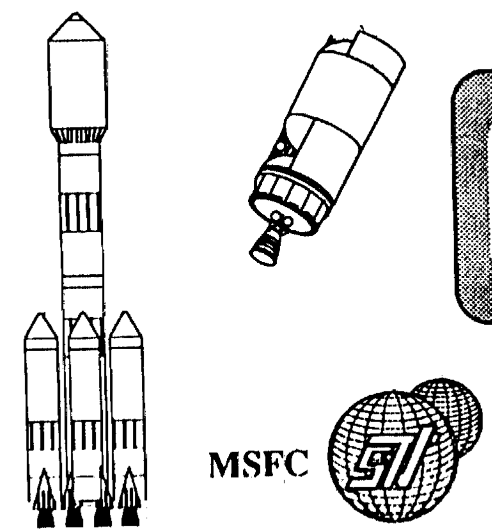
Space Transfer Vehicle
Concepts & Requirements Study

(NAS8-37856)



Heavy Lift Launch Vehicle
Technical Interchange Meeting (TIM) #10

HLLV/TLI Stage Vehicle Health Management Review

18 November 1992

Jim McKinnis
(303) 977-9895
Ron Welborne
(303) 971-5253

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HLLV/TLI Stage Vehicle Health Management Review

VHM Task Overview	Jim McKinnis
VHM Requirements	Jim McKinnis
VHM Technologies & Benefits Assessment - Architectures - Electrical/Electronics - Ground Processing - Power - Software - Propulsion	Ron Welborne
System Recommendations - Where - How Much - Why	Ron Welborne
Technology Recommendations - Development Cost Projections - Demonstration Candidates - Additional Analysis	Jim McKinnis

Integrated Vehicle Health Management

IVHM Definition: *

Integrated Vehicle Health Management (IVHM) is the Capability to Efficiently Perform Checkout, Testing, and Monitoring of Space Transportation Vehicles, Subsystems, and Components Before, During, and After Operations. This Includes the Ability to Perform Timely Status Determination, Diagnostics, and Prognostics. VHM must Support Fault-Tolerant Response Including System/Subsystem Reconfiguration to Prevent Catastrophic Failures; and VHM must Support the Planning and Scheduling of Post-Operational Maintenance.

IVHM Goals: *

- Increase Safety and Reliability Providing Increased Probability of Mission Success
- Reduce Processing and Operations Time, Manpower and Costs
- Increase System Availability and Utility

IVHM will be Accomplished by: *

- Enhancing the Effectiveness of Development Testing and Supporting the Development of Design Databases and Simulations
- Preventing Catastrophic Failures in Test and Flight Operations
- Predicting Component End-of-Life or Degradation
- Automating Checkout and Monitoring Functions to Significantly Reduce Manpower
- Reducing Need for Scheduled Maintenance
- Improving Analytical Capabilities and Human/System Interfaces

* OAST Research and Technology Goals and Objectives for IVHM (October 10, 1992) under Auspices of Strategic Avionics Technology Working Group (SATWG)

Lessons Learned from Transfer Orbit Stage

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The Recent Titan III/TOS Launch Provides Information Which Supports the Potential Benefits of a VHM System

Transfer Orbit Stage (TOS) Experience	Suggested Improvements
• Cabling and Instrumenting the TOS Electronic Units During Integration and Test was very Laborious (3 days).	• More Automated Built-In-Test Capability and the use of a Databus throughout the Avionics Suite would have Reduced the Cabling and Instrumentation down to 1 or 2 Test Cables and Less than 1 Day.
• Once TOS was Enclosed in the Payload Faring and on the Pad, Ground Personnel were no Longer able to Perform a 'Full-Up' Functional Test or Deployment Simulation. No Provisions were Made to Perform an Integrated (Launch Vehicle/Upperstage/Payload) Functional Test Which Eventually Led to a Launch with an Undetected Failure.	• An Onboard Automated Health Management System would Provide the Capability to Perform Full Functional Tests and Deployment Simulations Outside of the Payload Handling and Servicing Facility.
• No Provisions or Procedures were made for Determining the Health of the TOS Vehicle in the Event of a Direct or Indirect Lightning Strike while on the Pad.	• Once Again, an Automated Health Management System would have Provided the Capability to Confidently Determine the Health of the Vehicle while on the Pad.

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HLLV/TLI VHM Trade Study Overview

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This Trade Study was Performed as a Subtask of the Overall TLI Study (STV Technical Directive - 12), which Is Responsible for the Further Definition of the First Lunar Outpost TLI System

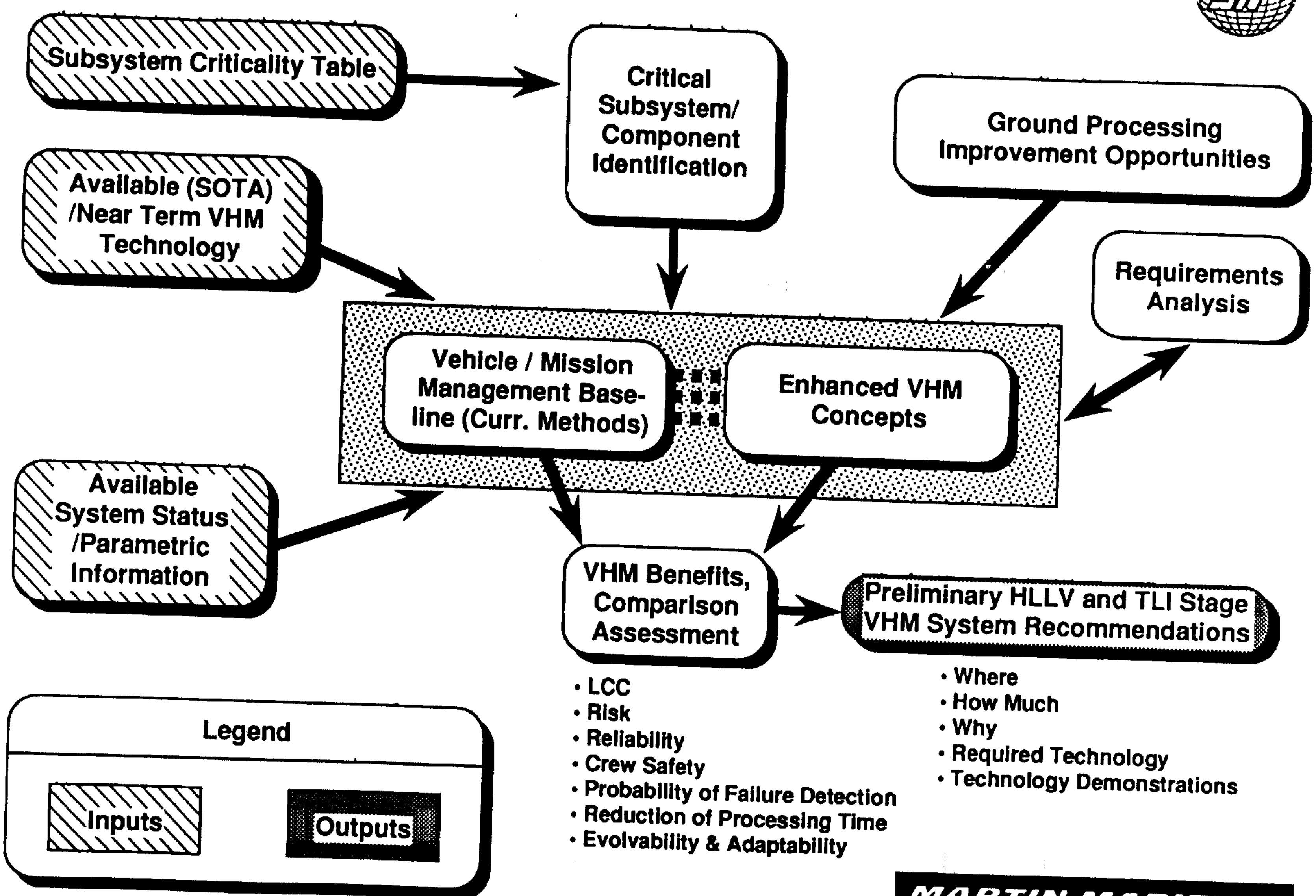
VHM Trade Study Description

Task Duration: 2.5 Months

Goals:
- Define the Bounds of HLLV/TLI VHM
- Determine Existing VHM Capabilities
- Determine the Extent of VHM Required
  - Where
  - How Much

Products:
- Preliminary HLLV/TLI Stage VHM System Recommendations
- Identification of Required Near-Term Technologies

HLLV/TLI Stage VHM Trade Study



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IVHM Requirements

FLO IVHM Requirements

FLO System Level Requirements

- For Safety and Mission Critical Functions, the System Shall Have the Capability to Detect and Isolate Failures, Reconfigure to Regain the Function and/or Safe the Failed Function.
- For Failures of Safety and Mission Critical Functions and Hazardous Conditions, Caution and Warning Information Shall Be Provided to the Crew and Mission Support .
- A Common Health Monitoring and Management Architecture Shall Be Used Throughout All FLO Elements

FLO IVHM Requirements

FLO Segment Level Requirements

- The Earth-to-Space Segment Shall Have the Capability to Detect Failures and Provide Real-Time Operational Health Data and Health Monitoring Information of Flight Element Functions to the Crew and Mission Support during all Phases of the Mission.
- The Earth-to-Space and Operational Support Segments Shall Provide the Crew and Mission Support the Capability to Monitor, Control, Over-Ride and Recover from Flight Element Failures Which Are Critical to Safety and Mission Functions .
- For Time-Critical Safety and Mission Functions, the Earth-to-Space Segment Shall Have the Capability to Automatically Isolate Failures, Reconfigure to Regain the Function and/or Safe the Failed Function Without Crew or Ground Support.
- When Communication Is Lost With the Crew or the Ground, the Earth-to-Space Segment Shall Have the Capability to Automatically Isolate Failures, Reconfigure to Regain the Function and/or Safe the Failed Function Without Crew or Ground Support.
- The Earth-to-Space Health Monitoring and Management System Shall Be At Same Level of Fault Tolerance As the Operational System It Is Monitoring and/or Controlling.

FLO IVHM Goals

- VHM Will Be Incorporated Into FLO Elements for Non-Critical Safety and Mission Functions Only If It Offers Significant Safety, Mission Success, Reliability or Cost Benefits .
- VHM Will Be Optimized for the FLO System and Take Advantage of Synergism Between Elements.
- Common Health Monitoring and Management Processes, Procedures, Hardware, Software and Database Will Be Used for System Assembly, Processing, Checkout, Maintenance, Flight Operations and Post Flight Analysis.
- Fault Detection, Isolation, Reconfiguration and Recovery Functions Will Be Performed at Lowest Practical Level Within Each FLO Element.
- VHM Functions Will Be Performed Automatically on Board FLO Elements When It Is More Cost Effective than Performing These Functions on the Ground.
- FLO VHM Will Use State-of-the-Art Technology (TRL Of 6 by 1995) to Reduce Schedule Risk and/or Program Cost. However, Advanced VHM technology (TRL Less Than 6 by 1995) Will Be Considered If It Offers Major Safety, Mission Success or Cost Benefits.
- FLO VHM Will Have the Capability to Evolve to Meet Mars Mission Requirements.

FLO Vehicle Health Management Groundrules

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- FLO VHM Will Not Be Extensive Due to Cost, Schedule or Need, but Will Provide a Proving Ground for the Evolution to Mars
- FLO VHM Will Provide a Capability to Detect/Predict, Isolate and Recover from Vehicle Faults
- Fully Automated VHM, with an Override Capability Is the Long-Term Goal
- To Support Manned Flight the Vehicle Must Be Fail Operational/Fail Safe
  - The Baseline HLLV/TLI Stage Design Will Provide a Fail Safe Capability
  - Study Will Identify Options where the Lander/Ascent System Can Provide the Fail Safe Capability
- State-of-the-Art or Very Near-Term Advanced Technology (1995) Will Be Baselined
- VHM Will Be Optimized for the Entire HLLV-TLI Stage System
- Study Will Identify Concepts which are Synergistic with the Lander & Ascent Vehicles
- The VHM Function Will Be Allocated to the Avionics Subsystems

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IVHM Architecture

Electronics

Ground Processing

Power

Software

Propulsion

IVHM Assessment

Technology Features

Comparisons

Benefits Analysis

Recommended Technologies

Assessment Methodology

- Divide Technologies into Subsystems
  - IVHM Architecture
  - Electronics
  - Ground Processing
  - Power
  - Software
  - Propulsion

Identify Current Methods of Checkout and Test, and Enhanced IVHM Concepts

Perform Comparisons and Assess Risk, Reliability and Safety Aspects of Each Technology or Improvement

Use Cost Benefits Analysis Considering Life Cycle of Vehicle and Number of Potential Missions

Select Recommended Technologies

Automated Health Management Technologies

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	Current Flight-Proven Technology	State of the Art Technology	Advanced Near-Term Technology
Condition Monitoring	• Data Recording/Formatting • Time Stamping • Threshold Detection (red lines) • <u>Limited</u> Data Qualification • Signal Processing	• Data Recording/Formatting • Time Stamping • Threshold Detection (red lines) • Data Qualification • Signal Processing	• Data Recording/Formatting • Time Stamping • Threshold Detection (red lines) • Data Qualification • Signal Processing
Diagnostics	• Fault Anomaly Detection • <u>Limited</u> Time Correlation • • •	• Fault Anomaly Detection • Time Correlation • <u>Limited</u> Pattern Recognition • Data Selection • Feature Extraction	• Fault Anomaly Detection • Time Correlation • Pattern Recognition • Data Selection • Feature Extraction
Decision Making	• • <u>Limited</u> Alt. Function Selection • • • <u>Limited</u> System Reconfiguration	• • <u>Limited</u> Failure Mode Selection • Alternative Function Selection • Component Evaluation • <u>Limited</u> System Reconfiguration	• Situation Assessment • Failure Mode Selection • Alternative Function Selection • Component Evaluation • System Reconfiguration
Prognostics	• • • • •	• <u>Limited</u> Component Life Assessment • <u>Limited</u> Trend Analysis • <u>Limited</u> State Estimation • • <u>Limited</u> Maintenance Scheduling	• Component Life Assessment • Trend Analysis • State Estimation • Mission Assessment • Maintenance Scheduling

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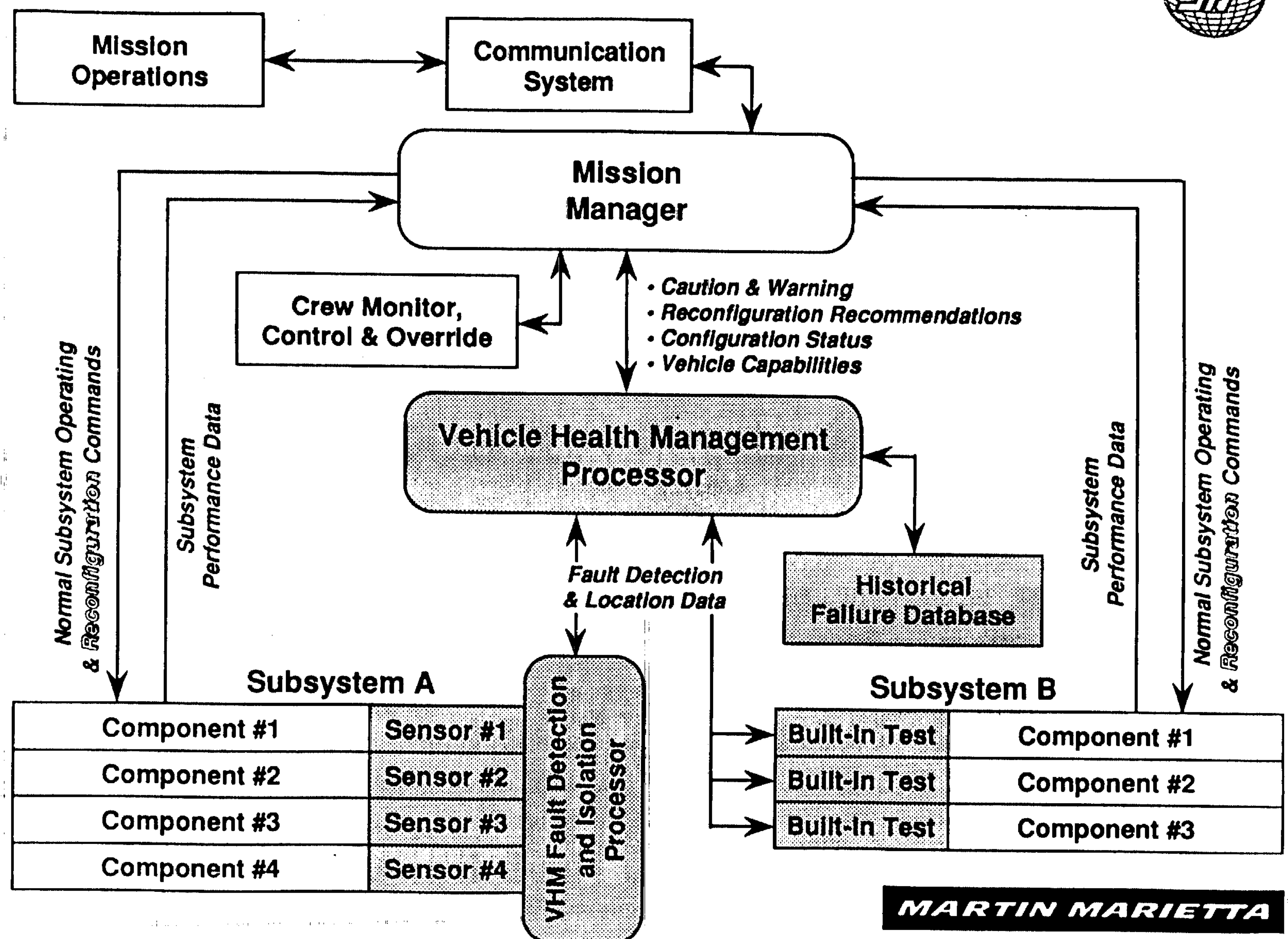
IVHM Architecture Example



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VHM Data Rates

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Level	Data Rates	Vehicle Interrogation	Vehicle Status
Mission Level	10 Kbps	• Vehicle Status Interrogation	• Go/No-Go Assessment
Vehicle Level	100 Kbps	• System Status Interrogation • System Level Arbitration & Fault Isolation	• Vehicle Configuration Status • Confidence Level
System Level	1 Mbps	• Subsystem Status Interrogation • Subsystem Level Arbitration & Fault Isolation	• Subsystem Level H&S • System Configuration Status
Subsystem Level	10 Mbps	• LRU Commanded BIT • LRU/Smart Sensor Fault Isolation	• Subsystem H&S • Subsystem Configuration
LRU Level	20 Mbps	• Component Level Interrogation & Fault Isolation • Component Level Fault Isolation	• LRU/Smart Sensor H&S • LRU/Sensor Configuration
Component Level	40 Mbps	• Gate Level Fault Detection & Isolation	• Component/Sensor H&S

Distributed Health Management Processing Provides Higher Fault Coverage, Faster Data Throughput, and Reduces Data/Processing Load at the Mission and Vehicle Levels

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IVHM Architecture

Electronics

Ground Processing

Power

Software

Propulsion

IVHM Assessment

Technology Features

Comparisons

Benefits Analysis

Recommended Technologies

IVHM Electrical/Electronic Technologies

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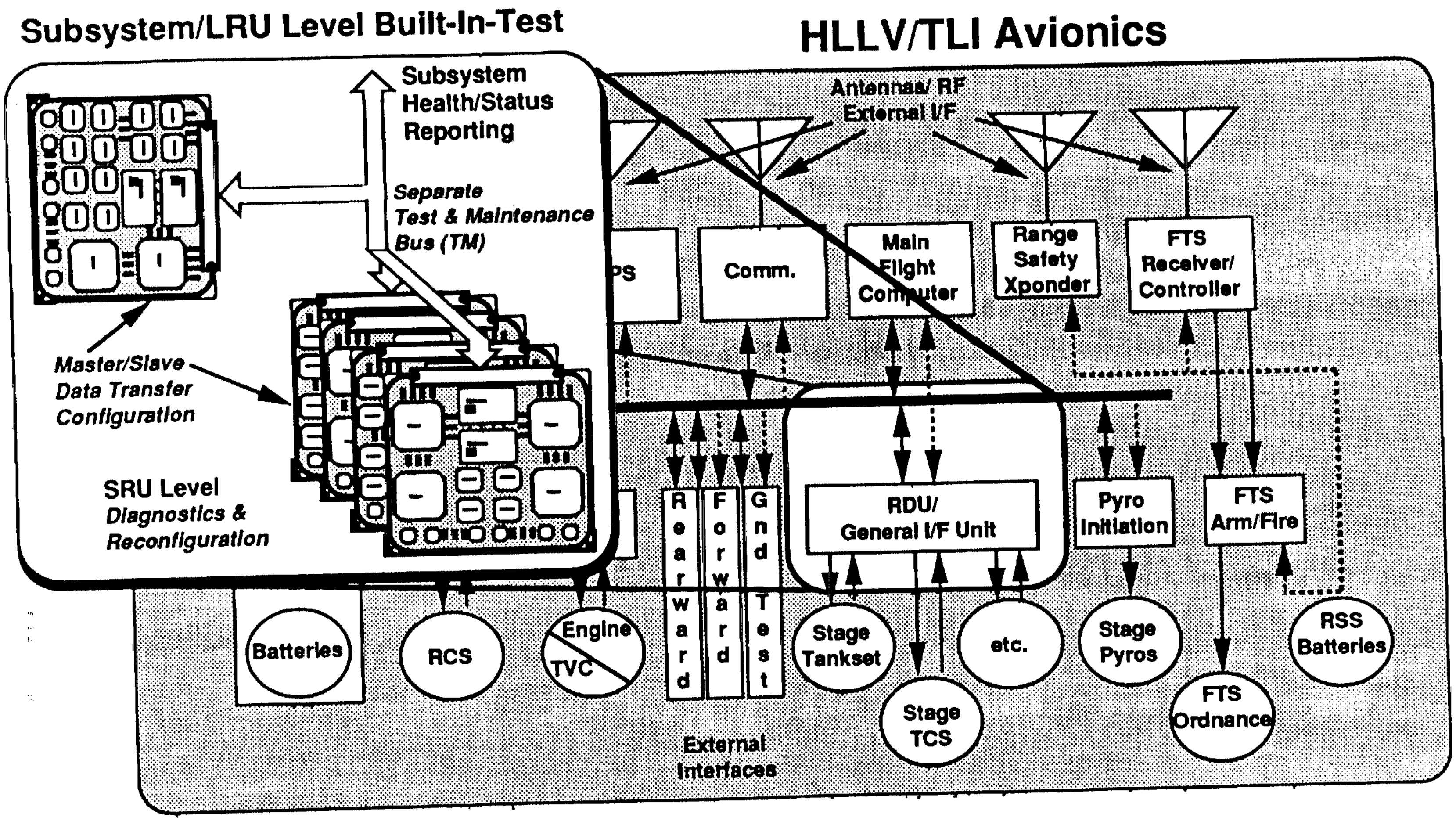
- Electronic Requirements Include Automated Data Management and Design for Test (DFT) Technologies

Current Flight-Proven Technologies	State of the Art Technologies	Advanced Near-Term Technologies
Test: • Micro-diagnostics • Card-edge Testpoints • Deterministic Test Vectors • Visual Indicators (Bit-balls, LEDs, Meters, etc.)	• Boundary Scan Design • Internal Scan Access Ports • Pseudorandom Test Vectors • Direct Access Test Interface • Backplane Test Bus (TM) • On-Chip ASIC Testability Architecture • Signature Analysis • FPGA Test Logic	• Analog Test Bus • Memory Cell Management • Microelectronic Sensors • Optically Coupled Mech. Sensors
Test Algorithm, Vector and Data Base Storage: • Tape Storage and Uplinked Data Loads • Semiconductor Mass Memory Storage	• Non-Volatile RAM (NVRAM) • R/W Optical Disk Storage	

-Recommended Technologies

Subsystem/LRU Level Built-In-Test



LRU- Line Replaceable Unit
SRU- Shop Replaceable Unit

Subsystem/LRU Level Built-In-Test cont'd

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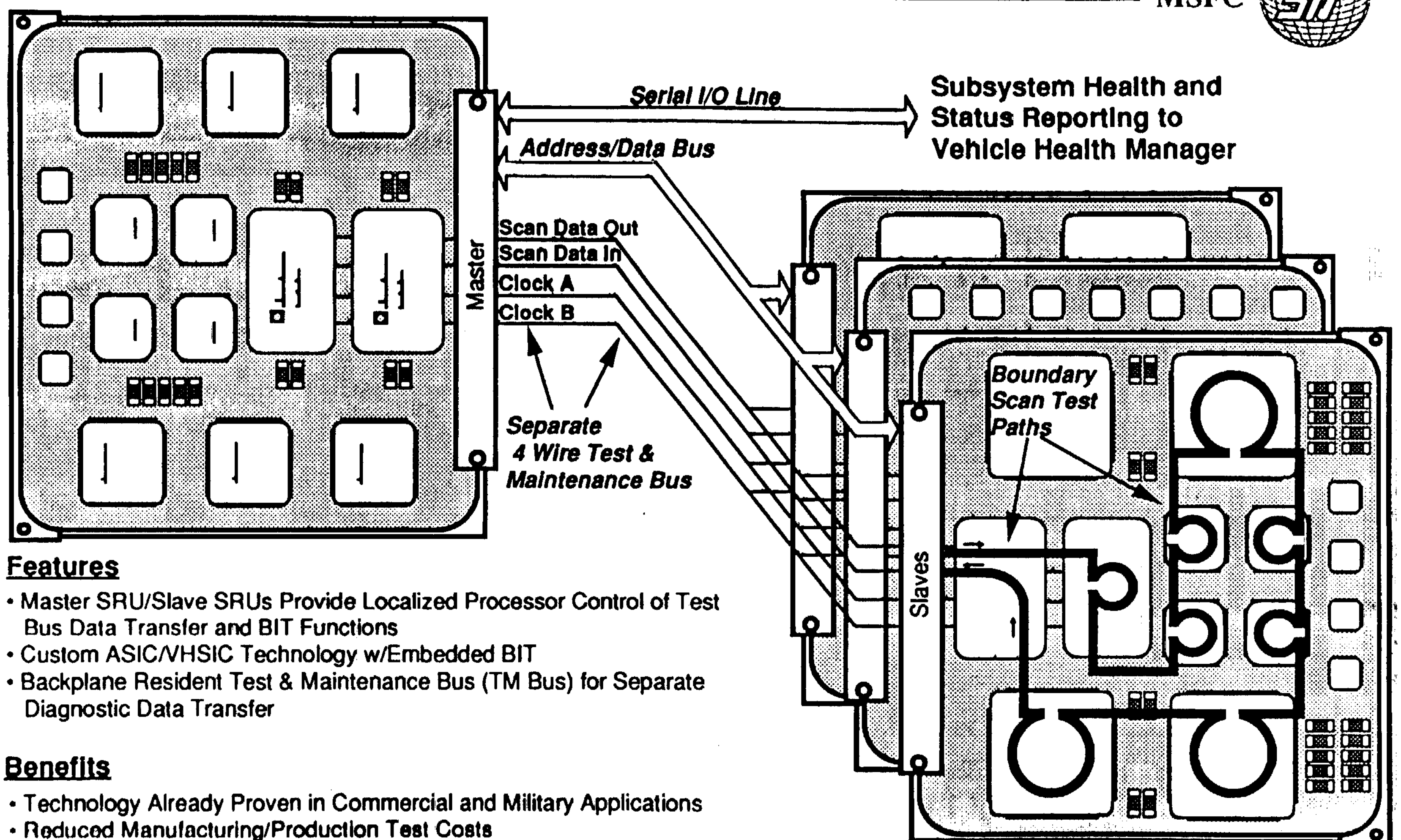

Features

- Master SRU/Slave SRUs Provide Localized Processor Control of Test Bus Data Transfer and BIT Functions
- Custom ASIC/VHSIC Technology w/Embedded BIT
- Backplane Resident Test & Maintenance Bus (TM Bus) for Separate Diagnostic Data Transfer

Benefits

- Technology Already Proven in Commercial and Military Applications
- Reduced Manufacturing/Production Test Costs
- Faster Checkout and Test
- Higher Fault Detection and Isolation Coverage
- Net Reliability Improvement
- ASIC/VSHIC Technology Results in Net Weight Reduction
- Separate Test & Maintenance Bus (TM Bus) Provides High Speed, Non-Intrusive access to all Circuitry (IEEE Supported)

* SRU - Shop Replaceable Unit
* LRU - Line Replaceable Unit
* ASIC - Application-Specific Integrated Circuit
* VHSIC - Very High Speed Integrated Circuit

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SRU/Sensor Health Management

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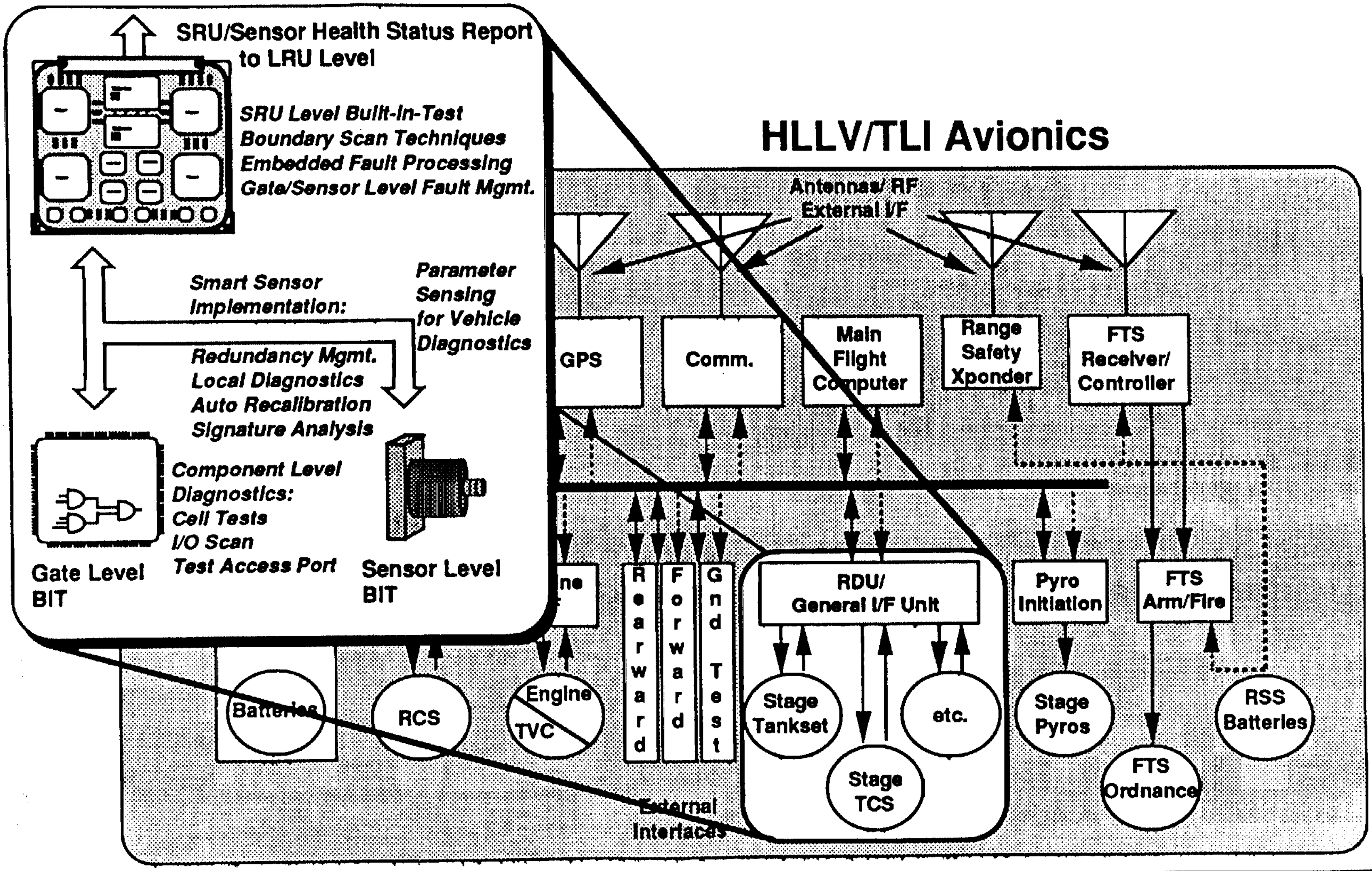

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SRU/Sensor Health Management cont'd

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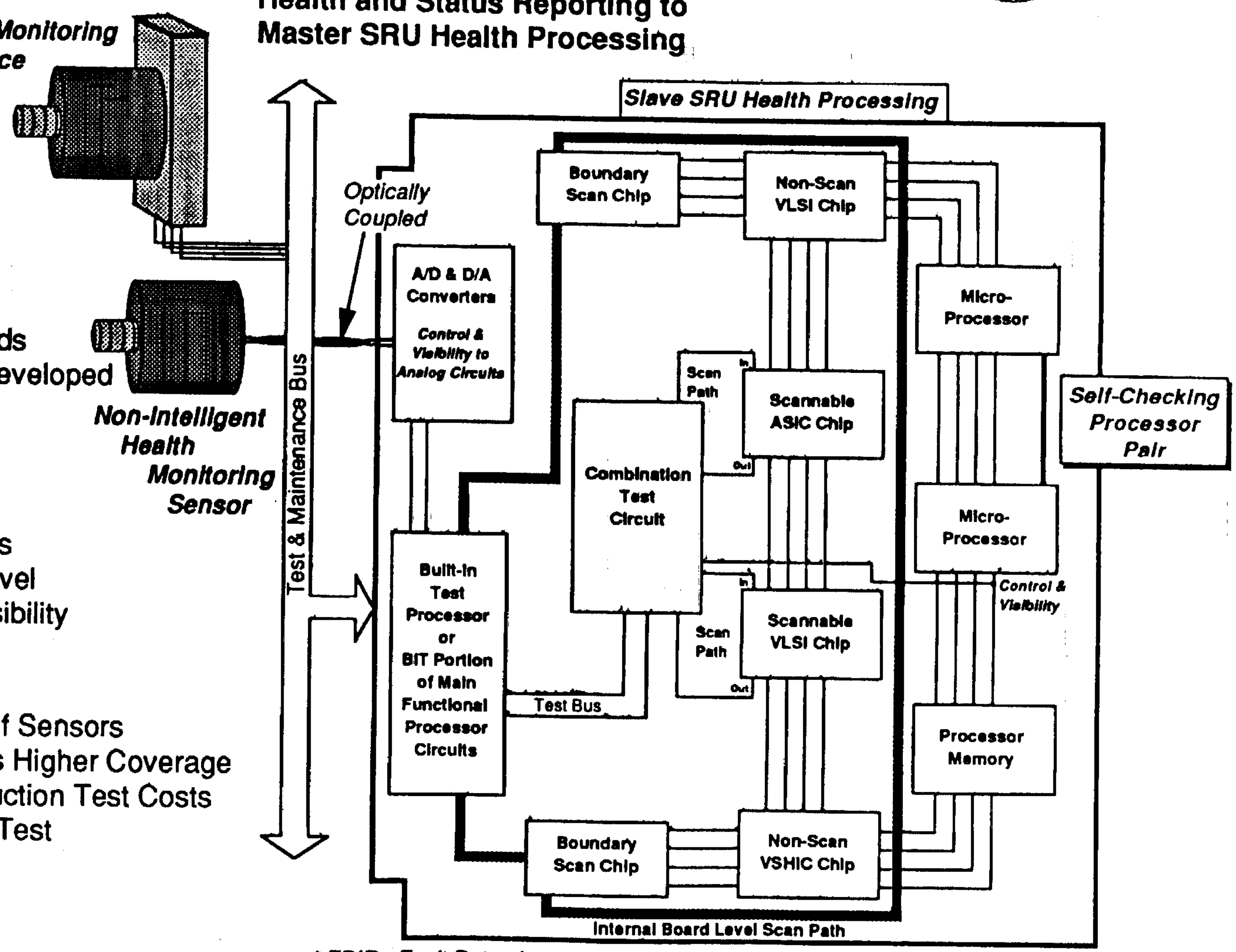

Features
- SRU Level Built-in-Test
- Self-Checking Processors
- Memory Cell Tests
- Device I/O Scan
- Boundary Scan Technology
  - Supported by IEEE Standards
  - Commercial/DOD Aircraft Developed
- Intelligent Sensor Processing
  - Auto-Recalibration
  - Validation
  - Data Fusion
- Redundancy Mgmt. of Sensors
- Fault Management to Gate Level
- Analog Circuit Control And Visibility

Benefits
- Smart/Efficient Management of Sensors
- Self-Checking µProc. Provides Higher Coverage
- Reduced Manufacturing/Production Test Costs
- Faster On-Pad Checkout and Test
- Improved SRU/Sensor FDIR
- Non-Intrusive FDIR
- Lower False Alarm rate

* FDIR - Fault Detection, Isolation and Recovery

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Benefits Analysis

Electronic Subsystems

Design/ Development	Production	Integration/Test	Flight	Post Mission Analysis
Cost: Net Increase in Cost as a Result of Extra Design Time for BIT Circuitry Reduced Development Test Time	**Cost:** Slight Increase in Production Costs Due to Addition of BIT Hardware, Overall Net Decrease in Cost as a Result of Reduced Checkout and Test Time Throughout the Production Cycle	**Cost:** Reduced Cost Due Improved Checkout and Test Capabilities		**Cost:** Increase in Available Health Data from Mission, Results in Cost Savings; Reduced Standdown Time, in the Event of Mission Failure, Results in Substantial Costs Savings
Reliability: Increase in Equipment Failure Rate as a Result of Additional Hardware; Net Increase in Reliability/Availability resulting from BIT and Added Redundancy		**Availability:** Increase in Launch Availability Due to Improved Checkout and Test	**Reliability/Safety:** Reliability and Safety Improved due to the Increased Fault Coverage	

Electronic Subsystems VHM Cost Analysis

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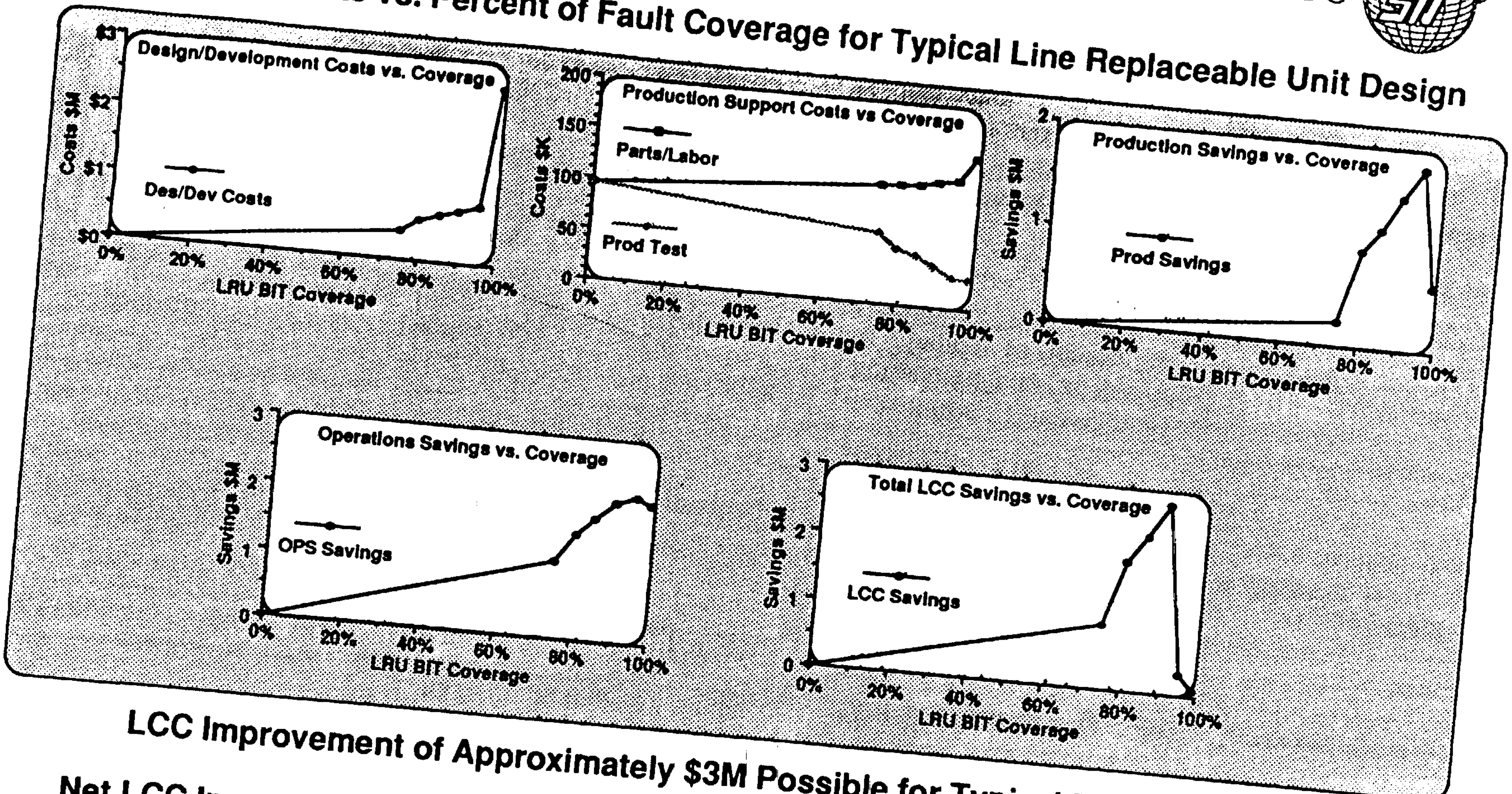
Life Cycle Costs vs. Percent of Fault Coverage for Typical Line Replaceable Unit Design



LCC Improvement of Approximately $3M Possible for Typical Electronic LRU

Net LCC Improvement for Avionics System Consisting of Multiple Electronic LRUs is Approximately $20-40 Million

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Recommended IVHM Technologies

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Technologies Recommended for Electrical/Electronic Subsystems

Recommended Technologies	Drawbacks Associated w/Technology	Development Requirements	Level of Effort
• Boundary Scan Design • Direct Access Test Interface	Although these are Design Techniques, there are Costs Associated with the Additional Circuitry to Implement them	None; Technology well Developed and used in Both Commercial and Military Applications	< 6 mos. <200K/Design
• Test & Maintenance Bus	Requires Additional Circuitry	Same as Above	Same as Above
• On-Chip ASIC Testability Architecture • FPGA Test Logic • Internal Scan Access Ports • Memory Cell Management • Non-Volatile RAM (NVRAM)	Susceptible to Single Event Upset (SEU) Most Technologies will Require Space Qualification	Technology well Developed and used in Both Commercial and Military Applications Some Designs will Require Radiation Hardening	Application Specific; 2 Yr. to Rad. Hard. & Space Qualify $1-2 Million
• Optically Coupled Mechanical Sensors	Current Technolgy Readiness Level of 4	Accelerated Development Schedule Required to Bring Technology to Appropriate Readiness Level; Will Require Space Qualification	Application Specific; 3 Yr. Development and Test $4-5 Million
• Micro-electronic Sensors	Current Technolgy Readiness Level of 5	Developed and used in Commercial Applications; Will Require Radiation Hardening and Space Qualification	Application Specific; 2 Yr. Development and Test $2-4 Million

FPGA - Field Programmable Gate Array
ASIC - Application-Specific Integrated Circuit

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IVHM Architecture

Electronics

Ground Processing

Power

Software

Propulsion

IVHM Assessment

Technology Features

Comparisons

Benefits Analysis

Recommended Technologies

IVHM Ground Processing Technologies

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- Ground Requirements Include Automated Data and Fault Management, Intelligent/Integrated Expert Support and Advisory Systems

Current Proven Technologies	State of the Art Technologies	Advanced Near-Term Technologies
Procedures: • Partial Automated Checkout • Data Monitoring & Analysis • Off-Line Processing	Procedures: • Automated Checkout & Test • Automated Inspection • Auto. Planning/Scheduling	Procedures: • Launch Decision Support • Autonomous Corrective Action Recommendations • Multiple Advisory Elements
Technologies: • Ground Telemetry Systems • Landline Instrument Systems • Video Monitoring • Standard Test Equipment • CAE Workstations	Technologies: • Fiber Optic Data Links • Knowledge based Support Systems	Technologies: • Robotics • Expert Systems for Launch Support, Commit & Anomaly Resolution

▒ - Recommended Technologies

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Ground Operations

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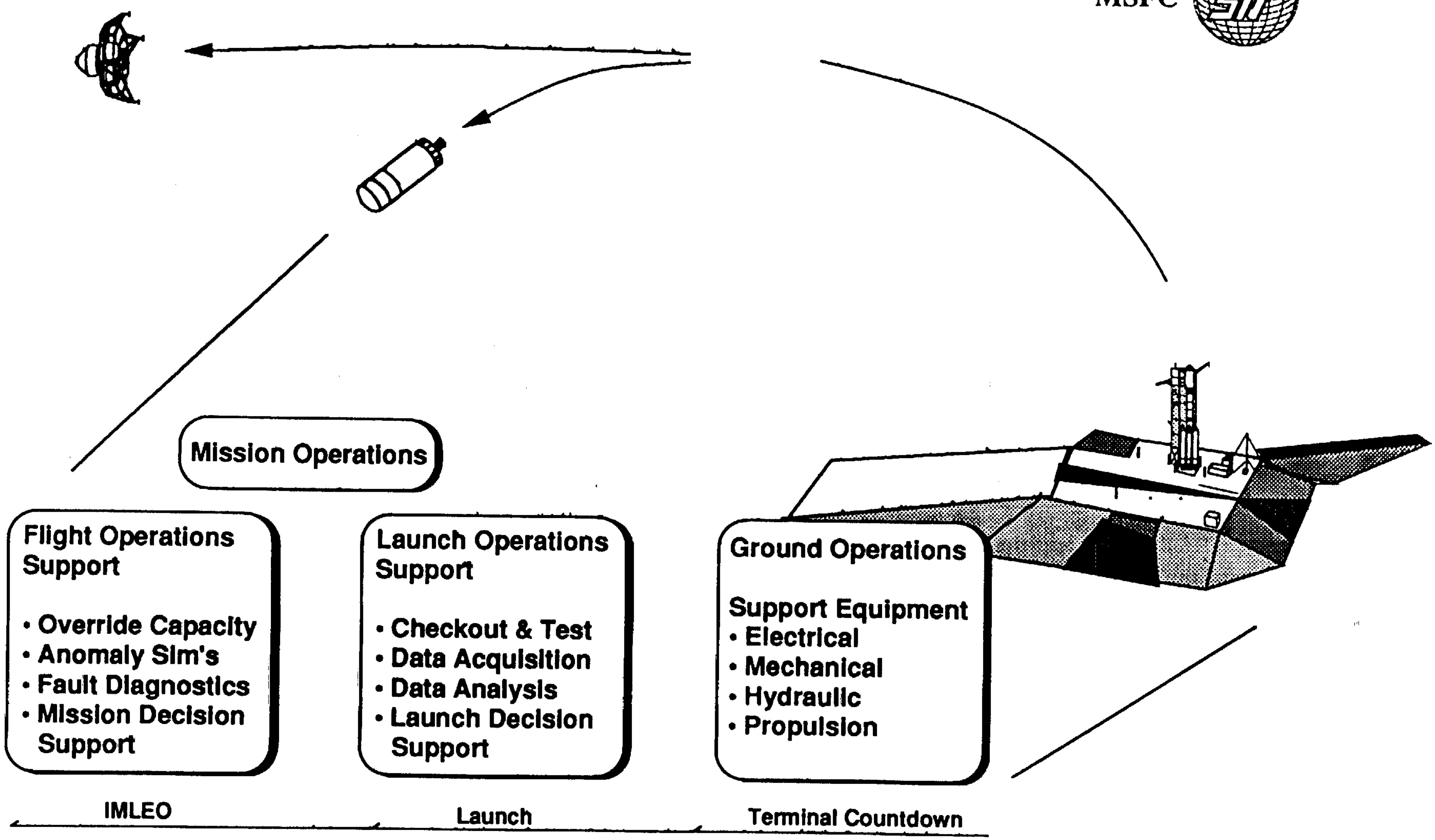

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Automated Ground Support Equipment

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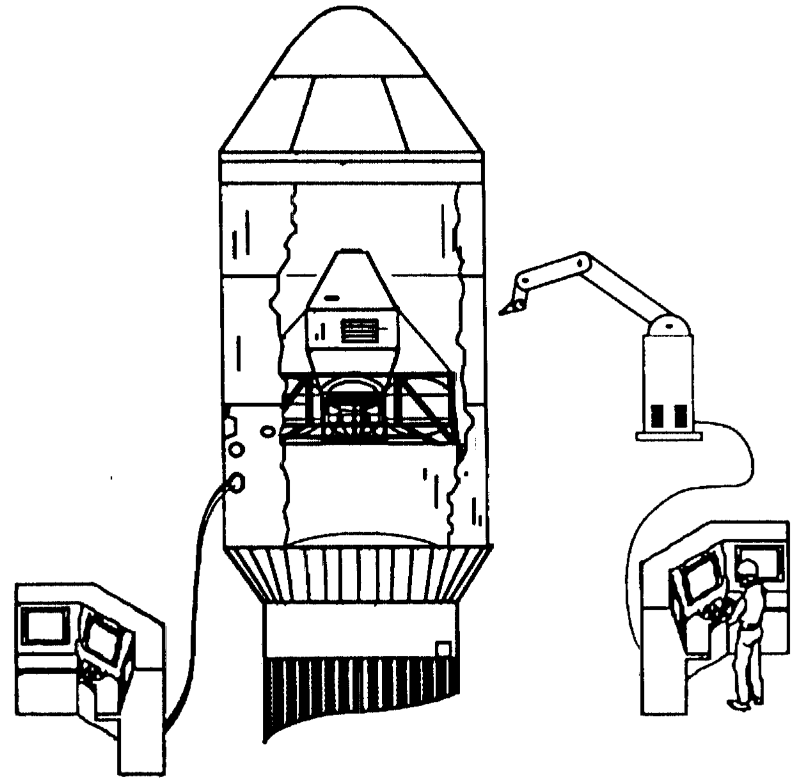
Features

- Robotic Inspection
  - Leak Detection (Hydraulic)
  - X-Ray
  - Tolerance Check
  - Thermal Imaging
- Automated Checkout and Test
  - Automates and Moves More Checkout Activity onto Vehicle
- Automated Propellant Loading
  - Level Sensing
  - Gas/Liquid Leak Detection
- Laser Ordnance Processing
- Fire/Smoke Detection for Electrical Systems



Benefits

- Increased Safety/Reliability
- Reduces On-Stand Time
- Less Hardware Maintenance
- Less Ground-Based Software Maintenance
- Reduction in Launch Site Manpower
- Reduced Costs

IVHM Enhances Automated Ground Support Processes and Equipment

Vehicle Ground Processing Benefits Analysis

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Automated Inspection (Robotics and Leak Detection)

Design/ Development	Production	Integration/Test	Flight	Post Mission Analysis
Cost: Design Costs Associated with Adding and/or Integrating Leak Detection Equipment onto the Vehicle Reduction in Dev. Testing Time	**Cost:** Moderate Cost of Adding Leak Detection Equipment to Vehicle	**Cost:** Significant Cost Savings Result from Less Diagnostics, Reduced On-Stand Time due to Fewer Safety Procedures, and Reduction of Launch Site Personnel	**Cost:** In-Flight Leak Detection Could Prevent the Loss of Vehicle/Mission	**Cost:** Increase in Available Health Data from Mission Reduces Post Mission Analysis; Reduced Standdown Time, in the Event of Mission Failure, Results in Substantial Cost Savings
		Availability: Increased Availability Through Reduced Stand-Down as a Result of Rapid Detection and Isolation of On-Pad Problems	**Reliability/Safety:** Reliability and Safety Improved due to the Increased In-Flight Fault Detection Capability	

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Vehicle Ground Processing Benefits Analysis

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Laser Initiated Ordnance

Design/ Development	Production	Integration/Test	Flight	Post Mission Analysis
Cost: Low; Less Hardware Required to Accomplish Same Operation as Current Methods; Basic Concept already Developed and Flown in Military Applications	**Cost:** Production H/W Costs are Low, Production Test Costs are Low Due to Built-In-Test	**Cost:** The Reduced Processing and Checkout Time, due to the Enhanced Built-In-Test, along with the Reduction in Safety Procedures, Results in a Substantial Cost Savings over Current Design		**Cost:** Increase in Available Health Data from Mission, Results in Cost Savings; Reduced Standdown Time, in the Event of Mission Failure, Results in Substantial Cost Savings
Size: Minimum Amount of Sheilding and No High Voltage Cable Required (ie. smaller size); Large Energy Storage Capacitors Unnecessary (reduced size and weight)	**Safety:** Reduced Safety Requirements	**Reliability/Safety/ Availability:** No Susceptibility to RF, Electrostatic, or Electromagnetic Induced Detonation or Dudding; More Reliable and Testable System will Increase Likelyhood of Launching On-Time	**Safety:** Improved Safety due to EMI Immunity **Availability:** In-Flight Checkout of Ordnance System Increases Confidence in Mission Success	

Vehicle Ground Processing Benefits Analysis

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Automated Checkout and Test

Design/ Development	Production	Integration/Test	Flight	Post Mission Analysis
Cost: Design Cost Increases by as Much as 20% as More Checkout and Test Functions are Automated and Allocated to Vehicle	**Cost:** Production H/W and S/W Cost Increase (Approximately 10%) due to added Vehicle Checkout and Test Capability	**Cost:** Reduced Integration Testing as a Result of Full-Up Functional Test Capability While on the Pad Also Functional Test Capability Provides Higher Confidence in Mission Success		**Cost:** Automated Data Collection from Checkout and Test System Reduces Post Mission Analysis Effort
Reliability: Increase in Component Failure Rate by as Much as 10% Due to Increased Component Count However, Automated Fault Detection Results in Net Reliability Improvement		**Availability:** Vehicle Checkout Time Reduced by as Much as One-Half Due to Enhanced Automated Capability	**Reliability/Safety:** Automated On-Board Checkout Capability Increases Vehicle Mission Success Capability by Detecting and/or Predicting Failures Before a Critical State is Reached	

Automatic Cable Checkout

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Features

- Portable Computerized Test Set
- Tests Continuity of Individual Lines
- Tests Payload/Launch Vehicle Interfaces
- Checks Grounding, Stray Voltage
- Verifies Isolation Between Lines ("Meggar")
- Tracks and Stores Vehicle Configurations

Benefits

- Faster Checkout
- Reduced Manpower
- Higher Fault Coverage
- Reduced Costs
- Greater Confidence in Mission Success

Benefits Analysis

Design/ Development	Production	Integration/Test	Flight	Post-Mission Analysis
Cost: May Require Additional Test Interfaces/ Points on Vehicles	**Cost:** Reduction in Quality Control Inspections and Personnel Test Manhours Reduced for Verifying the Correct Cable Assembly	**Cost:** Reduction in Quality Control Inspections and Personnel **Availability:** Test Time Reduced for Verifying Correct Cable Wiring	**Reliability:** Improved Test Capability Increases Confidence in Mission Success	

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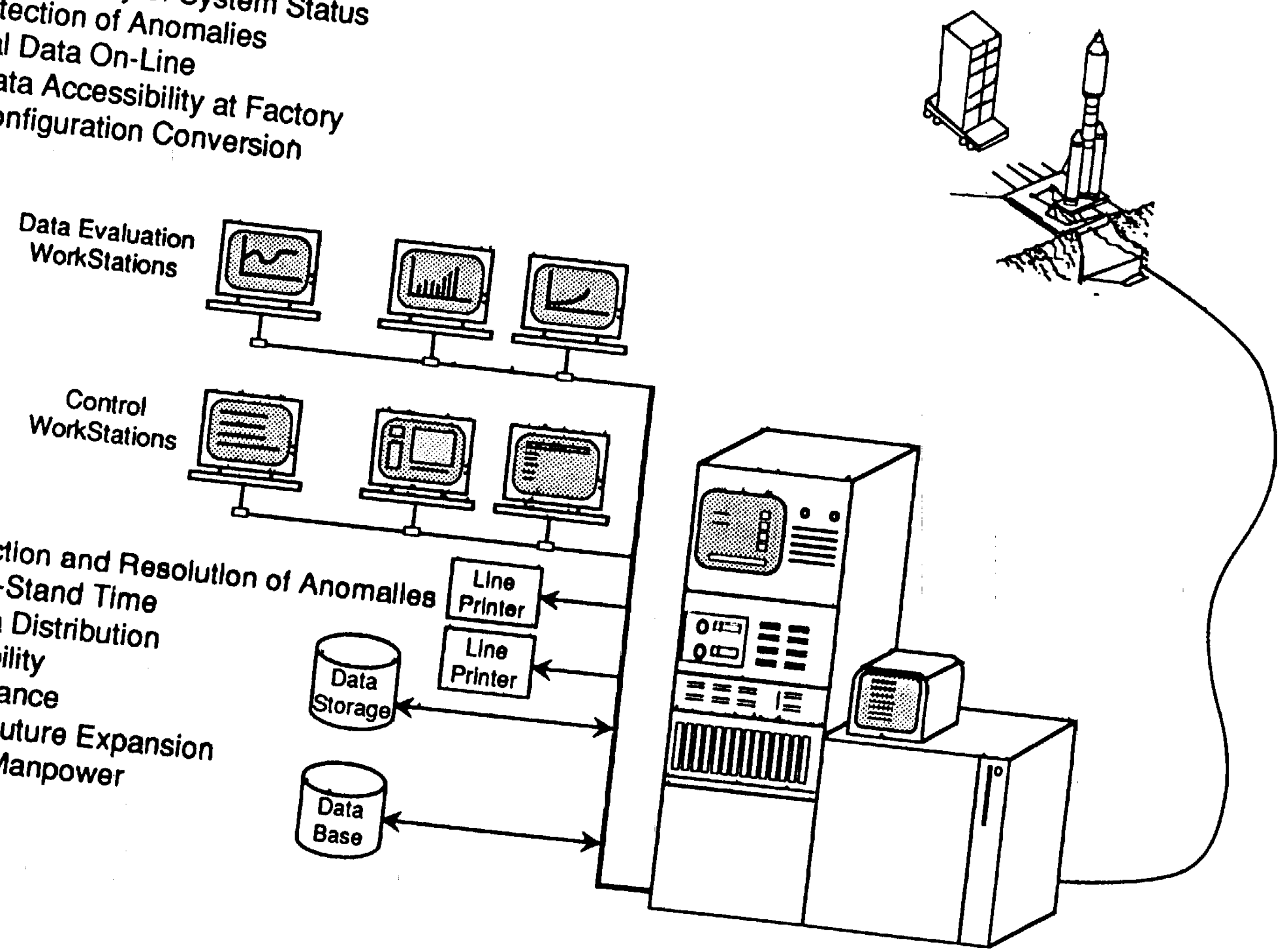
Automated Launch Support Operations

Features
- Fully Computerized Data Acquisition and Graphical Display
- Real Time Display of System Status
- Auto Detection of Anomalies
- Historical Data On-Line
- Rapid Data Accessibility at Factory
- Rapid Configuration Conversion



Benefits
- Faster Detection and Resolution of Anomalies
- Reduces On-Stand Time
- Efficient Data Distribution
- Higher Reliability
- Less Maintenance
- Provides for Future Expansion
- Reduction in Manpower

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Flight Operations Support

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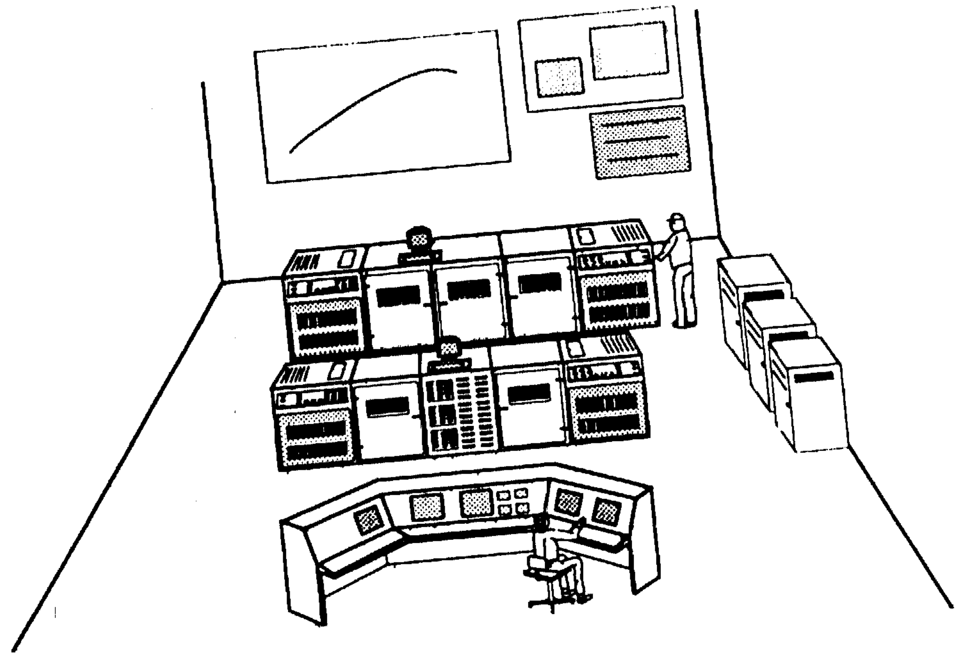
Features

- Advanced Software Systems
- Computerized Data Acquisition and Transfer
- Graphical Display Environments
- Automated Advisory Tools
- Auto Notification of In-Flight Anomalies
- Optimized Allocation of Ground Supported and On-Board Vehicle Health Management Diagnostics
- Historical Data On-Line
- Vehicle and Fault Simulations for Real Time Anomaly Resolution



Benefits

- Faster Resolution of Anomalies
- Efficient Data Distribution
- Less Maintenance
- Reduction in Mission Manpower
- Reduced Costs
- Increased Safety Through Faster Anomaly Resolution

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Launch and Flight Support Operations

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The Benefit Comparison Below Assumes a Modified Shuttle Infrastructure is used for the HLLV/TLI Launch/Flight Support Ops

Current Launch/Flight Ops for STS

Just Prior to Liftoff, Over 300 Personnel Watch Screens and Monitor Shuttle Systems

Four Firing Rooms are Required Each with Approximately 15 Consoles, each Console Typically Having Three Sets of Displays, a Strip Chart Recorder, and Several Hundred Failure Indicator Lights. These Consoles are Monitored By Personnel and Require Cooperation Among Engineers to Make the System Work.

In Addition to the Firing Rooms Several Hundred Engineers and Support Personnel are on Standby or are Performing Background Functions

An Extensive Data Management System is Required to Process all of the Ground Operations Information. Over Seven Million Lines of Computer Code are Needed to Support this Processing with over 490 Man-Years per Year to Maintain.

Potential Launch/Flight Ops for HLLV/TLI

Automation of Launch Support Functions and the use of Expert or Knowledge based Shell Systems Could Result in a Personnel Reduction of Approximately Half. Savings of Over $5 Million Per Launch Could be Realized.

The Number of Firing Rooms Could be Reduced to Eventually One Room Resulting in Several Million Dollars Per Year in Hardware and Maintenance Cost Savings.

Software Maintenance Costs Could be Cut in Half Providing at Least $50 Million in Savings Per Year.

In Addition to Cost Savings, Automated Operations Increases the Quality of Flight Decisions

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Recommended VHM Technologies

Technologies Recommended for Vehicle Ground Processing

Recommended Technologies	Drawbacks Associated w/Technology	Development Requirements	Level of Effort
• Automated Inspection - Robotics - Leak Detection	Development and Implementation Costs Technologies Most Likely to be Ground-Based, but Some could Fly	Most Technologies Well Developed and used in Commercial Industry Some Will Need to be Space Qual. & Ruggedized	Robotics: Application Specific, 2-3 Yrs. $5-20 M Leak Detection: 2 Yr. Development $2-3 Million
• Laser Ordnance Processing	Implementation Cost, Requires Change in Present Ordnance Processing Infrastructure	Similar Technology Dev. and used in Military Some Application Specific Development Required	2 Yr. Test and Demonstrate $2 Million
• Automated Checkout & Test - Automatic Cable Testing	Development and Implementation Costs	Most Technology Rqm'ts Already Developed	2 Yr. Test and Demonstrate $2-4 Million
• Launch Decision Support - Data/Control Workstations - Auto Anomaly Detection/ Resolution	Development and Implementation Costs Cultural Change Required	Development of Advanced Expert Systems Required	2-5 Yr. Development and Demonstrate $20-40 Million
• Flight Operations Support - Multiple Advisory Elements - Automated Diagnostics - Historical Database	Development and Implementation Costs Cutural Change Required	Development of Advanced Expert Systems Required	2-3 Yr. Development and Demonstrate $10-20 Million

- IVHM Architecture
- Electronics
- Ground Processing
- Power
- Software
- Propulsion

IVHM Assessment

- *Technology Features*
- *Comparisons*
- *Benefits Analysis*
- *Recommended Technologies*

IVHM Power Technologies

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- IVHM Power Generation, Distribution and Switching Technologies All Provide Opportunities for Cost and Reliability Improvements

Current Flight-Proven Technologies	State of the Art Technologies	Advanced Near-Term Technologies
Primaries: • Temperature Sensing • Voltage Monitoring • Current Monitoring • Battery Simulation, Manual On-Pad Installation or Remote Activation Due to Short Life	Primaries: • Pre Pad Installation of Long Activated Life Lithium Batteries	Primaries: • Chemical Status Sensing
Distribution: • Automated Performance Tests (Meger, Continuity and DWV) • Simulator Based Fit Checks • Partial Post Mate Continuity C/O • Multiple Isolated Busses	Distribution: • Standard Interface Configuration Automated Performance Tests • Full Post Mate Connector Continuity Checkout • External System Checkout Ports	Distribution: • Build and Integration Time Quality Monitoring • CAD Solid Modeling of Cable Sys. • Standard Protocol External Checkout Ports
Switching/Isolation: • Mechanical Fuses, Switches, Breakers	Switching/Isolation: • Smart Switches; Remote Power Control (RPC) Hybrid µ-circuits	Switching/Isolation • Monolithic Smart Remote Power Control (RPC) Microcircuits

- Recommended Technologies

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Recommended IVHM Technologies

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Technologies Recommended for Electrical Power Subsystem

Recommended Technologies	Drawbacks Associated w/Technology	Development Requirements	Level of Effort
• Temperature Monitoring for Primaries, Control Elements • Voltage, Current Sensing at Primaries, Loads	Additional Circuitry, Processing	None; Continuation of Current Practice	Integration Only
• Use of Long Activated-Life Primaries (Lithium Thyonel Chloride)	Tradeoffs Between Energy and Power Density Passivation "Burn-Off" Prior to Use Load Management Critical	Space Qualified System Available for Centaur 1Q 93 Additional Development Desirable for High Power Density, and for 120 V Systems	Integration Only 1-2 Yr. to Space Qualify 120 V Battery; $1 Mil
• Automated Cabling Performance Tests • Standardized Cable Ports for Performance Test I/F	GSE and Operator Training Costs Requires System Level Standardization, Greater Front End Design Effort	None; Continuation of Current Practice Development of Standard(s)	Develop First Cut Standard as an Adjunct to System Design; $500 k
• Hybrid Microcircuit Remote Power Controllers / Smart Switches	Hybrid RPCs: None Smart Switches Work Best with Highly Accurate Sensing Systems	Hybrid Components in Use in Military Space Systems Smart Switch Systems Are Currently at Technology Level 5	Integration Only ~1 Yr to Space Qualify; $1-2 Mil

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IVHM Architecture

Electronics

Ground Processing

Power

Software

Propulsion

IVHM Assessment

Technology Features

Comparisons

Benefits Analysis

Recommended Technologies

IVHM Software Technologies

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- IVHM Software Techniques for Fault Tolerance, and Vehicle Level IVHM Specific Support Elements

Current Flight-Proven Technologies	State of the Art Technologies	Advanced Near-Term Technologies
• OS Locks and Timeouts • Checksum / CRC Data Integrity and Error Correction Methods	• FPGA / PLD Programmable Hardware for Non-Standard Interfaces • Standard, Open, Commercially Derived Software Platforms	• Multiple Dissimilar Operating System Code Execution Environments
• Multiple Execution-Platform Strings with Voting at Effector • Diverse, Proprietary, Specialty Application Software Platforms	• Partial Automation of Test System Driver S/W Generation, Procedure Development and Test Interface Hardware Design	• Standardized Software Links for Importation of Design Knowlege into Analytical and Test Environments
• Standardized Inter-System Communications Hardware Drivers		
• Partial Linkage of Design Knowlege into Analytical and Test Environments	• Automated Vehicle Operations Status and Tracking Software • Mission Time Management Decision Support Software	• On-Board Vehicle Item-Level Status and Relationship Model • Allocation of Voting to Self-Monitoring Multi-Platform Processing System • Mechanism / Item Operational Signature Analysis, Failure / Remaining Life Prediction Algorithms • Supervised Autonomy
• Mission Management Software for Decommutation,Reduction, and Disply of Vehicle TLM • Post Mission TLM Reduction and Presentation Software		

▨ - Recommended Technologies

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IVHM Software Technologies

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- Specific IVHM Applications to Software Development, Validation, and Maintenance

Current Flight-Proven Technologies	State of the Art Technologies	Advanced Near-Term Technologies
Design: • Structured Analysis / Design	**Design:**	**Design:** • Object Oriented Analysis and Design • Automated Code Generation From Analysis Environment Knowledge Capture
Development: • In-Circuit Emulation of Target Platforms • Target Platform Based Code Development Environment	**Development:** • Code Development Frames • Target Independent Common Code Development Environment • Standardized Inter-System Information Transportation, Presentation and Format Protocols	**Development:** • Block Reuse Frames from Existing Validated Code Sources • Adaptive, Independent Internode Communications Networks
Verification: • Unit Test / Formal Qualification Test Validation • N-Version Software Validation	**Verification:** • Partial Automation of Code Documentation	**Verification:** • Application of Formal (Proof) Methods • Fully Automated Code Documentation and Validation Procedure Development

- Recommended Technologies

IVHM Software Applications

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Software Product Support		Software Process Support	
Product Functions	Product Attributes	Process Functions	Process Attributes
• Automated FDIR • Fault Tolerance • S/W Development • Validation • Maintenance	• Object Oriented Analysis • Auto Code Generation • Distributed Parallel S/W • High Assurance S/W • N-Version S/W	*Process Tools* • Rqmts Engineering • Design • Development • Qualification • Product Support	*Process Mgmt. Tools* • Cost Estimations • Planning & Control • QA • Risk Assurance
Reuse Libraries			
Software Engineering/Reengineering Principles, Languages and Frameworks			
Systems Software: Compilers, Operating Systems, Database Management Systems, Object Management Systems, User Interface Management Systems, Instrumentation			

Features
- Time-Critical FDIR
- Trending, Analysis & Prognosis
- Process Tools for Development and Management of S/W
- Modular Reusable Elements

Benefits
- Higher Fault Coverage
- Real-Time Decision Capability
- Effective Human-Machine Interfaces
- Reliable Software
- Affordable Software

Software Constitutes Approximately 70% of a Health Management System, therefore Emphasis is Placed on Doing it Faster, Cheaper and Better

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Recommended IVHM Technologies

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Technologies Recommended for S/W Fault Tolerance and IVHM Support

Recommended Technologies	Drawbacks Associated w/Technology	Development Requirements	Level of Effort
• Conventional Operating System Watch Dog Functions	Increases Memory Space Demanded by Operating System, Reduces Speed	None; Continuation of Current Practice	Integration Only
• Conventional Error Detection and Correction Techniques (CRC, Checksums, Etc.)	Require Increased memory Space, Lengthen Access and Message Turnaround Times, Additional Complexity	None; Continuation of Current Practice	Integration Only
• Standardized Inter-Node "Reliable Communications" Software	Some Questions of Determinism in such Systems Remain	Flight Critical / Manned Systems Apps of Distributed Systems Management	1-2 Yr., $2-3 M
• TLM Decommutation, Data Reduction, Information Presentation SW	Currently Requires Extensive Hardware Simulation for High Fidelity Input During Test, Extensive Custom S/W Each Mission	None; Continuation of Current Practice	Integration Only
• Mission Time Management Decision Support S/W	Extension of Previous Bullet	Extension of Current Practice to Automate Mission Timeline Replanning, "Malfs"	S/W Development for ETO Apps., On-Orbit Checks; 2 Yr., ~$2-3M

Recommended IVHM Technologies

MSFC

Technologies Recommended for S/W Fault Tolerance and IVHM Support (Cont.)

Recommended Technologies	Drawbacks Associated w/Technology	Development Requirements	Level of Effort
• Programmable Hardware (FPGA / PLD) Non-Standard to Standard I/F Information Conversion	Current Hardware Components Operate Slowly Compared with High Bandwidth Network Rates	In Use for Commercial, Military Systems; Technology Level 4-5	1-2 Yr. To Cover ETO / US Interface Applications; ~$2M
• Standard, Open, Commercially Derived S/W Platforms (Hardware, Operating Systems, Development Systems)	Large Systems May Need Extensive Test and Adaptation To Conform to NASA Requirements	Extension of Adaptation of '386 µP, FDDI and ASCM Elements to OS, Development Environments	2-3 Yr. for Selection, Test and Adaptation ~$5M
• Partial Automation of Test System Driver S/W & Test I/F Hardware Design	Level of Automation Possible Is Inversely Proportional to the Variability of Test Activity	Extensively Used for Simple System Performance Tests, e.g. Cabling Evaluation	Integration Only
• Automated Vehicle Operations Status and Tracking S/W	Requires Distributed System Database Management, Manual Source Data Input	Widely Used in Commercial Manufacturing and Logistics Environment, Military Production and Maintenance	2-3 Yr. of Prototype Development Prior to Initial Integration of Flight: $3-4M

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Recommended IVHM Technologies

Technologies Recommended for Software Development, Validation and Maintenance

Recommended Technologies	Drawbacks Associated w/Technology	Development Requirements	Level of Effort
•Object Oriented Analysis and Design	Emerging Methodology, Requires Extensive Training	Common Use in Industry, Needs Adaptation to NASA	1-3 Yr. for Moderate Size Project: ~$1M
•Automated Code Generation from Analysis Environment Knowledge Capture	Reliability Effects Are Unknown, Systems Are New to Industry	Reliability Analysis of Product Code, Similar to New Compiler Validation	2-3 Yr. N-Version Comparison with Existing Codes:$2M
• In-Circuit Emulation of Target Platforms	Development Support Equipment Costs, Emulation Reliability	None; Continuation of Current Practice	Integration Only
• Target Independent Code Development Environments	Development Environment Validation, Maintenance	Development Required To Standardize Environments Used	1-2 Yr. To Describe Requirements, Select among Alternatives
• Standardized Intersystem Information Exchange Protocols	Typically Requires Higher Information Exchange Media Bandwidth	CCSCDS Standards Need To Be Extended to FDDI-Based Systems	1 Yr., $1M for Adaptation Work; ? Yr for Acceptance
• Block Reuse Frames from Existing Validated Code Sources	Emerging Practice	Approximately Technology Level 3	4-5 Yr. Validation of Concept, >$5M

Recommended IVHM Technologies

MSFC

Technologies Recommended for S/W Development, Validation and Maint. (Cont.)

Recommended Technologies	Drawbacks Associated w/Technology	Development Requirements	Level of Effort
• Conventional Unit Test and Formal Qualification Test Validation Sequence	Typically Tests Software with Only Limited Set of Inputs and Conditions	None; Continuation of Current Practice	Integration Only
• N-Version Software Development and Cross-Version Testing	Multiple Independent Development Projects and Extensive Test Cycles Are Expensive	Adaptation, Standardization and Acceptance at NASA	2-3 Yr. Adaptation, Standardization and Evolution: $2-3M
• Partial Automation of Code Documentation	Primarily Documentation Aid for Compliance with MIL-S-2167; Requires Human Analysis Input, Review and Oversite	Currently in Use in Commercial Environment, and on Some Military Projects; Technology Level 5-6	1 Yr. to Adapt to NASA Specific Requirements; ~$2M
• Application of Formal Proof Methods	Applications to Software Constructs Often Difficult Fits; Base of Expertise Is Small or Non-existent in NASA	Currently in Use in Some Commercial Mission Critical Applications; Technology Level 4-5	2-3 Yr. Development for Conventional ETO / Upper Stage Software Applications; ~$5M

IVHM Architecture

Electronics

Ground Processing

Power

Software

Propulsion

IVHM Assessment

Technology Features

Comparisons

Benefits Analysis

Recommended Technologies

VHM Propulsion Technologies

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- Propulsion Requirements Include Sensor and Sensor Management Technologies

Current Flight-Proven Technologies	State of the Art Technologies	Advanced Near-Term Technologies
Sensors:		
• Thermocouples	• Capacitive Pressure	• Plume Spectroscopy
• Eddy Current	• Capacitive Blade Clearance	• Holographic Leak Detect.
• Inductive Debris Monitoring	• Silicon-on-Sapphire Pressure	• Mass Spect. Leak Detect.
• Limit Switches	• Silicon-on-Insulator Pressure	• IR Absorption Leak Det.
• Accelerometers	• Fiber Optic Twin Core Strain	• Solid State Leak Detection
• Resistance Temperature Detector	• Fiber Optic Deflector (Shaft)	• Ultrasonic Flowmeter
• Kistler Pressure	• Fiber Optic Laser Vibration	• Hydrogen Mass Flowmeter
• LVDT/RVDT (Position)	• Radioscope Wear	• Surface Layer Activation
• LOx Mass Flowmeter	• Optical Pyrometer (Temperature)	• Acoustic Emission (Bearing)
	• Laser Blade Tip Clearance	
Sensor Management:		
• Data Compression	• Sensor Bus	• Sensor Failure ID Algorithms
• Data Transmission	• Data Storage	• Redundant Sensor Algorithms
• Signal Processing	• Engine Control Modeling	• Parameter Estimation
• Safety/Redline Assessments	• Fault Response Library	• Health Assessment Algorithms

▒ - Recommended Technologies

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IVHM Propulsion Design Issues

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- What Parameters Can be Sensed?
  - Existing Hardware
  - Current Technology
  - Advanced Technolgy
- Which Parameters Should Be Sensed?
  - What Benefit Can be Derived from a Sensed Parameter?
  - What Is the "Cost" to Sense a Parameter?
- How Frequently Should the Parameter be Sensed?
- At What Resolution Should the Parameter be Sensed?
- Where should the Data on the Parameter be Sent?

Parameter	IVHM Objectives Supported				
	Safety	Mission Accomplishment	Failure Analysis	Re-Use	Ground Ops
Pressure					
Chamber	●	●	●	●	
Pump Discharge		●	●		
Pump Inlet		●	●		
Propellant Tanks		●	●		
Helium Spheres		●	●		
Temperature					
Propellant			●		
Turbine Blades		●	●		
Vibration	●	●	●	●	
Wear				●	
Leaks			●		●
RPM		●	●	●	●
Torque			●		
Erosion		●	●		

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Parameter Sensing Selection Process

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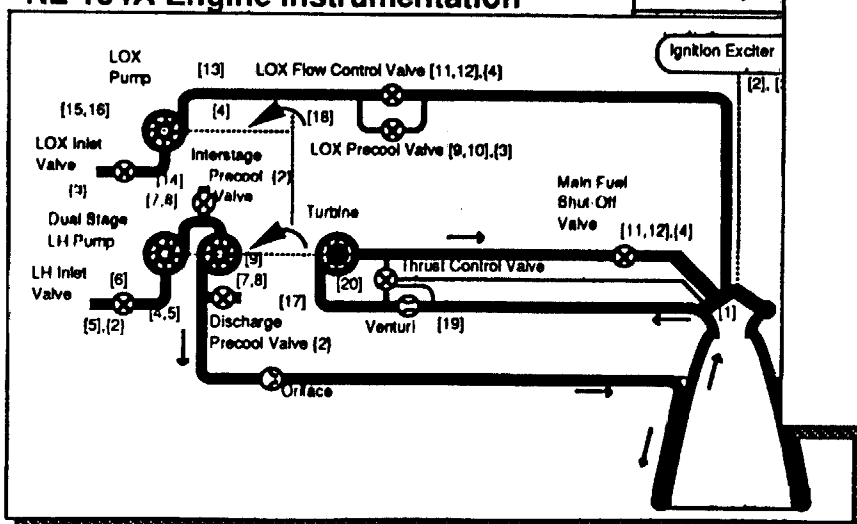
HLLV/TLI Stage Subsystem Criticality

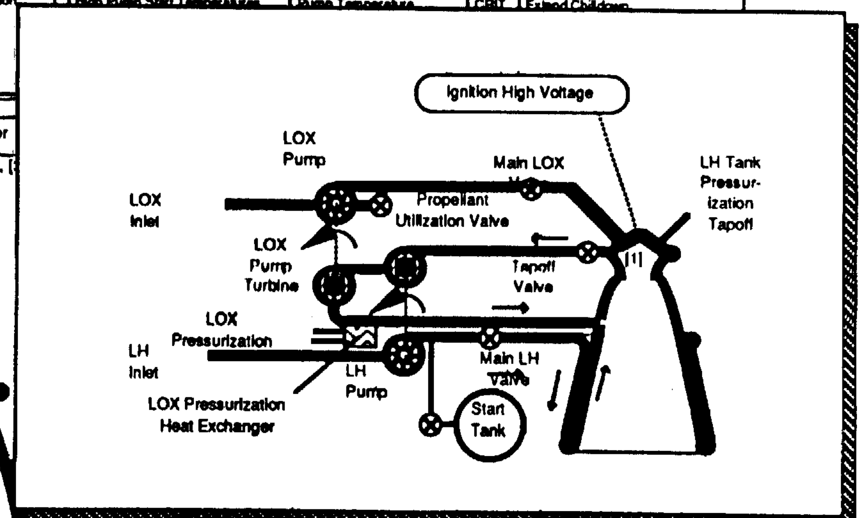
Mission Phase:	Propulsion: HLLV M.P.	TLI M.P.	RCS	Avionics: GN&C	Mission Mgmt	EPDS	Comm.	FTS
0) Prelaunch	N/C	N/C	N/C	N/C	CRIT	N/C	N/C	N/C
1) Core Ignition	CRIT	N/C	N/C	CRIT	R/T Crit	R/T Crit		
2) Liftoff/Boost	R/T Crit	N/C	N/C	R/T Crit	R/T Crit	R/T Crit		
3) Booster Staging	R/T Crit	N/C	N/C	R/T Crit	R/T Crit	R/T Crit		
4) Core Staging	CRIT	R/T Crit	R/T Crit	R/T Crit	R/T Crit	R/T Crit		
5) Suborbital Burn	N/C	R/T Crit	R/T Crit*	R/T Crit	R/T Crit	R/T Crit		
6) Coast	N/C	N/C	CRIT	CRIT	R/T Crit	CRIT		
7) TLI Burn	N/C	R/T Crit	R/T Crit*	R/T Crit	R/T Crit	R/T Crit		

HLLV Main Propulsion Critical Elements

Element Identifier	Failure Modes/Conditions	Monitors	Crit.	Response (Core Ignition Phase)
Internal Elements: Tanks - Fuel, Oxidizer	• Structural Failure • Leaks	None Tank Pressure	NC	Non-Credible Failure
Pressurization - High Pressure Source, Isolation, Press. Reg. Vents Feedlines Thermal Control (SOFI)	• Tank Overpressurization • Inadequate Turbopump NPSH • Fail Open, Closed • Leaks, Blockage • Cracking, Adhesion Loss	Tank(s) Pressure(s) Tank(s) Pressure(s) Tank(s) Pressure(s) Flowmeter, Diff. Press. Visual Inspection	CRIT CRIT CRIT CRIT	Partially Vent Tank(s) Halt Ignition Sequence Abort Ignition Abort Ignition
Main Engines - Main Fuel, Ox Valves, Purge Valves, Propellant Utilization, Throttling, Turbine(s), Pumps, Seals, Bearings	• Valves Fail Fully or Partly Open • Valves Fail Fully Closed • High Pump Start Temperatures	Valve Position, Press. Valve Position, Press. Pump Temperature	CRIT CRIT CRIT	Red. Valve Ctrl, Abort Ignition Abort Ignition Sequence Extend Chilldown

RL-104A Engine Instrumentation





J-2S Available Engine Instrumentation

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Engine Instrumentation

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Parameter	Current Sensing Method	Advanced Sensing	Potential Benefits of Advanced Sensors
Pressure Chamber Pump Discharge Pump Inlet Propellant Tanks Helium Spheres	Kistler Pressure	Capacitive SOS/SOI	Small, Lightweight, Smart, Easy to Calibrate
Temperature - Propellant - Turbine Blades	Thermocouple N/A	No Change Optical Pyrometer	N/A Enabling Technology
Vibration	Accelerometers	Acoustic Emission	Higher Fault Coverage, Potential Fault Prognosis
Wear	Post-Test Disassembly	Isotope	Rapid, No Disassembly Required
Leaks	Soap Solution/Bubble Helium Sniffer	IR Absorption Holography Acoustic	Rapid, Pinpoints Specific Location & Leakrate
RPM	Proximity Sensor	Magnetic Strip	Combined RPM/Torque for Power Info
Torque	N/A	Magnetic Strip	Enabling Technology
Erosion	N/A	Plume Spectroscopy	Enabling Technology

Cycle-Independent Parameters

- Pc Chamber Pressure
- GG or PB Pressure
- Pump Inlie Pressures (Ox & Fuel)
- Pump Discharge Pressures (OX & Fuel)
- Pump Inlet Temperature (Ox & Fuel)
- Turbine Inlet Temperature
- Pump Case Temperature
- Engine Compartment Temperature
- Valve Position Indicators
- Accelerometers for Vibration Level
- Leak Detection (Ox & Fuel)

Cycle-Specific Parameters

- Fuel Venturi Inlet Pressure
- Gas Generator Pressure
- Preburner Pressure

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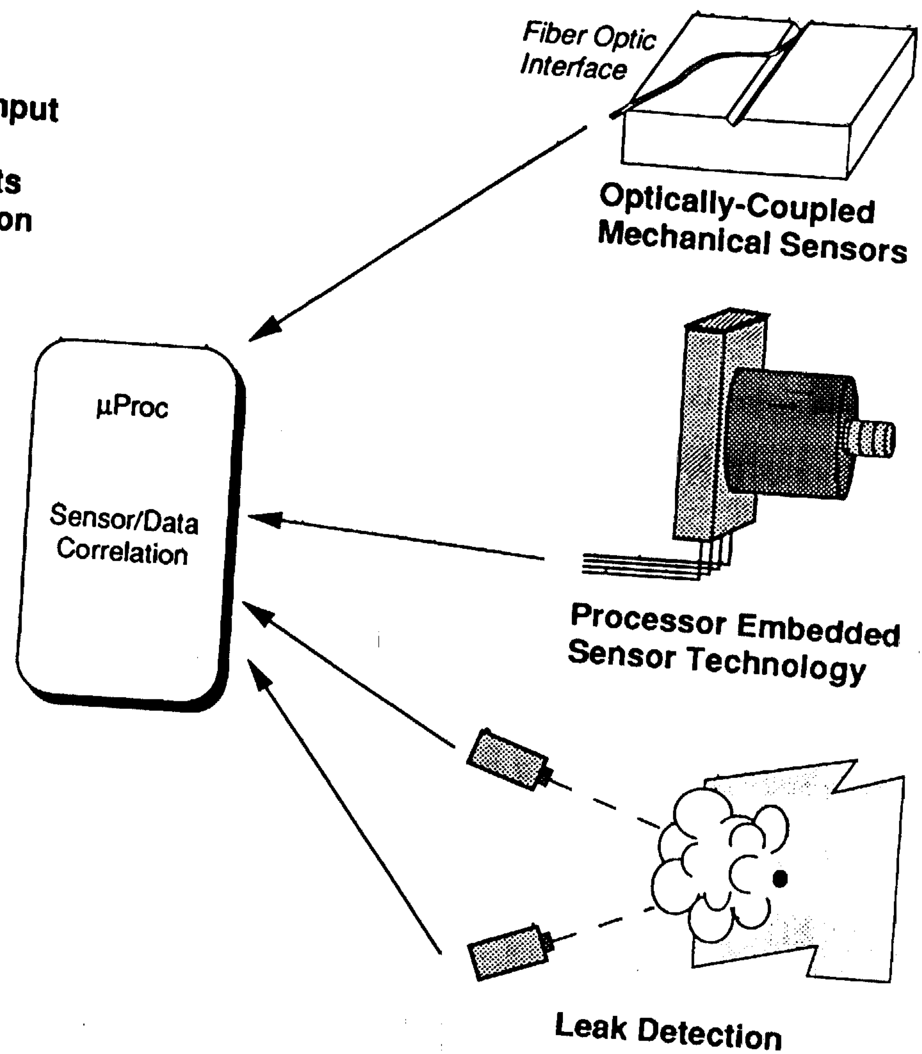
Propulsion Sensor Technologies

Features

- Optically Coupled Sensors for Noise Immunity and Data Throuhput
- State-of-the-Art Silicon Sensors
- Non-Intrusive Flow Measurements
- Infrared Absorption Leak Detection
- Improved Data Correlation for Processing Multiple Sensors



Benefits

- Increased Sensitivity
- Reduced Weight
- Reliable Operation in High Temp., Noise and Vibration Levels
- In-Flight Detection of Potential Problems
- Higher Fault Coverage
- Automated Leak Detection During Preflight Operations
- Reduced Post Flight Analysis and Inspection

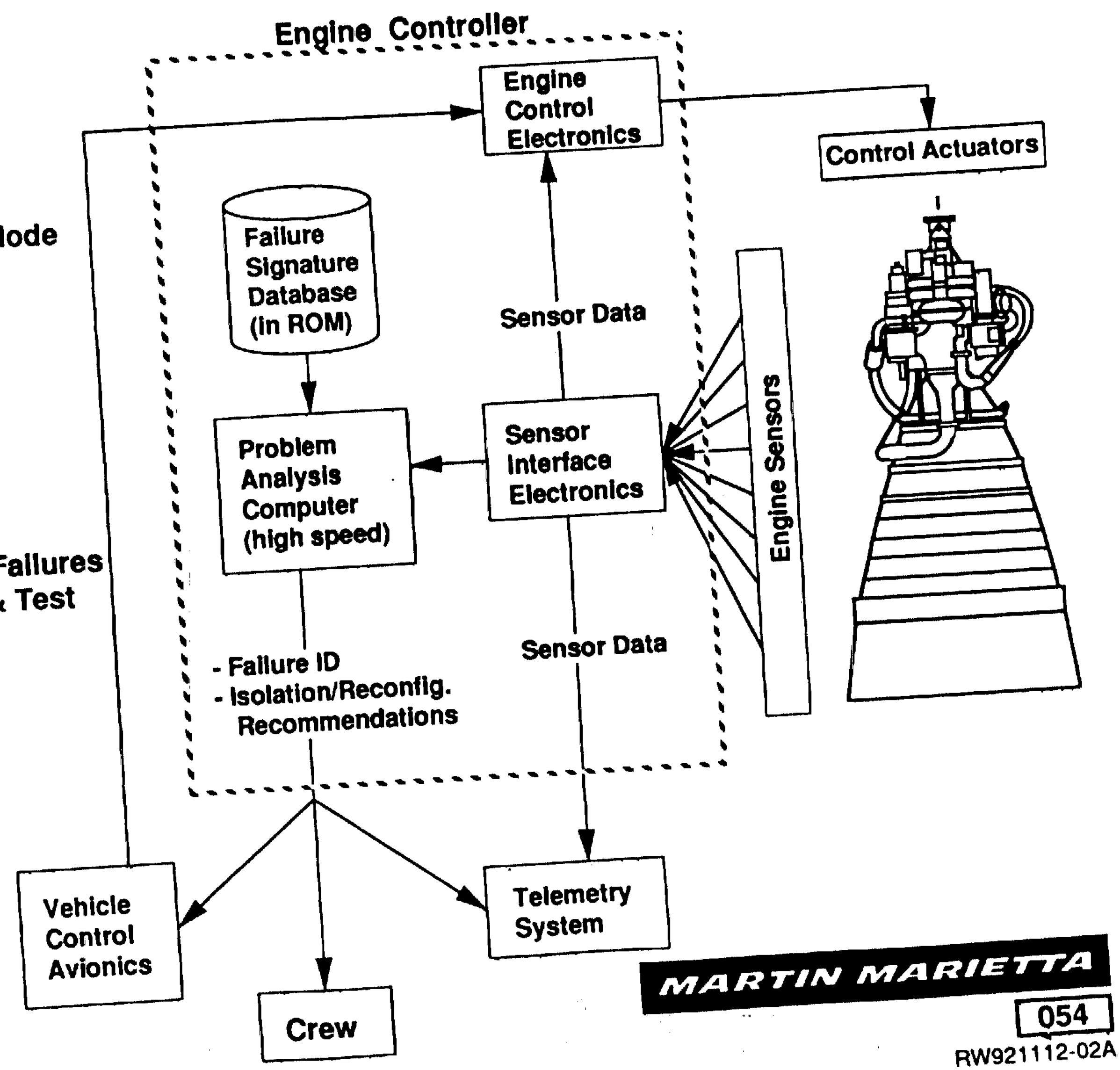
Automatic Failure Identification System

Features

- On-Line Database of Failure Signatures
- Real Time Sensor Fusion
- Rapid Identification of Failure Mode
- Potential Failure Prediction capabilities

Benefits

- Enabling Technology to Meet Automatic Failure Isolation & Reconfiguration Requirement
- High Fault Coverage
- Can respond to Time-Critical Failures
- Enhances Ground Checkout & Test



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Recommended IVHM Technologies

Recommended Technologies	Drewbacks Associated w/Technology	Development Requirements	Level of Effort
• Automatic Failure Identification	Development & Implementation Costs	Knowledge Base Capture & Incorporation Into Computer Data Base	Knowledge Capture: 2-3 Yrs. $1M/Yr Data Base Dev. 2 Yrs. $2M/Yr.
• Leak Detection	Development & Implementation Costs	Ruggedized IR & UV Leak Detection Equipment	3 Yr. Development & Demonstration $10-15 Million
• Optically-Coupled Sensors, Capacitive Pressure Sensors, Sensor Processing	Development & Implementation Costs Some Technologies Most Likely to be Ground-Based Others Will Require Space Qualification	Some Already Developed in Commercial Application. Some Application Specific Develop. Required	Application Specific: 2-3Yr. Development and Test $5-12 Million

System Recommendations

System/Vehicle Health Management

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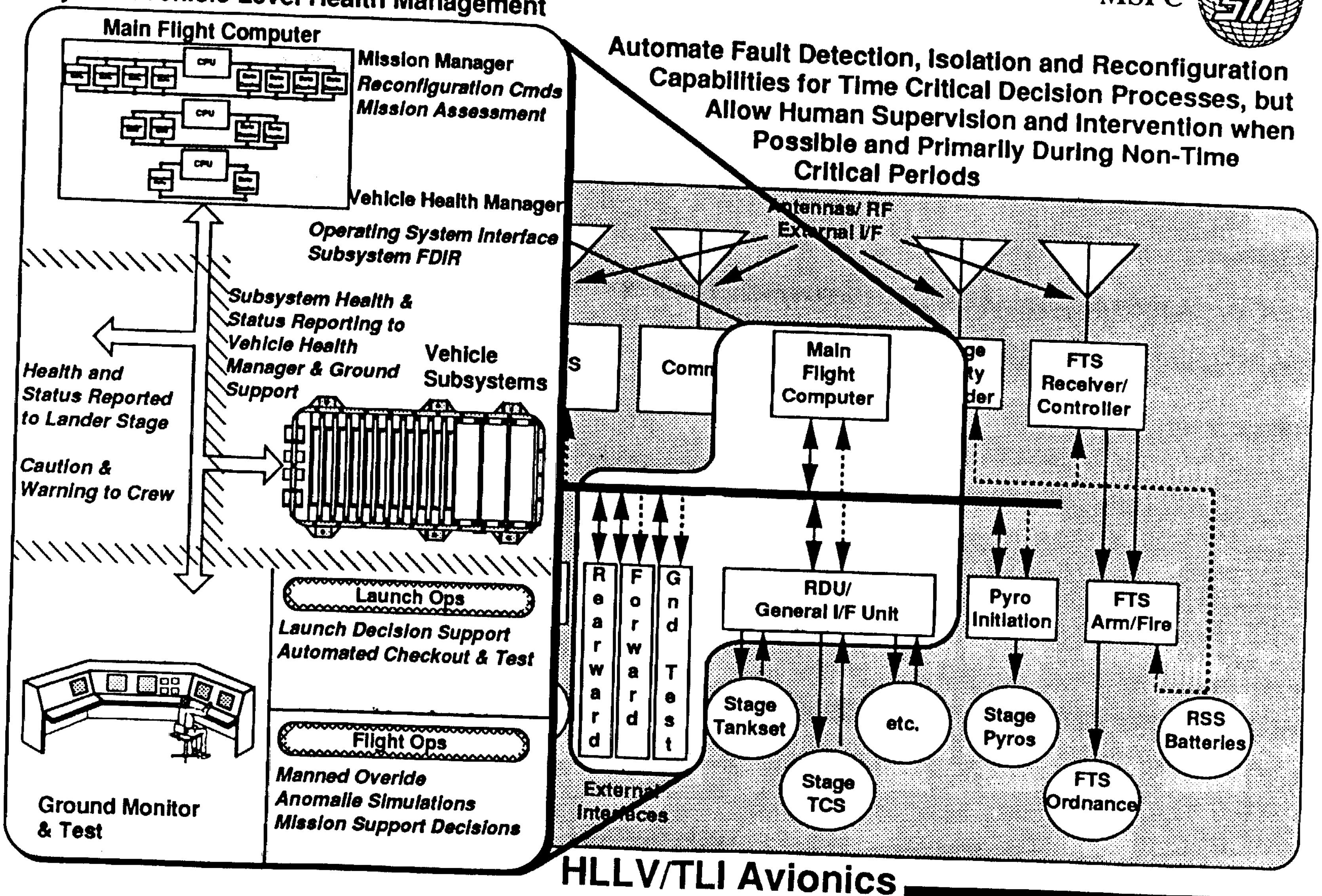
Automate Fault Detection, Isolation and Reconfiguration Capabilities for Time Critical Decision Processes, but Allow Human Supervision and Intervention when Possible and Primarily During Non-Time Critical Periods



HLLV/TLI Avionics

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VHM Allocation Recommendations

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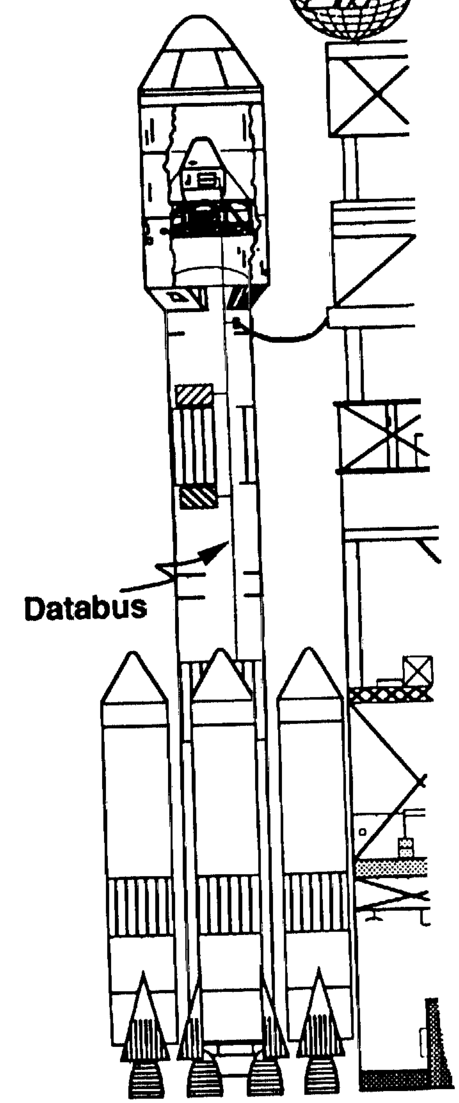
Features

Lander Module Health Management System
- On-Board Automated Fault Detection and Isolation System
- Crew Interface to VHM System
- Caution and Warning to Crew and Ground
- Reconfiguration Recommendations to Lander Mission Manager and Crew
- Tied to HLLV and TLI Stage via Fiberoptic Databus
- Communications with GSE via Fiberoptic Umbilical
- Configuration Status to Crew and Ground

TLI Stage Health Management System
- On-Board Automated Fault Detection and Isolation System
- Caution and Warning to Lander Module and Ground
- Reconfiguration Recommendations to Lander/Cargo Vehicle
- Tied to HLLV and Lander via Fiberoptic Databus
- Communications with GSE via Fiberoptic Umbilical
- Configuration Status to Lander/Ground

HLLV Health Management System
- On-Board Automated Fault Detection and Isolation System
- Caution and Warning to Lander Module and Ground
- Reconfiguration Recommendations to Lander/Cargo Vehicle
- Tied to TLI Stage and Lander via Fiberoptic Databus
- Communications with GSE via Fiberoptic Umbilical
- Configuration Status to TLI/Lander/Cargo Vehicle



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Vehicle Life Cycle Applications for VHM

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Design

•Activities
- Functional Verification Requirements Integration
- Std. Control and Test Interface
- Access and Observability
- Knowledge Capture and Crossover to Analysis

•Benefits
- Early Understanding of Test, Verification Approach
- Reduced Design Workload Due to Use of Std. I/F's
- Better Inter-disciplinary Cooperation
- Design and Anal. Knowlege Portability (CAD/CAE)

•Costs
- Higher Design Complexity
- Extensive Designer Training
- CAD/CAE Equipment Purchase and Maintenance
- Knowlege Carry Forward "Bridge" Development

Development and Production

•Activities
- Functional and Parametric Performance Observations
- Interface Simulation and Modeling
- Piece Part Procurement and Screening (Knowlege Capture, Carry Forward)

•Benefits
- Development to Well Defined and Trusted Criteria
- Faster Debug of Anomalies Due to High Observability
- Easier, More Programmed Access to System Solutions for "To-Hard" Problems
- Environmental Test Performance Observability

•Costs
- System Level Flexability
- Modification of Environ-mental Test Facilities
- Knowlege Carry Forward "Bridge" Development

Integration & Checkout

•Activities
- Interface Fit/Mate Checks
- Cross-Interface Control, Power, Etc., Performance Verification
- Ordnance/Hazardous Item Integration and Checkout
- Pre-Flight Capacities Status (Propellants, Power, Etc.)

•Benefits
- More Thorough Verification Coverage of Integrated Interfaces
- Greatly Reduced Need for Invasive Test Procedures
- Reduced Integration, Test Personnel Requirements
- Fewer Hazardous Activities

•Costs
- Integration of Knowledge Carry Forward into Integration System
- Updating of Integration and Test Equipment to Standard Test Access Compatability

Mission (Flight)

•Activities
- "Fail-Safe" Management (Active Safing of Failure Hazards)
- Redundancy Management; In-Flight Verification of Redundancy, Reconfig. in Event of Failure
- Priority and Load Balancing
- Flight Sensor Data Compression, Reduction

•Benefits
- Rational, Understandable System Managment
- Reduced Design Complexity, Hardware Use
- Higher Reliability Through Effective Redundancy Use
- Reduced Real-Time Flight Monitoring and Mission Management Personnel

•Costs
- VHM-Specific Equipment Weight and Complexity

Post-Mission Analysis

•Activities
- Automated Mission Safety Compliance, Discrepancies, Hazard Effects, Reliability, and Performance Analysis Report Generation
- Mission Anomaly Resolution

•Benefits
- Reduced Post-Mission TLM Processing
- Fewer Anomalous Mission Results which Require Resolution or Waiver
- More Complete Knowledge of Vehicle Behavior During the Mission
- Easier Access Back to Design, Development and Test Database To Improve Performance for Later Missions

•Costs
- Automated Report Generation Software Development

IVHM System Recommendations

MSFC

- Automate Vehicle Health Management and Reconfiguration Capabilities for Time Critical Decision Processes, but Allow Human Supervision and Intervention into the VHM System when Possible and Primarily During Non-Time-Critical Periods
- Assess Individual Parameters at the Subsystem or Component Level, but Resolve and Correct Vehicle and Mission Critical Problems at a Centralized Location (Lander with Crew Involvement or Ground)
- Plan Allowable Recovery Times to Match the Time Criticality of the Function
- Utilize Common Test Procedures throughout the Subsystem/Component Life Cycle
- Simple Fault Tolerant Architecture Is Key to Affordable VHM
  - Complex Hardware and Software Will Increase VHM Costs
  - VHM Must Be at the Same Fault Tolerance Level as the Vehicle It Is Monitoring

IVHM System Recommendations cont'd

- Risk of using Off-the-Shelf Electronic Hardware is High, Much of it is Outdated and Contains Minimal Built-In-Test Capability. Cost Effective Technology Exist Today that can Provide the High Failure Coverage Necessary to Meet the Requirements for FLO and Beyond.

- Highest Near Term VHM Payoffs are in the areas of:

  - Avionics Electronic Equipment to Improve Fault Coverage and Reduce Manufacturing and Test Costs
  - VHM Added to the Propulsion System to Improve Fault Coverage and Increase Safety
  - New Technologies (Laser Ordnance, Automated Cable Test, etc.) that Improve Ground Processing and Reduce Costs Through Faster On-Pad Checkout and Test.

Technology Recommendations

HLLV/TLI Stage IVHM Life Cycle Costs

MSFC

Technologies	2 Missions	10 Missions	50 Missions
Electrical/Electronics	+$12.9 M	+$10.3 M	+$0.0 M
Ground Processing	+$61.7 M	+$49.1 M	-$0.2 M
Power	+$0.2 M	-$6.7 M	-$33.4 M
Software	+$15.5 M	-$10.2 M	-$110.9 M
Propulsion	+$16.5 M	+$1.9 M	-$55.2 M
TOTALS	+$106.5 M	+$44.4 M	-$199.7 M

Overall HLLV/TLI Development Program Estimated to be $5-10 B

IVHM Technology Recommendations

Technologies	2 Missions	10 Missions	50 Missions
Electrical/Electronics	X	X	X
Ground Processing	?	X	X
Power	X	X	X
Software	X	X	X
Propulsion	X	X	X

Technology Recommendations Based on Life Cycle Cost, Safety, Reliability and Performance Considerations for a Selected Number Of Planned Missions

IVHM Demonstration Candidates

Demonstration Suggestions Benefiting HLLV/TLI Stages Listed in Order of Importance

Demonstration	System	Subsystem/ Component	Description
In-Flight Checkout, Verification of Redundancy, Reconfiguration, Redundancy Management; Utilize Martin IRADs and NASA Information and Electrical Systems Lab	X		Demonstration of State of the Art Fault Detection, Isolation, and Recovery Strategies; Techniques for Periodic In-Flight Verification of Redundant Paths; Integration of Built-In-Test with Redundancy Management Techniques and Comparison of Methods
RCS-Valve Current Traces, Thruster Action Signatures, Automated Reconfiguration		X	Demonstrate and Compare Signature Analysis Techniques for Thruster Valves; Demonstrate New Automated Reconfiguration Strategies for RCS Systems
Improved/Automated Propellant Loading (Level Sensing, Leak Detection, Sensor Fusion); Utilize MHTB and Leverage from Previous AUSTS Demo		X	Demonstrate Sensor and Subsystem Sensor Processing, Sensor Placement, Propellant Level Sensing, and Leak Detection Techniques, Hardware and Software Algorithms; Demonstrate Data Correlation from Multiple Sensor Types, and Fault Tolerant Processing of Redundant Sensors
Laser Initiated Ordnance Checkout		X	Demonstrate Effectiveness and Time Savings Associated with Laser Ordnance Processing; Demonstrate Inherent End-To-End Checkout and Test Characteristics of Laser Ordnance System.
Fiber Optic Data Bus		X	Demonstrate Inherent Characteristics of Fiber Optic Data Buses for Automating Data Bus Checkout and Test; Demonstrate Fault Tolerant Apsects of Fiber Optic Links and Interconnections

IVHM Demonstration Candidates

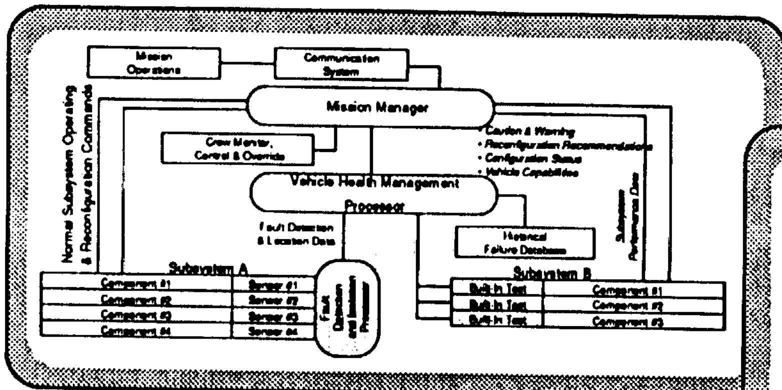
MSFC

Demonstration	System	Subsystem/ Component	Description
Electromechanical Actuators for TVC, Main Valve Actuation		X	Demonstrate Inherent Characteristics of Electro-Mechanical Actuators for Automating their Checkout and Test. Demonstrate New Technologies for Fault Detection and Management of Redundant Actuators
Semi-Automated Recording , Retention, and Reporting of LRU Maintenance Records (failure history, hours of use, configuration changes, etc.)		X	Demonstrate the Effectiveness and Operation of an Automated On-Board Maintenance Record Keeping System used for the Diagnostic Evaluation of Reusable Hardware
Engine Altitude Start/Restart; Engine Data Correlation and Evaluation (automated)		X	Demonstrate the Hardware/Software Improvements and/ or Additions Required to Implement Altitude Start and Restart Operations; Demonstrate New Sensors, Sensor Placement, Sensor Processing and Data Fusion Techniques

IVHM Demonstration Recommendations

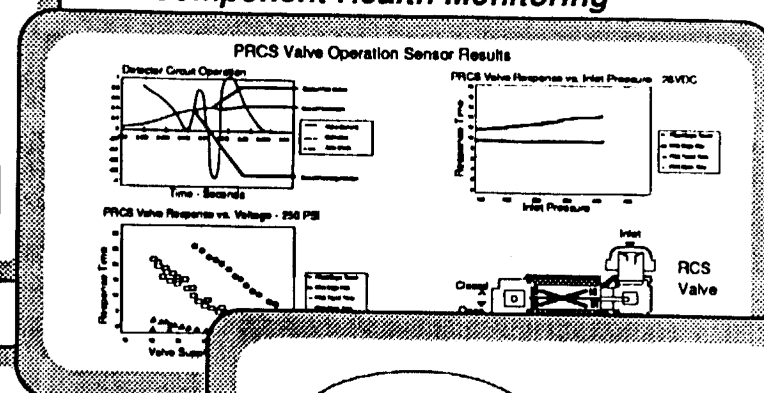
MSFC

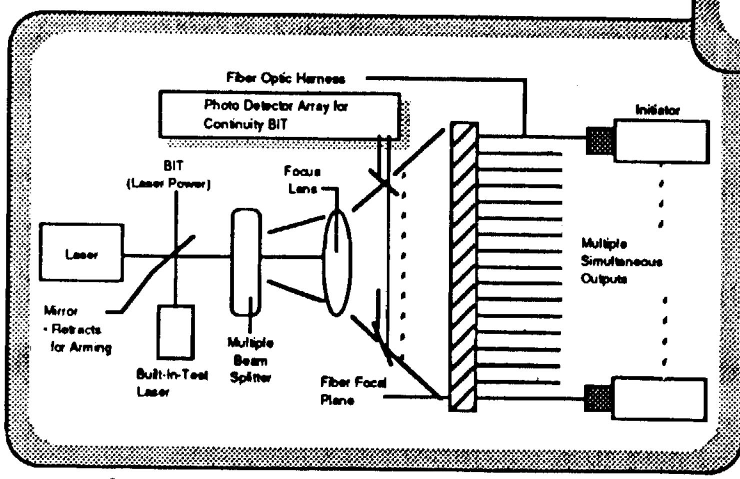
Recommended IVHM Technology Demonstrations Offer the Most Benefit to the HLLV/TLI Stages and SEI Missions



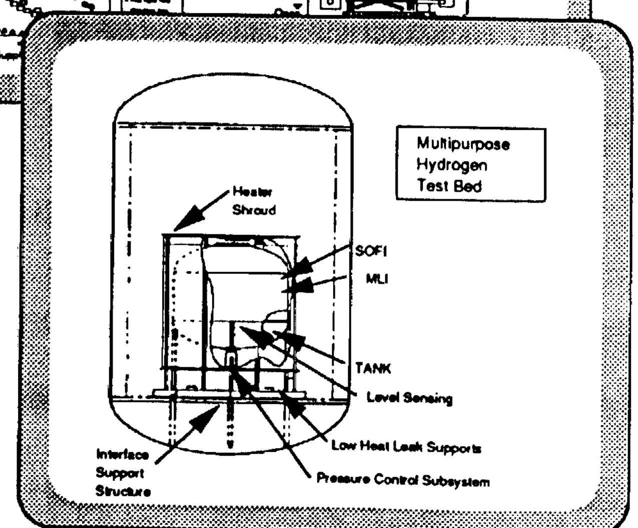
IVHM Architecture

Component Health Monitoring





Laser Initiated Ordnance Checkout



Improved/Automated Propellant Mgmt.

MARTIN MARIETTA

067

RW920928-02A

IVHM Application Issues

Issues

- How Will Technology Development Costs be Divided Between Programs ?
- How Will Technology Development be Centralized and/or Coordinated ?
- How Do We Ensure that Appropriate Technologies are Evaluated ("Black" Programs, Foreign, etc.) ?
- How Do We Address the Intangible Benefits Associated with IVHM ?

Recommendations

- Additional Direction Needed in the Area of IVHM Requirements, Definition of Fault Tolerance Levels, and Requirements for the HLLV-TLI-Lander Stack
  - Will Provide More Information for Determining Where and How Much IVHM is Needed on the Stack
- IVHM Requirements and Technology Needs for Mars Evolution
  - What Needs to be Done Differently
- Near Term Laboratory Demonstrations are Required to validate IVHM Concepts and Verify IVHM Designs
  - Provide Confidence in Technology

Technical Directive 15

Fluid Acquisition and Resupply Experiment (FARE) Data Analysis and Consultation

Fluid Acquisition and Resupply Experiment, Flight I
Technical Directive 15, Final Report

Contract NAS8-37856
July 1993

prepared by
James Tegart

Martin Marietta Astronautics
Denver, Colorado

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Foreword

This report presents the results of an analysis of the data from the first flight of the Fluid Acquisition and Resupply Experiment (FARE). The effort was performed as Technical Directive 15 for contract NAS8-37856, under the direction of the principal investigator, Susan Driscoll of the NASA Marshall Space Flight Center, Alabama. The FARE project was managed by Sam Dominick of Martin Marietta. It is only fair to acknowledge the contributions of the STS-53 crew who performed the tests, in particular Rich Clifford, Jim Voss and Guy Buford.

I. Introduction

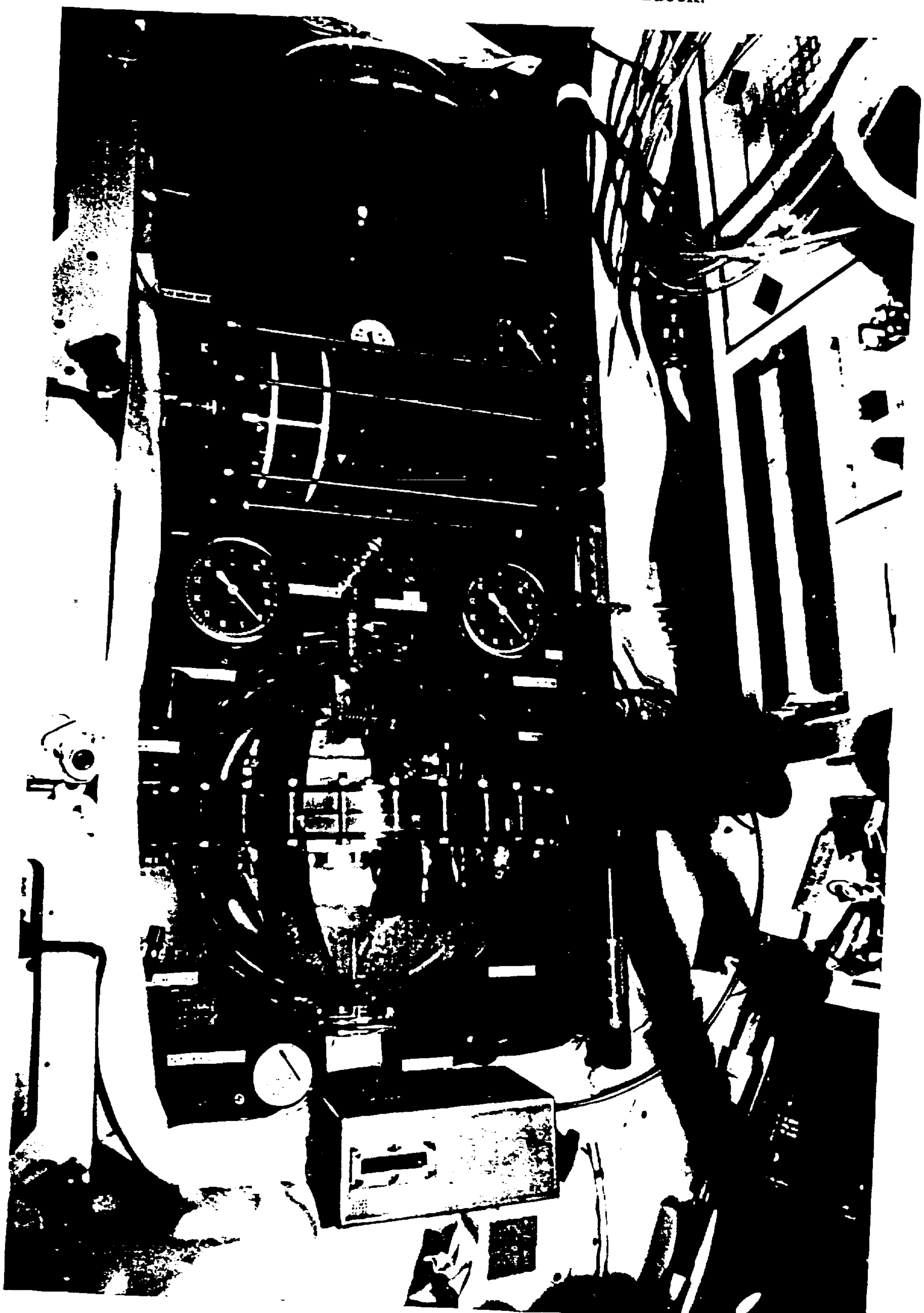
The Fluid Acquisition and Resupply Experiment (FARE) is a shuttle middeck payload that was launched on STS-53 on December 2, 1992. Over the next six days, eight tests were performed, investigating the zero-g transfer and expulsion of liquids from a subscale model tank. The test objectives were as follows:

- Demonstrate the low gravity operation of a total communication screen channel type acquisition device during tank expulsion and refill.
- Demonstrate the low gravity venting of a tank while filling, making use of the capillary liquid orientation and controlling the inflow momentum.
- Demonstrate the static behavior of the liquid under ambient low gravity conditions and its dynamic behavior with specifically applied accelerations.

The experiment consisted of two modules that mounted in place of four lockers in the middeck of the orbiter (Figure 1). The lower module had the supply tank, which used an elastomeric diaphragm for expulsion. The tests began with the supply tank completely filled and some additional liquid stored in the calibrated cylinder mounted on the front of the module. The upper module had the receiver tank, which was the primary interest in the testing. The receiver tank had a four channel, total communication, surface tension type expulsion device, and a fill nozzle and baffle as a means of filling the tank. The tanks were interconnected for transfer, pressurization, and venting. A pressurization system provided regulated air at 10 psig from a 2000 psig source. A port allowed the modules to be connected to the orbiter waste management system, serving as an overboard vent. Functioning of the experiment was achieved with valves operated by the astronauts per a procedure. Figure 2 is a plumbing schematic. Each test consisted of filling the receiver tank from the supply tank and then reversing the flow to expel the receiver tank.

Also included in the modules was an orbiter powered back-lighting system for the receiver tank, a flow meter to accurately determine the flow from the supply tank to the receiver tank, and a NASA provided acceleration measuring and recording system. Data was collected primarily by two video cameras, one aimed at the upper half of the receiver tank and one aimed at the lower half. Crew comments were recorded along with the video and were annotated on the test procedure. 35mm still photos supplemented the video data. Further description of the experiment can be found in reference 1.

Figure 1. FARE installed in orbiter middeck.



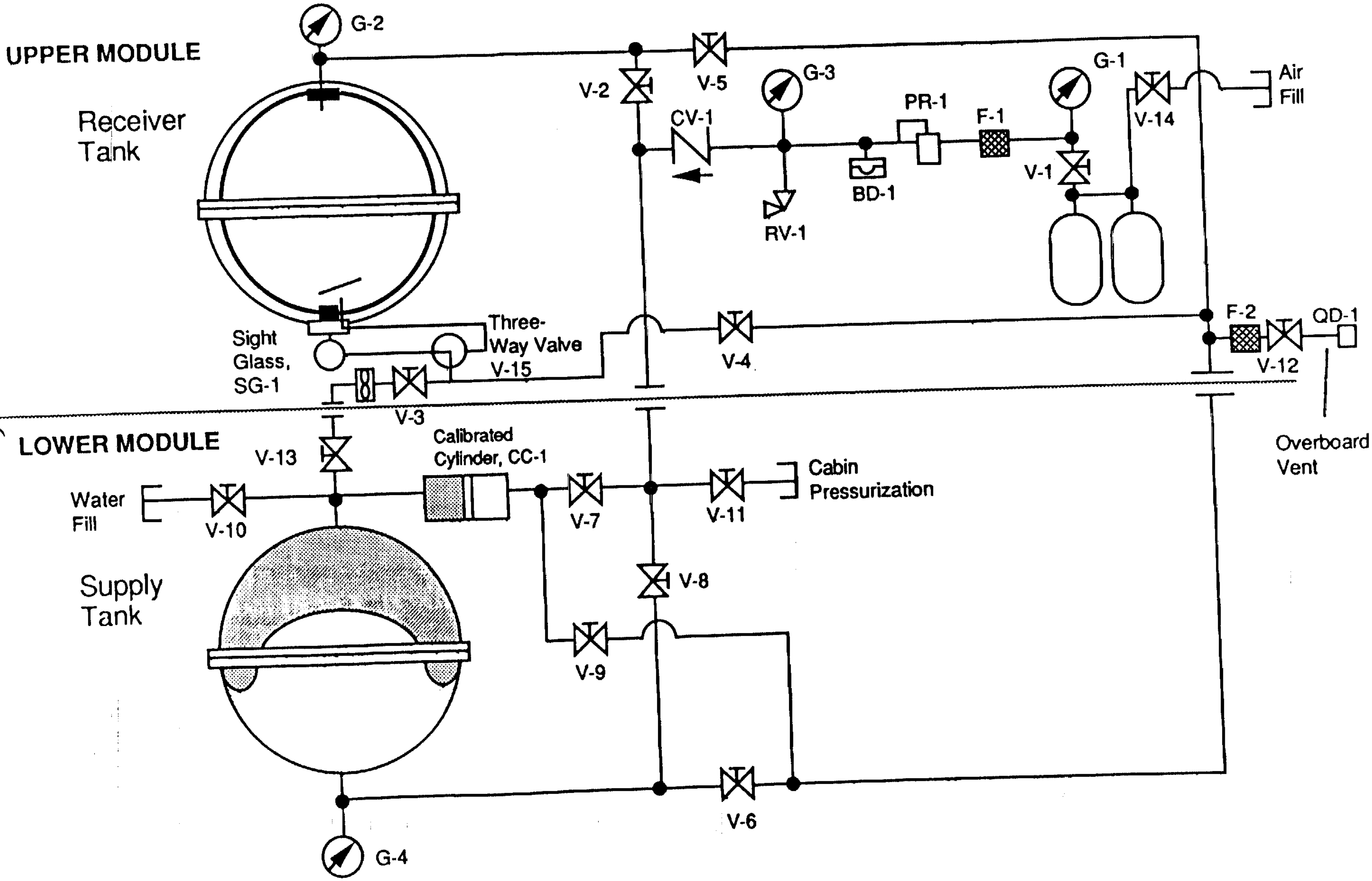


Figure 2. Plumbing schematic

II. Test Description

The following description of the FARE I tests was derived from the video data, including comments on the sound track, still photos and annotations on the test procedure. Table 1 is the test matrix.

<u>Test 1</u> -Evacuated fill at 1.2 gpm

There was a small quantity of liquid initially in the receiver tank, that could not be emptied following the functional tests performed prior to installation in the orbiter. With the orbital vertical for launch, this residual was oriented to what is called the front of the tank. At the beginning of test 1 it could be seen in the gap between the channel and tank wall, only in the vicinity of the tank girth. The initial reading on the air bottle pressure gage was 1950 psi which closely matched the initial load of 2000 psi, measured with a precision gage.

After evacuating the receiver tank for 30 to 40 minutes the pressure could never be reduced to less than -28 in. Hg. The test conditions were 14.8 psia (30.1 in. Hg) cabin pressure and a liquid temperature of 76.5 °F. At this temperature the saturation pressure of water is 0.46 psi or 0.93 in. Hg. Since there was water in the receiver tank, the lowest pressure to which the tank could be evacuated, unless it was completely dryed, would be -29.2 in. Hg. With the ground support equipment, including a vacuum pump, it typically took 15 to 20 minutes to evacuate the receiver tank to around -29 in. Hg. The difference is attributed to either a greater flow resistance in the orbiter overboard system or a small leak downstream of the FARE modules. This same behavior was noted for the first flight of this hardware in 1985. A specific pressure did not need to be reached during evacuation, but the lower the pressure in the receiver tank the greater the volume of water vapor that would be generated. The key is the purging of non-condensible gases (air) from the receiver before filling begins. The gas bubbles remaining after fill show how much air remained in the tank after evacuation. After noting the excessive time required for evacuating the tank, the crew requested guidance. They were instructed to vent until no further pressure reduction was observed, and then to continue the test.

During fill, liquid entered the receiver tank through the channels of the screen device. Due to the low pressure some vaporization of the liquid occurred as it entered from the supply tank (at 10 psia) and some of the air dissolved in the water evolved. This vapor and air created a foamy fluid that could be observed leaving the channels and covering the walls of the receiver tank. The effect of the wetting agent in the water was to make the bubbles

Table 1. Test Matrix

Table 1. Test matrix

Test	Procedure	Fill/Drain Flowrate	(G-2)		(G-3)	V-13 turns,ccw		GPM		Est. Flow Duration, min; Inflow/ Outflow
						Fill	Expulsion	Inflow	Outflow	
1	Evacuated Fill/Expul	High/High	-29	10	10	5	5	1.2	1.2	3.2/3.2
2	Evacuated Fill/ Expul with Pulsed Flow	High/High	-29	10	10	5	5	1.2	1.2	3.2/3.2
3	Evacuated Fill/Expulsion	High/High	-29	10	10	5	5	1.2	1.2	3.2/3.2
4	Evacuated Fill/Expulsion with Accelerations	Med/High	-29	10	10	3	5	.7	1.2	5.4/3.2
5	Evacuated Fill/Expulsion to 5%	High/Med	-29	10	10	5	3	1.2	.7	3.2/5.4
6	Vented Fill/Expulsion to 5%	Low/Med	0	10	10	1/2	3	.1	.7	29/5.4
7	Vented Fill/ Expulsion to 5%	Med/Med	0	10	10	1 1/2	3	.2	.7	14.5/5.4
8	Vented fill/ Expul to gas ingestion	High/Med	0	10	10	3	3	.3	.7	9.5/5.4

produced by these gases persist, inhibiting their coalescence. The filling continued from the wall inward until it appeared that the tank was filled with the bubbly mixture. When the tank was nearly full the flow rate began to slow and the tank pressure rose. At this time the vapor bubbles condensed and some of the air redissolved, so the liquid dramatically cleared. It took 20 seconds for complete collapse of the small bubbles after reaching full pressure. One larger bubble (maybe 2 inches in diameter) and about 9 smaller bubbles could be seen in the receiver tank at the end of fill. The crew estimated the fill at 98%.

Test 1 - Expulsion to gas ingestion at 1.2 gpm

The supply tank was evacuated to -28 in. Hg for the expulsion of the receiver tank. It appeared that the time required to evacuate the supply tank was about the same as the receiver tank, but a detailed comparison was not possible without a plot of tank pressure versus time. The crew observed a soapy mixture in the sight glass when expulsion was initiated, but allowed outflow to continue. Some air must have been entrapped within the channels when the receiver tank was filled and was not evacuated.

When expulsion began, the pressurant entering the liquid surrounding the pressurization port at the top of the receiver tank, caused bubbles to form. Again, due to the wetting agent, the bubbles persisted, so the tank filled with bubbles from the top downward. Coalescence of the bubbles continued throughout the expulsion, but not at a fast enough rate to clear the liquid. It was not until gas ingestion had been detected in the sight glass and flow stopped, that the coalescence was complete. The continued coalescence gave the appearance that flow was continuing after it had been stopped. Upon nearing depletion, most of the liquid was in the gap between the channels and the tank wall, and then the gap began to empty. Gas ingestion occurs when the the flow area of the liquid in the gap is so small that the pressure losses of flow into the channels exceeds the retention capability of the screen. When gas ingestion occurred the channel gaps, visible from the front of the tank, looked completely empty. Wicking of the residual liquid and the completion of the bubble coalescence resulted in some refill of the channel gap after flow had been stopped. From the video, some residual liquid could be observed in the channel gap at the tank girth, and at the top and bottom of the tank. Since the channel gap for the two channels located at the back side of the tank could not be seen, their condition remained unknown throughout all the tests. The crew recorded an estimated residual of 1% in the receiver tank. The supply tank was recorded as being 98% full, with a few bubbles. Those bubbles were entrapped in the channels of the screen device during receiver tank fill, as a consequence of not completely evacuating the receiver tank before fill.

Test 2 -Evacuated fill at 1.2 gpm

The conditions for this evacuated fill were the same as test 1. This time the receiver tank was evacuated to -27.5 in. Hg and the fill appeared similar to test 1. However the liquid did not clear of small bubbles when pressurized as before. Somehow the liquid must have been filled with small air bubbles during flow, that were not absorbed into the liquid. During the preparations for the expulsion some coalescence of the small bubbles into the larger bubbles was apparent. An orbital maneuvering system burn was also performed during this period, which oriented all the gas bubbles toward the back of the tank. After this event the liquid was clear and the gas had coalesced into a single bubble. The crew estimated the fill as 99%.

Test 2 - Expulsion to gas ingestion with pulsed flow at 1.2 gpm

The initial expulsion was the same as test 1. As the expulsion proceeded, small bubbles could be observed rising from the tank girth within the channel to tank gap. These bubbles must have entered the gap at the girth, out of sight behind the flange, and then capillary pumping due to an increasing gap width caused their rise. Flow was to be stopped when 5% was remaining, but the crew estimated it may have been 10%. Outflow was resumed by opening and closing the toggle valve with one second intervals until gas ingestion was detected. It required 18 pulses to reach ingestion. When the pulsed flow began, the gas bubbles were still coalescing and did not cease until after gas ingestion. When gas ingestion occurred the channel-to-tank wall gap was almost empty but it filled with the residual liquid after flow was stopped. A few bubbles were left entrapped in the channel gap. Compared to test 1, the pulsed flow increased the residual. The crew estimated a 3% residual. The supply tank was recorded as being 97 to 98% full and a still photo of the supply tank showed a bubble corresponding to that fill. Gas that became entrapped in the channels could be transferred back and forth between the supply and receiver tanks.

Test 3 - Evacuated fill at 1.2 gpm

For this fill the receiver tank was evacuated to -27.2 in. Hg and the fill looked the same as tests 1 and 2. At the end of the fill the crew recorded that there were 3 large bubbles, about 2 inches in diameter, and many smaller bubbles from 0.1 to 0.5 inches in diameter. Most of those bubbles were visible in the video. The fill was estimated as 96% by the crew. The higher evacuation pressure and shorter evacuation time contributed to this larger gas volume.

Test 3 - Expulsion to gas ingestion at 1.2 gpm

While the supply tank was being evacuated the bubble motion produced by the operation of an exercise machine could be observed. The machine had an audible inertia reel and the acceleration produced was of a sinusoidal nature. With each extension of the machine the bubbles would shift about one quarter of an inch and on retraction they would return to their original position. Over a period of time there was a gradual drift of the bubbles.

Just before gas ingestion the gap between the channels and the tank wall emptied, but it refilled after the flow was stopped and the bubble coalescence ceased. The crew estimated the residual as 5% and recorded that only three bubbles in the channel gap could be seen. Some liquid could be seen around the baffle supports. The supply tank was recorded as being 100% full, with no bubbles. The bubbles present in the previous tests could be observed being purged through the screen as the receiver tank began to fill.

Test 4 - Evacuated fill at 0.7 gpm

Test 4 was not run in sequence. Test 5 followed test 3 and test 4 was run after test 8. Therefore the final conditions of test 8 became the initial conditions for test 4 and the same for tests 3 and 5. This change in order should not have had any effect on the test results, but it needs to be remembered in evaluating the changes in liquid volume recorded by the calibrated cylinder.

The receiver tank was evacuated to -27.2 in. Hg. A fill to 99% was estimated by the crew. This time the liquid did clear when the tank reached full pressure. One larger and three smaller bubbles could be seen. Otherwise this fill appeared the same as prior fills, even though it was performed at a lower flow rate.

Test 4 - Expulsion to gas ingestion with accelerations at 1.2 gpm

This expulsion was performed in three stages: expel from initial fill to 50% fill where a series of accelerations were applied; to 5% where some more accelerations were applied; and a final expulsion to gas ingestion in conjunction with an adverse axial acceleration. It was difficult to estimate when the 50% level was reached during the first expulsion due to the bubbly liquid. Estimates based on the flow rate and time, and visual estimates possible after coalescence was complete gave a 50% fill.

At 50% fill there was a series of 27 distinct accelerations applied to the tank. Three of these were the planned -Y (liquid moved right as view tank from front), +Y (liquid moved left) and +Z (liquid moved to top of tank) accelerations, while the others were produced when the orbiter was returned to its proper attitude following each of the above accelerations. One minute of free drift was allowed for previous disturbances to damp, the planned acceleration was applied and then there was another minute of free drift to observe the liquid response. The period between accelerations was long enough in 17 cases to allow most of the liquid motion to damp, but in the other cases the accelerations follow one another in a rapid sequence. All of the accelerations were produced with the larger primary thrusters, so significant liquid motion resulted. Even so, there was no breakup of the liquid due to the accelerations. In some cases the folding over the the surface produced some larger bubbles, but no spray or gas entrainment was observed. The liquid motion at 50% fill was characterized by bulk motion of the liquid, including swirl and a single, geyser like, instability rising from the surface. In some cases the instability passed across the center of the ullage bubble to impact the opposite side. The instability is known as a Rayleigh-Taylor instability (Ref. 2 and a more recent Ref. 3), with the number of such instabilities forming on a surface being a function of the relative magnitude of acceleration and surface tension forces. The liquid motion damped in a short time, requiring from 15 to 20 seconds for the bulk motion to cease and some additional time for the ullage bubble to reach a final static orientation. This final adjustment was most noticeable at the top of the tank. After the bulk motion ceased the quantity of liquid at the top of the tank significantly decreased as the final ullage bubble orientation was achieved. At the 50% fill volume the baffle still influenced the bubble position, causing most of the liquid to orient at the tank bottom.

When the first acceleration was applied at 50% fill there had been some coalescence of the smaller bubbles, but it was not complete. The first few accelerations increased the coalescence rate and after the first ten acceleration events most of the small bubbles were gone.

Expulsion was resumed to reduce the fill to 5%. Some small bubbles formed at the pressurization tube but they quickly coalesced with the ullage bubble. The channel gap was beginning to empty when flow was stopped. After stopping the flow the channel gap refilled. Some liquid could be seen collected around the baffle.

There were two planned acceleration events (-Y and +Y) with one minute of free drift before and after. Including the corrections to the orbiter attitude there was a total of 17 acceleration events, all performed with the primary thrusters. The longer duration accelerations made the bulk liquid move to one side of the tank. In most cases the channel gap on the opposite side of the tank partly emptied, while the gap near the bulk liquid remained full. In less than 10 seconds all the liquid motion had damped and the channel gap was completely refilled. For the shorter duration attitude correction accelerations the channel gap remained full.

Expulsion to gas ingestion was resumed 3 seconds after the final +Z acceleration was applied. The liquid had moved to the top of the tank and the visible channel gap emptied before gas ingestion. The upper channels refilled about one-sixth of the way after the flow was stopped. This expulsion was a worst case condition for the screen device because the bulk liquid was oriented at the top of the tank, the farthest from the outlet, and an adverse acceleration was acting during outflow. The crew noted that 80% of the liquid in the channel gap had been expelled and a residual of 3% was estimated. The supply tank was 100% full. After outflow was stopped three more acceleration events occurred as the orbiter attitude was corrected, causing some shifting of the residual liquid.

Test 5 - Evacuated fill at 1.2 gpm

The receiver tank was evacuated to -27.5 in. Hg. The filling was stopped before the receiver tank had reached full pressure so the liquid was filled with numerous very small bubbles. One bubble, one to two inches in diameter, was noted. The crew estimated the fill as 98%. The exercise machine was again in operation during the fill and the crew noted oscillations of the bubbles, but they are difficult to see in the video.

Test 5 - Expulsion to 5% at 0.7 gpm

When the receiver tank was pressurized for expulsion the liquid cleared, but a number of bubbles were introduced at the top of the tank by the pressurant. After the liquid cleared two small bubbles could also be seen in the lower half of the tank and the crew said that there was a large bubble at the back of the tank.

When expulsion started some small bubbles were noted in the sight glass and later in the expulsion a single bubble was observed to be dancing in the flow through the sight glass. The flow was stopped when bubbles first began to enter the channel gap. Large bubbles remained in the gap of the channels in the lower half of the tank. The supply tank was again completely full.

Test 6 - Vented fill at 0.1 gpm

For this filling test the receiver tank pressure was reduced to 0 psi gage so additional venting was necessary to maintain a constant flow rate during fill. For this particular test the technique used by the crew to control the flow rate was to monitor the flowmeter, waiting until the flow rate was less than 0.09 gpm, and then opening the receiver tank vent enough to return to 0.1 gpm. This process was repeated throughout the test until liquid free venting was no longer possible. Then, with no further venting, fill was allowed to continue until flow stopped.

At this flow rate the momentum force of the inflowing liquid was less than the surface tension force, so the incoming liquid collected about the fill port in a stable manner. This behavior was expected based on the various regimes for the inflow into tanks in zero-g that have been experimentally and analytically studied, as summarized in reference 4. There was a continued, periodic oscillation of the liquid surface when filling began and it continued until the region below the baffle had been filled. At that point the liquid had the form of a bulge that covered the baffle above the fill port. As the tank continued to fill, liquid spread over the tank wall. The baffle continued to influence the liquid orientation, maintaining symmetry with respect to an axis through the fill port and the baffle, which was offset with respect to the screen channels. A single ullage bubble was maintained, with no bubbles being generated by the filling liquid. As the filling continued the ullage bubble approached a spherical shape, that fit within the space between the baffle and the opposite tank wall, and aligned with the baffle axis.

Each time the receiver tank vent was opened it appeared that a small quantity of liquid was being vented overboard. These slugs of liquid could be observed were the vent passage penetrated the transparent cap on the tank and along the transparent line from the modules to the orbiter vent system. This apparent flow was most likely bubbling of the liquid slugs produced as the flow and pressure in the line was varied. Downstream of the vent valve the line remained open to vacuum. The quantity of liquid oriented at the top of the tank gradually increased until the first hole on the vent tube was covered. When venting was then attempted, liquid was continuously vented so filling continued with the vent closed. The first vent hole was 1.5 inches from the tank wall and 1 inch from the manifold of the screen device to which it was mounted. When the vent tube could be seen penetrating the ullage bubble there was only a slight distortion of the surface in the immediate vicinity, so the tube apparently did not influence the positioning of the ullage bubble. It was a one-fourth inch outside diameter tube. The vent tube was completely submerged when filling stopped. The crew estimated that the tank was 60% full. The bubbles that were initially entrapped in the channel gap during expulsion remained fixed during the fill.

Since the ullage bubble was being viewed through the liquid there was considerable optical distortion, making the edges of the bubble difficult to discern. Even the crew, who could view the tank directly from various angles, had trouble determining the location of the ullage. Spherical volumes can be deceptive when estimating their volumes. Consider that a 50% ullage bubble in this 12.5 inch diameter tank is 10 inches in diameter.

After the fill some vernier thruster maneuvers displaced the ullage bubble to the side of the tank. When the maneuver was complete the bubble returned to its original position, aligned with the baffle axis. This shows that the baffle was still influencing the bubble orientation.

Test 6 - Expulsion to 5% at 0.7 gpm

Since there was a large ullage bubble, there were far fewer bubbles generated by the pressurization during the expulsion and they coalesced quickly. This test provided one opportunity to observe the production of drops during the bubble coalescence. This phenomena has been studied on a much smaller scale in one-g (Ref. 5). When the film between a smaller bubble that is tangent to the ullage thins and bursts the bubble surface is not in equilibrium with the adjacent surface. As the bubble surface flattens a jet is formed that pinches off into a drop and leaves the surface. These drops, on the order of 0.1 inches in diameter, could be seen traveling across the ullage bubble. The drops could be seen in some of the other tests as coalescence occurred. The crew reported that they could see some of the smaller drops bounce off the liquid surface after transversing the ullage bubble. The wetting agent that resides on the liquid surface and oblique angles of impact could account for this phenomena.

Since the bubbles had coalesced, the bottom of the tank could be observed much better than prior expulsions as the tank emptied. Liquid collected around the baffle could be seen to drain away to keep the channel gap full. When flow stopped three large bubbles could be seen in the channel gap. The supply tank was again 100% full.

Test 7 - Vented fill at 0.2 gpm

When fill started there was some liquid oriented around the supports below the baffle. A liquid jet from the fill port could be seen to combine with the liquid already around the baffle to fill that region. The jet did not penetrate the baffle. The undulations of the liquid surface started as fill began and continued throughout the test, diminishing only when the flow began to slow. The behavior of the liquid was similar to the prior fill, even though the flow was twice as fast. Again the receiver tank was intermittently vented to maintain the flow until it was observed that liquid was being vented after the first vent hole became covered. During this fill the liquid was full of small bubbles, so it became difficult to see the ullage bubble. The liquid cleared as the pressure rose when the flow stopped, but it was still difficult to see the bubble edges. The crew judged the final fill to be 70%, but agreed that it was all subjective and that it was difficult to estimate the size of the bubble. They said that the bubble was about 2 inches away from the baffle, which helped support their estimate of the volume. In general, they felt that this fill was an improvement over test 7.

Test 7 - Expulsion to 5% at 0.7 gpm

The liquid cleared further and bubbles were introduced when the tank was pressurized for expulsion. There were some maneuvers performed before expulsion began which produced some motion of the ullage bubble. As the expulsion proceeded, most of the pressurization was directly into the ullage bubble, so only a few smaller bubbles were produced. Some oscillation of the ullage surface was apparently due to the coalescence of those small bubbles. The crew noted that the back of the tank cleared of the smaller bubbles before the front, suggesting that this may be due to the heat produced by the lighting. In a thermal gradient, the resulting gradient in the surface tension causes liquid motion along the interface in the direction of the cooler temperature, which fits the observations. When the flow was stopped at 5%, the channel gap was left full, but most of the liquid had been drained from the region of the baffle. The supply tank remained 100% full.

Test 8 - Vented fill at 0.3 gpm

When the inflow began, a jet could be distinctly seen, impacting and covering the bottom of the baffle. The region below the baffle filled and the filling proceeded in approximately the same manner as the previous two vented fill tests. Due to the much higher flow rate the undulations in the surface were larger. The oscillations originated from the baffle region and propagated over the entire ullage bubble surface. During this test the receiver tank vent valve was adjusted so as to match the inflow, so the tank was almost continuously vented to hold the flow rate constant. The crew thought that the ullage bubble was not in alignment with the baffle axis, as it was for the prior tests, but shifted more toward the screen device axis. Again the depth of the liquid at the top of the tank gradually increased until the holes in the vent tube began to cover and only liquid could be vented. The fill volume estimate was 60%, there were no other bubbles and the channel gap was full.

Test 8 - Expulsion to gas ingestion at 0.7 gpm

Many bubbles were added to the receiver tank when it was pressurized for expulsion. During expulsion, small bubbles were again seen moving along the gap between the channels and the tank wall, from the bottom of the tank to the girth in this case. When gas ingestion occurred most of the channel gap had emptied. After flow stopped, liquid could be seen draining from the baffle region to partly refill the gap in the lower dome of the tank. No liquid could be seen around the baffle. From the appearance of the tank this was the most efficient expulsion, with liquid collected in just a few places in the channel gap. The crew estimated the residual as 1% and the supply tank was 100% full.

III. Surface Tension Device Expulsion

The surface tension propellant management device in the receiver tank provided a flow path from the liquid within the tank, regardless of its orientation, to the tank outlet. The fine mesh screen on the side of the channels facing the tank wall allowed liquid to enter the channels while excluding gas. When the pressure differential due to flow and accelerations exceeded the capillary pressure retention capability of the pores of the screen, gas entered the channels. The device was designed so as to postpone gas ingestion until the quantity of liquid remaining in the tank was very small.

The performance analysis of the screen device established that gas-free expulsion would continue until only 3 square inches of the screen was in contact with the liquid outside the channels. Beyond that point the pressure drop due to flow through the screen, when added to the other flow and acceleration pressure differentials, exceeded the capillary pressure retention capability of the screen. It was assumed that as soon as gas entered the channels it would immediately be seen in the sight glass, so none of the liquid inside the channels could be expelled; a conservative assumption. The internal volume of the channels was 2% of the tank volume. If the channel to wall gap was completely full, that volume was 1% of the tank volume. Therefore the best expulsion efficiency predicted was 98% of the tank volume and the worst that could be expected would be 97%, for the case where none of the liquid in the gap was expelled.

Five tests were performed in which the expulsion of the receiver tank was continued until gas ingestion was observed with the sight glass at the outlet of the tank. The other three expulsions were stopped when the liquid remaining was around 5% of the tank volume, which was not a challenge to the capabilities of this device. Under the ambient low gravity conditions of the orbiter, it was demonstrated that the liquid collects around the channels of the surface tension device, keeping the channel-to-wall gap full. As long as that gap was full the screen was submerged in liquid and gas ingestion was not possible. It was only as the last few percent of the tank volume was being expelled and the screens began to be exposed to the ullage that gas ingestion became possible. This device was designed for the low-gravity environment of the shuttle and for relatively high flow rates, requiring only at few minutes to empty the tank. The device was insensitive to the ambient shuttle accelerations around 10^{-4} g and could readily withstand reaction control thruster firings of up to 10^{-1} g. Flow rates of up to 1.2 gallons per minute were used, that could empty the tank in 3.2 minutes.

The expulsion tests have already been described in Section II of this report. Tests 1 and 3 were identical, expelling the tank at 1.2 gpm in both cases. In test 8 the flow rate was 0.7 gpm. For test 2 the last 5% of the liquid was expelled with pulsed flow and for test 4 the last 5% was expelled while an adverse acceleration was acting. No ingestion of gas was detected until most of the liquid had been expelled. Gas ingestion was an abrupt event, with the gas-free flow being replaced by flow that was mostly gas. The crew stopped the outflow when gas bubbles were first noticed in the sight glass and estimated the quantity of liquid remaining in the tank. This estimate was only based on the visible liquid and could not include any liquid trapped inside the channels.

The calibrated cylinder, in addition to providing additional liquid for the operation of the experiment, was a means of measuring the expulsion efficiency. By recording the cylinder position at the beginning and end of a test, when the supply tank was completely full of liquid, the change in the readings gave the change in the residual volume in the receiver tank. Various factors influenced the accuracy of this measurement, such as: bubbles in the supply tank, liquid lost due to venting, and the operator response in closing the valve when gas ingestion was detected. This data helped in assessing the expulsion efficiency, but the most accurate evaluation was obtained from the video and still photo data.

The photos clearly showed the location of residual liquid for the two channels facing the front of the modules, but the rear ones could not be seen. Since the photos were taken some time after the flow was stopped they did show the final orientation of the residual liquid, which aided in judging the quantity. The video showed the liquid orientation when gas ingestion occurred and then there usually was some refilling of the channel gap by capillary pumping. Liquid held in the film of the bubbles and around the baffle supports did not wick to the gap as fast as liquid was being expelled. Neither the still photos nor the video permitted the quantity of liquid inside the channels to be determined.

The best expulsion was obtained in test 8. The flow rate was the lower 0.7 gpm value, the orbiter was in free drift during the test and all the smaller bubbles had been allowed to coalesce, so the liquid was well oriented around the channels at gas ingestion. Residual drops of liquid were seen in the channel gap near the girth and the top of the tank. With the conservative assumption that the channels were full at gas ingestion, the expulsion efficiency for this test would be close to 98%. Next best in expulsion efficiency was test 1. There was liquid collected within the gap at the top and girth of the tank, but in this case the entire width of the channel was filled with liquid in those areas. Coalescence of bubbles was still continuing at the end of the test, which held some liquid away from the channels.

The expulsion efficiency was estimated to be somewhat less than 98%. For test 3, which was identical to test 1, the channel gap was full of liquid except for a few bubbles. The difference between tests 1 and 3 appeared to be due to the bubble coalescence and the resulting effect on the liquid orientation. The expulsion efficiency was estimated to be close to 97% for test 3.

In test 4 the tank was expelled to 5%, the bubbles were allowed to coalesce and the liquid was statically oriented about the channels. A four second axial acceleration of 0.01 g was applied and after the acceleration had been acting for 3 seconds, orienting the liquid to the top of the tank, expulsion resumed until gas ingestion. Compared to the other tests, orienting the liquid away from the outlet increased the flow path length inside the channels, increasing those flow losses, and added an adverse hydrostatic pressure. Also, the liquid orientation caused some emptying of the channel gap and displacement of liquid away from the channels. In spite of these adverse conditions the residual fell between those of tests 1 and 3, at about 97.5%. The increases in the flow and hydrostatic pressure differentials were insignificant in comparison to the pressure drop due to flow through the screen, so the screen flow area at which gas ingestion occurred did not change appreciably.

Finally, test 2 had the largest residual. Bubble coalescence was not complete when pulsed expulsion resumed at the 5% fill volume. Pulsing of the outflow, using the toggle valve, produced water hammer type pressure transients that added to the total pressure differential experienced by the screen. While this added effect can not be quantified, a larger residual than the other tests was obtained. At the end of the test the channel gap was full, indicating an expulsion efficiency of at least 97%.

IV. Tank Filling

Two methods of filling the receiver tank under low-gravity conditions were used: an evacuated fill and a vented fill. The evacuated fill requires that the tank be initially evacuated to zero absolute pressure. Then the vent is closed and the tank is filled with only liquid, so the tank and any acquisition devices will be completely filled. This is a fairly simple method of ensuring a tank will fill in zero-g. One disadvantage is that any residual in the tank could be lost when the tank was vented.

As discussed in Section II, the receiver tank could not be evacuated to zero pressure. The small pressure of 1 to 2 in. Hg resulted in some gas being present in the tank and screen device when filling was complete. The filling method was repeatable and was successfully demonstrated in all of those fill tests.

The vented fill method was performed by venting the tank to maintain a constant pressure during fill. For this fill method to be successful, some means of orienting the liquid away from the vent is needed. A baffle was used to suppress the inflowing liquid jet and dissipate the liquid momentum so that it would remain oriented near the inlet and away from the vent.

The Weber number, the ratio of the momentum force to the surface tension force, is used to correlate inflow test results.

$$We = \frac{\rho V^2 r}{2\sigma}$$

where

ρ = density,
V = flow velocity at inlet,
r = inlet radius, and
σ = surface tension.

For test 6, with an inflow rate of 0.1 gpm, the Weber number was 0.6. The test demonstrated that the liquid collected about the inlet, with no jet forming. This result is consistent with prior drop tower tests (as summarized in Ref. 4) that established a critical Weber number of 1.5 for flow into a bare tank. Test 7 had an inflow rate of 0.2 gpm, giving a Weber number of 2.3. A jet did form, as predicted, and it impinged on the barrier. Barriers, depending upon their configuration, can permit much higher stable inflow rates.

Weber numbers from 6 to 180 for stable inflow have been obtained in Earth based tests (also summarized in Ref. 4). In test 8 the flow rate was 0.3 gpm and the Weber number was 5.2. The jet impinged on the baffle and liquid collected about the inlet. All three tests demonstrated stable initial inflow with increasing flow rate.

These tests show that there are two phases to the vented fill of a tank. The first phase, the initial inflow, that fills to about 30% was discussed above. For the first phase it is enough control the momentum of the inflowing liquid, to prevent liquid jets or excessive flow along the tank wall. The second phase is the final filling of the tank, up to the point at which liquid free venting can no longer be maintained. None of the prior Earth based tests have been able to simulate this phase of vented tank fill because of the need for long test times.

For filling to continue in the final phase, some means of orienting the ullage bubble at the tank vent is needed. In a cylindrical tank the liquid tends to remain oriented to one end of the tank with some stability, under the influence of surface tension. When the diameter of the ullage bubble became less than the diameter of the tank, control of its orientation would be lost. For the FARE receiver tank the intention was to use the liquid momentum to control the orientation of the ullage. The thought was that a uniform flow directed across the ullage bubble would push the bubble toward the top of the tank where the vent was located. The baffle was solid in the center to avoid the jet, but perforated on the periphery to allow some flow to pass through rather than having all the flow directed toward the wall.

When the receiver tank reached about 60% fill, the ullage bubble was tangent to the top side of the baffle and the inner surface of the screen device channels on the opposite side of the tank. If the ullage bubble were to only remain in contact with the baffle, so the filling occurred at the top of the tank, liquid would begin to cover the vent holes when the tank was about 70% full. The vent tube was aligned with the channel device axis, rather than the inflow port and baffle axis, which was also considered in determining when the port would first become covered. To continue beyond 70% fill the ullage had to be oriented to the top of the tank over the vent. At the lowest flow rate, in test 6, the tank could be vented until reaching an estimated 60% fill, which is close to the minimum that can be achieved with no means of orienting the ullage bubble. It is doubtful that using a smaller flow rate would have improved the fill. Doubling the flow rate for test 7 resulted in an increase in the fill volume before the tank could no longer be vented. The crew was very emphatic that the fill was better than test 6. Estimates from the video were that the tank could have been as much as 80% full. The flow acted to orient the ullage bubble off of the baffle and toward the top of the tank, but liquid still collected at the top of the tank to finally prevent gas venting. However a further increase in flow rate for test 8 resulted in a decrease in the fill, to a value estimated to be somewhere between test 6 and 7, at about 70%.

It has been suggested that a longer vent tube would have improved the fill. This is true to some extent, but it assumes that the exact orientation of the ullage bubble is known. Even for this configuration and an ullage orientation obtained, there would have been some improvement. Beyond some fill volume, trying to chase an ullage with an orientation that is not specifically defined would no longer offer an improvement.

V. Liquid Slosh

Test 4 provided a demonstration of the effects of accelerations on the behavior of the liquid. The applied accelerations and the test results are described in section II. In this section an analytical correlation of selected tests is described.

A computational fluid dynamics model, FLOW-3D (a commercial product of Flow Science, Inc.) was used for the correlation. A representation of the FARE receiver tank was developed for the analytical model. The tank axis was aligned with the baffle, since it had a significant effect on the liquid orientation. The channels of the screen device were aligned with the same axis, rather than being canted by 15 degrees. This change was necessary to align the channels with the computational mesh so that their shape could be properly resolved. For the same reason the channels were also rotated 45 degrees about their axis so that the front view was directly at one channel rather than between two channels as it was in the video. The channels had a significant effect on the liquid motion, but it was not expected that the orientation was as important. The correlation concentrated on the tests performed with a 50% fill. The liquid would be difficult to resolve at the 5% fill level, so no attempt was made to correlate those tests. The best values found for the accelerations were 9.1×10^{-3}g for +Y and -Y, and 1.4×10^{-2}g for +Z. These accelerations were rotated 15° so that they align with the tank as in the test. The duration of the acceleration was 4 seconds in all cases.

The liquid motion was found to be best represented by a model that included the baffle, channels, and included the liquid surface tension. None of the viscous dissipation models were used, relying only on the inherent numerical dissipation in the model. Prior successful correlations have used the same approach.

Photographs of the slosh tests, obtained by freezing a video frame, are shown in Figures 3, 5 and 7. There is a two second interval between photos. The liquid motion calculated using FLOW-3D follows the corresponding photos in Figures 4, 6 and 8. The FLOW-3D pictures are arrayed the same as the photos. The video tape provides a better quality image of the tests than the photos, so it was also used to make the comparison with the analysis.

The -Y test still had quite a few bubbles in the liquid so the comparison of the test and analysis was difficult and the bubbles may have influenced the energy dissipation. What can be seen of the basic liquid motion and the time required for the liquid to come to rest (15 seconds) appear to compare favorably. When the +Y test was performed the bubbles had coalesced. The basic liquid motion of the analysis was the same as the test. A liquid wave formed, moved half way across the tank from the left, then swirled along the wall and returned to the initial orientation. The bulk liquid was at rest in 16 seconds. The +Z case analysis also closely matched the liquid motion seen in the video of the test. Due to the larger acceleration, an jet of liquid formed and traveled across the tank, interacting with the baffle. The motion made a transition to flow along the wall and returned to its original orientation within 20 seconds.

In all the tests there was a final orienting of the liquid driven by capillary forces, after the bulk motion of the liquid had damped. In the video view of the top of the tank this motion was most obvious. Liquid slowly flowed away from the top of the tank orienting toward the bottom, about the baffle. The baffle was influencing the static orientation of the liquid once the momentum was sufficiently damped. For the three correlations above, the video shows the capillary orientation continuing until around 30 to 35 seconds from the beginning of the test. The analytical model predicted some of this capillary reorientation, but in general there was more liquid near the top of the tank than was observed in the video.

Figure 3. -Y acceleration, 50% fill

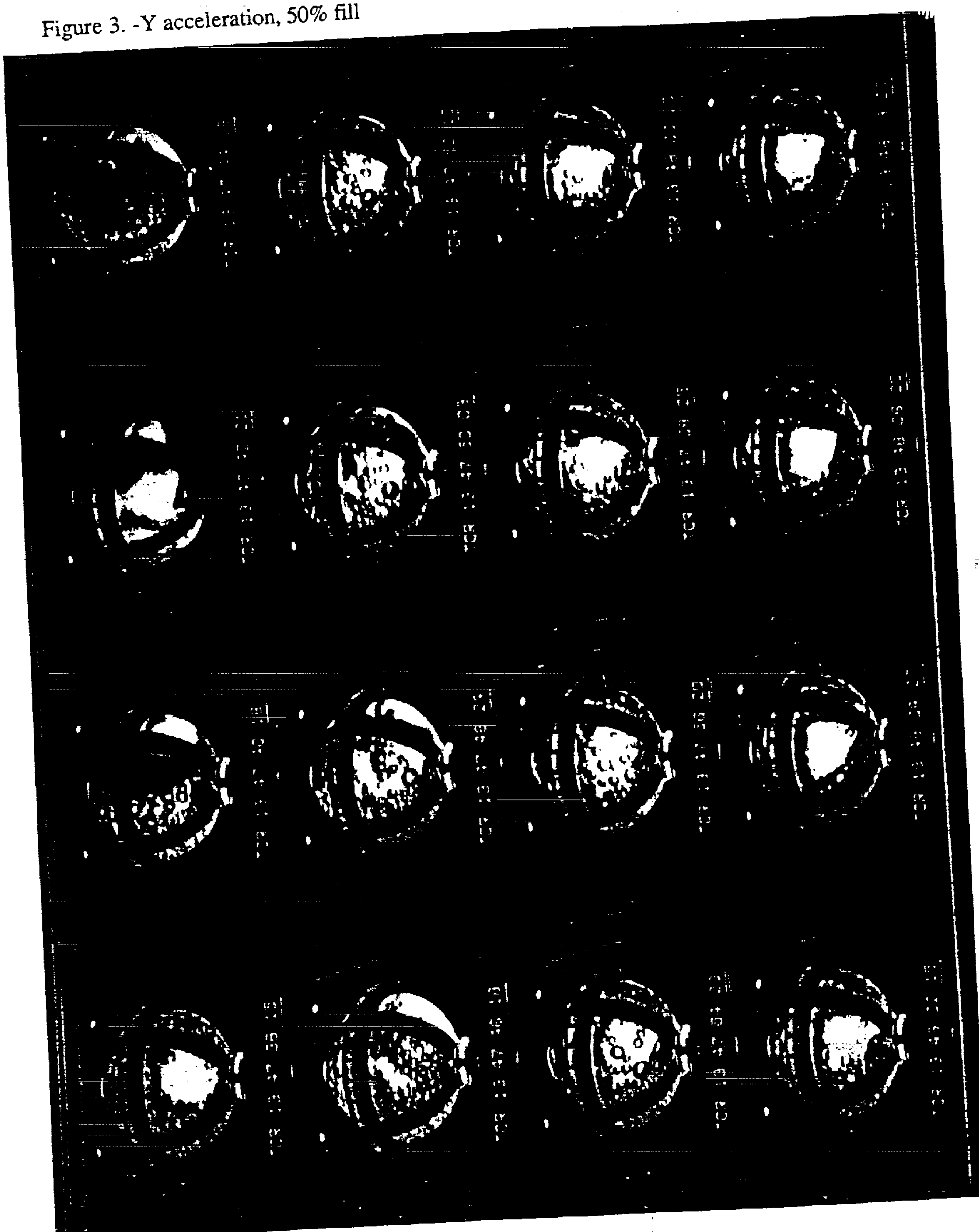


Figure 4. FLOW-3D analysis (-Y, 50% fill)

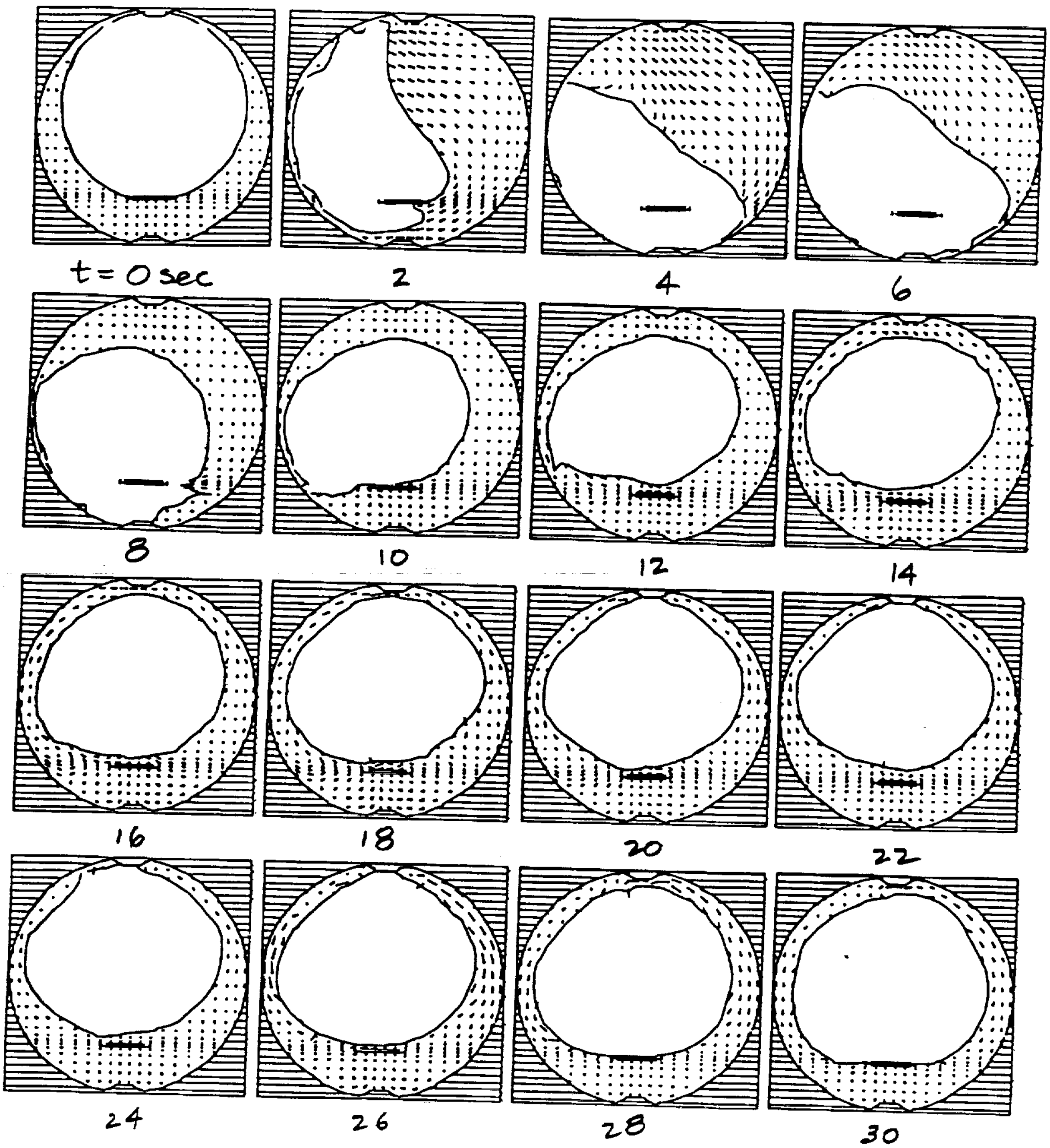


Figure 5. +Y acceleration, 50% fill

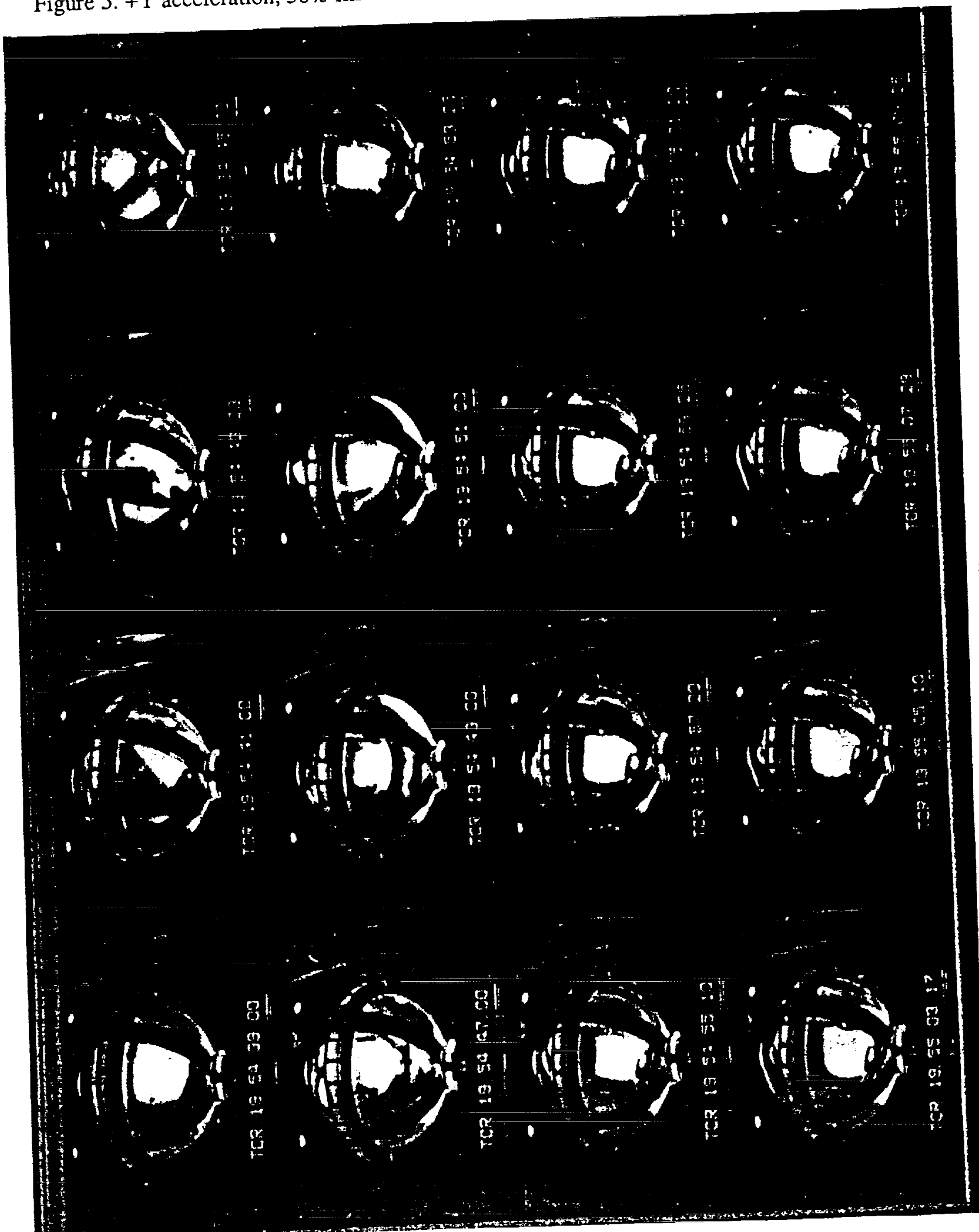


Figure 6. FLOW-3D analysis (+Y , 50% fill)

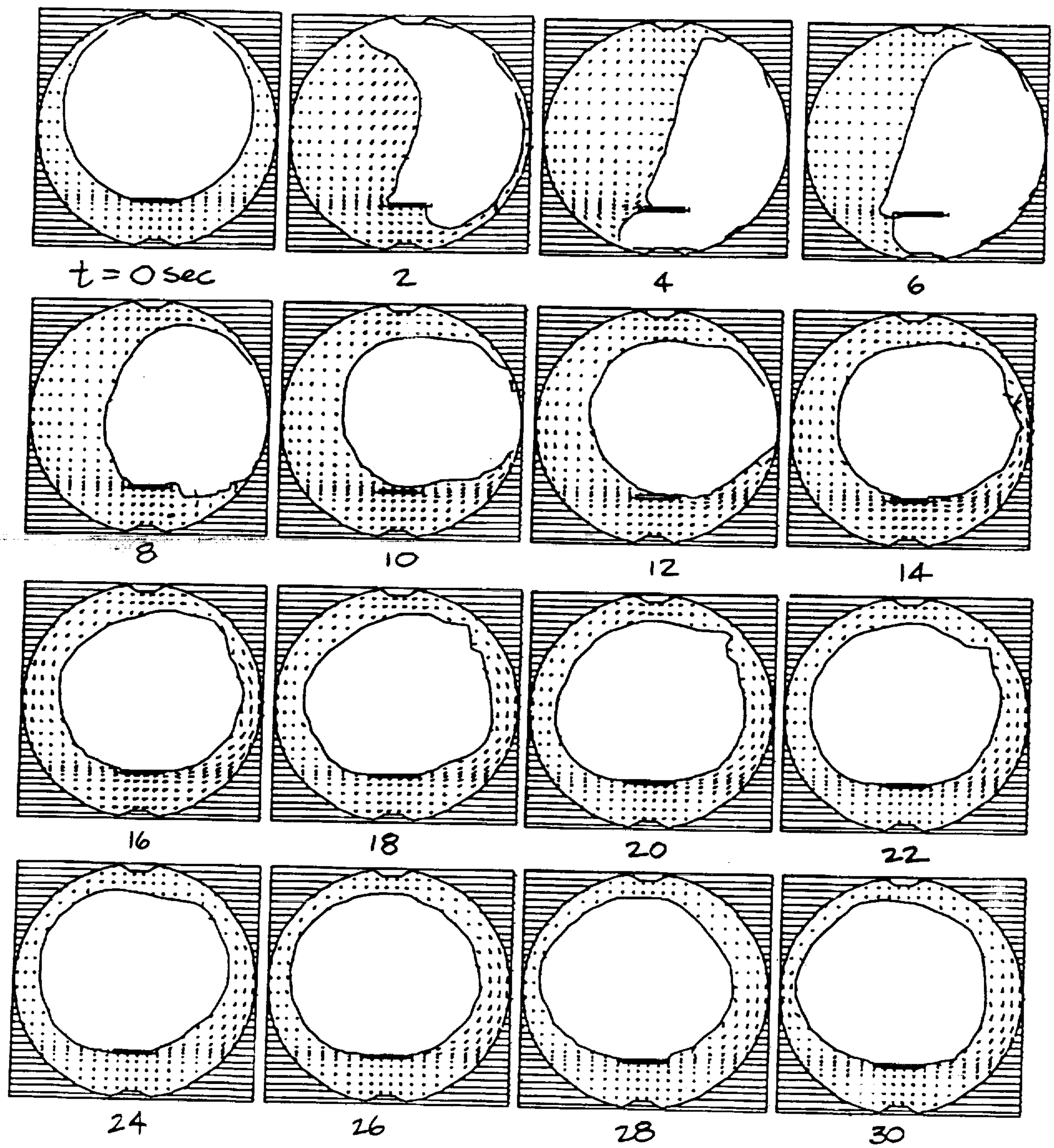


Figure 7. +Z acceleration, 50% fill

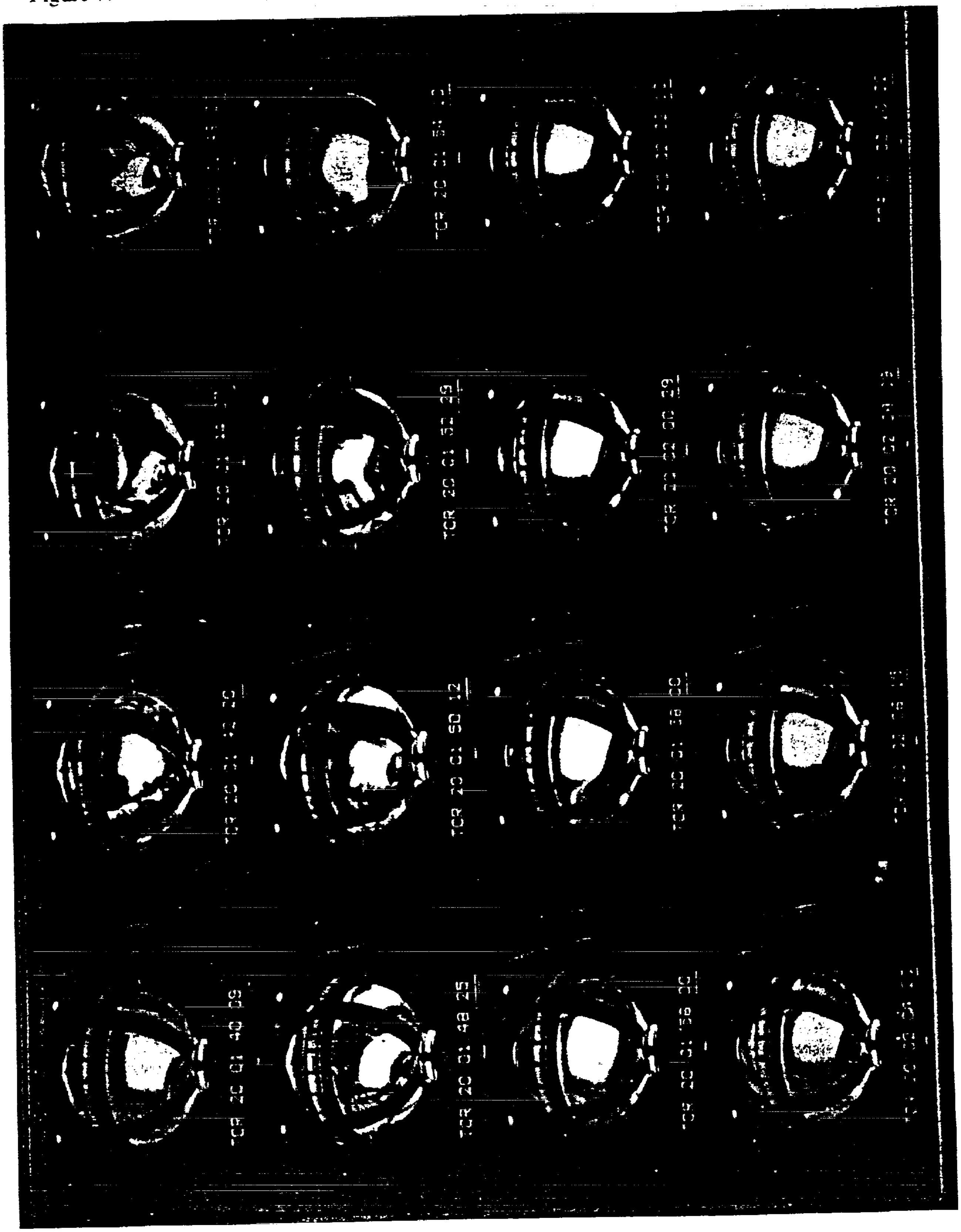
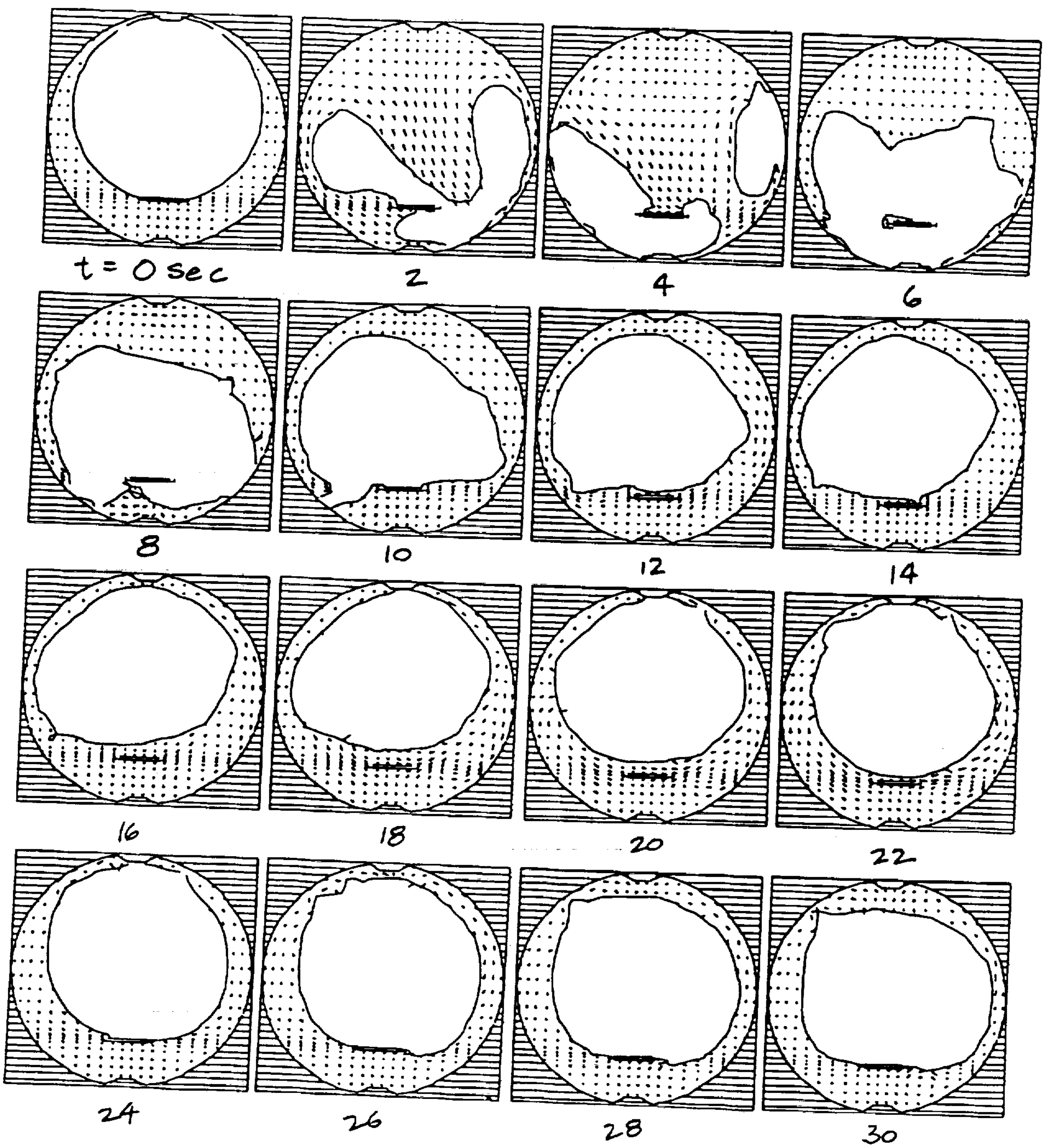


Figure 8. FLOW-3D analysis (+Z, 50% fill)



VI. Conclusions

The tests performed with this shuttle middeck experiment were highly successful. All the hardware functioned as required and all the tests were completed as planned. The tests provided a unique opportunity to directly view the operation of a subscale tank system under extended low-gravity conditions. The astronauts ability to directly observe the experiment and react to what they saw added considerably to the success of the experiment.

While screen type propellant acquisition devices have been well proven in a number of flight applications, this opportunity to see the operation added to the understanding of how they function. This device was designed, with large margins, to operate in the shuttle environment. It was fairly insensitive to the effects of flow, including pulses and accelerations. As high an expulsion efficiency as can be expected for a device of this size and volume, was obtained. Performance matching the pre-flight predictions was obtained.

One of the more interesting aspects of the expulsion tests was the behavior of the liquid in the gap between the channel and the tank wall. In all of the expulsions to gas ingestion this gap was almost empty when gas ingestion occurred, as expected. When flow was stopped, the gap refilled for most of the tests. This result indicated that liquid was being withdrawn from the gap by the outflow at a faster rate than wicking of the liquid from elsewhere in the tank could refill the gap. In some cases the delayed coalescence of bubbles further slowed this wicking process.

One test demonstrated that bubbles that entered the channel gap could remain in place after the tank was refilled. In this case the bubbles were positioned so there was no capillary driving pressure to displace them. In many other cases, bubbles that entered the gap during expulsion were expelled from the gap as it filled with liquid. It was also noted that small bubbles generated by pressurization entered the gap and would travel along it, driven by capillary pressure, to accumulate at one end of the channel.

This fluid behavior needs to be considered in the design of the channel gap if the performance of the device is to be optimized. Bubbles trapped in critical locations could reduce the device performance. Consideration should be given to controlling the maximum gap and avoiding changes in gap that could result in bubble entrapment. Tapering the gap from one end of the tank to the other is one approach to improve the filling of the gap with liquid.

The evacuated fill tests again confirmed the success of this fill method. The surface tension device was filled along with the tank so that gas free expulsion of liquid could resume. The fill was successful even though the tank could not be vented to the vapor pressure of the liquid, allowing some non-condensible gas to remain. When applying this fill method to flight systems though, every effort should be made to ensure that all non-condensible gases are purged before filling the tank.

The vented tank fill was reasonably successful, but it did demonstrate that further investigation of this fill method is needed to give sufficient confidence to apply it to flight systems. The results for initial filling phase of the tank were consistent with prior tests and analysis. The success of this phase of the fill process can be predicted based on a Weber number correlation. A simple baffle configuration was adequate in controlling the inflow at fairly large inflow rates.

During the final filling of the tank the ullage centered with respect to the inflow axis and one fill that approached 80% was achieved. Fills of at least 60% appear certain, with this tank configuration, but beyond that point the factors influencing the ullage orientation with respect to the vent port could not be clearly established. Nor did there appear to be any simple relation between the inflow rate and the success of the fill. With a more positive means of orienting the ullage or different vent port configuration, filling to higher levels would be expected. It is speculated that more success would be achieved in filling a cylindrical tank, using this method, due to the inherent orientation of the interface in zero-g, at least as long as the ullage completely filled the tank diameter. Additional testing, considering various tank and inflow port configurations along with flow rate, is needed.

The liquid slosh tests provided a dramatic demonstration of the dynamics of the liquid in a maneuvering spacecraft. Large amplitude motion resulted due to the dominance of acceleration forces over the capillary forces. However, following the acceleration the surface tension forces played a significant role in bringing the liquid to rest. At the 50% fill level, the relatively small inflow baffle caused the liquid to orient around it and collect toward the bottom of the tank, symmetric with the baffle axis. At 5% fill the channels of the screen device quickly collected the liquid set in motion by the accelerations. The successful correlation of the slosh tests at 50% fill using a computational fluid dynamic model added to the confidence in the use of such models. Accurate modeling of the liquid motion required that internal tank details (that is, the baffle and screen channels) and the surface tension of the liquid be included.

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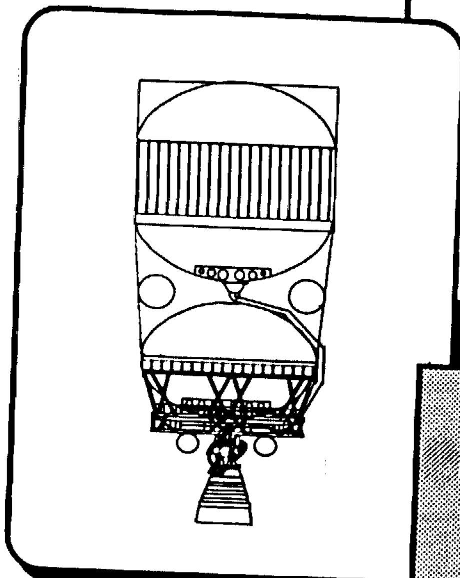
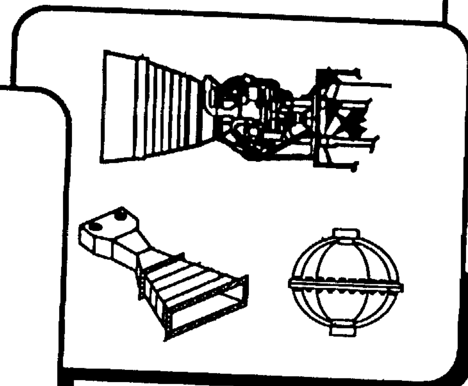
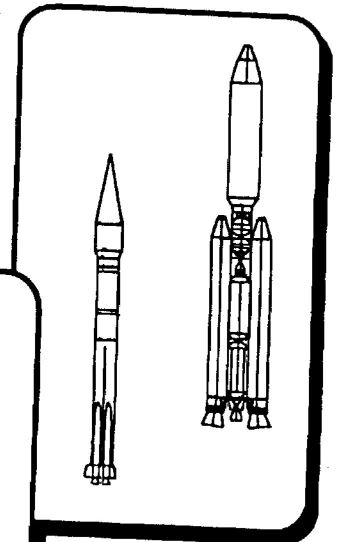
Technical Directive 16

Upper Stage Requirements and Architecture Study







Space Transfer Vehicle

MSFC

Technical Directive 16 - Upper Stage Requirements & Architecture Study
NASA Headquarters Technical Interchange Meeting
May 25, 1993

STV Study Team
- MSFC/PT
- Martin Marietta

JH930520-01A

Topics

MSFC

Topic	Presenter
◦ Introduction	Dan O'Neil (MSFC)
◦ Requirements	John Hodge (MMC)
◦ Architectures	Bob Spencer (MMC)
- Baseline/Alternative Concepts	
- Option Roadmaps	
- Upper Stage Specifications Sheets	
- Technology	
- Development Infrastructure	
◦ Significant Business Factors	John Hodge (MMC)
◦ Future Vision	Dan O'Neil (MSFC)

Upper Stage Req'ts & Architecture Study

MSFC

Objectives

Formalize An Upper Stage Requirements & Architecture That:

- Defines Upper Stage Concepts that Enhance Industry Competitiveness and a Growth Path for Future Systems
- Defines Associated Technology and Infrastructure Requirements
- Provides Context for Upper Stage Concepts within Current Upper Stage Market
- Builds a Foundation for Phase A Studies Answering Specific Program Questions

Accomplishments To Date

Established a Baseline Upper Stage Approach For:

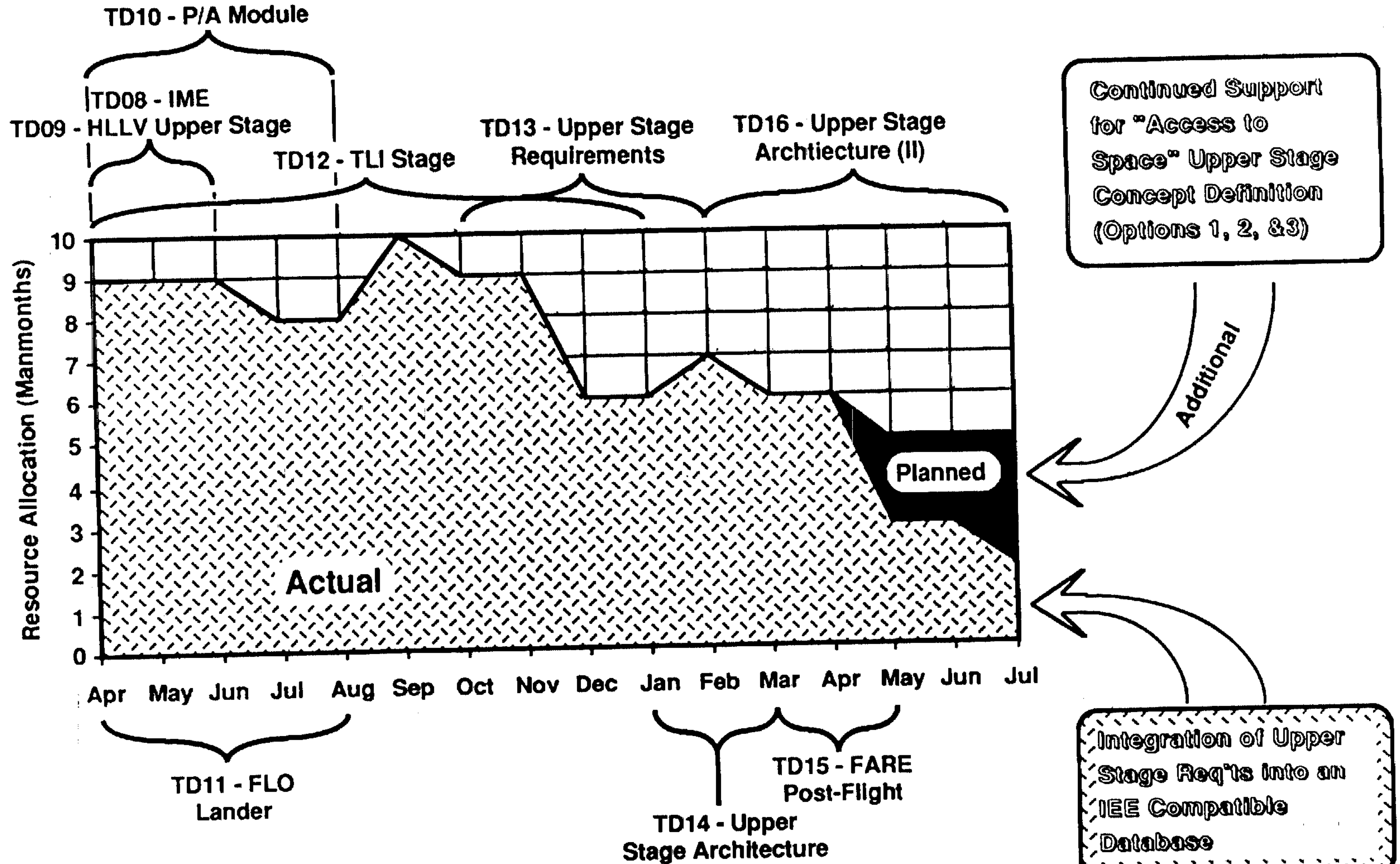
- Programmatics/Marketing Analysis
- Technical Requirements Document
- Upper Stage Architecture Options

Technical Directive 16 Efforts

Analyses & Trade Studies That Support

- Refinement of the Baseline Architectures
  - Concepts
  - Technologies
- Development of Quantitative Req'ts Rational
- Identification Of Dual-Use Technologies and the Infrastructure They Require
- Establishing Relationships Between the Marketing, Req'ts, & Architectural Pieces of the Approach

Resource Utilization



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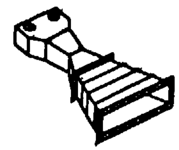
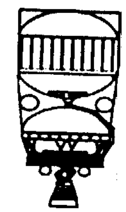
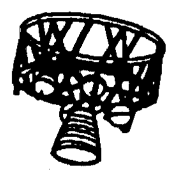
Products/Customers

MSFC

TASK		PRODUCT/CUSTOMER	
TD08	Integrated Modular Engine Study		MSFC/EP John Cramer Aerojet
TD09	HLLV Upper Stage Definition Study		MSFC/PT Warren Pattison - Establishment of Upper Stage/Mission Database
TD10	Propulsion/Avionics Module Study		MSFC/PT Warren Pattison - Concept to Support System & Subsystem Commonality
TD11	FLO Lander Alternative Concept Study		MSFC/PT - Warren Pattison JSC/EXPO - Ron Kahl - Alternative Lander Concept Capable of Meeting Initial Req'ts

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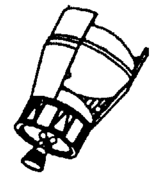
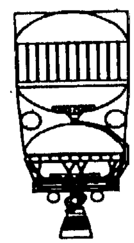
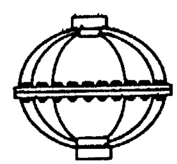
Products/Customers

MSFC

TASK		PRODUCT/CUSTOMER	
TD12	TLI Stage Study		MSFC/PT - Dan O'Neil JSC/EXPO - Dwayne Weary - TLI Stage Concept (SatV/NLS) - Systems Engineering Approach
TD13	Upper Stage Requirements Definition		MSFC/PT Dan O'Neil MSFC/PD Steve Cook MSFC/EP Rick Bachtel - Upper Stage Requirements Database - Options 1, 2, & 3 Upper Stage Concepts - Technology Infusion Plan
TD14	Upper Stage Architecture Study		
TD16	Upper Stage Architecture Study (II)		
TD15	FARE Post Flight Data Reduction		MSFC/EP - Susan Driscoll - Analysis of Data Gathered During the 1992 Flight Experiment

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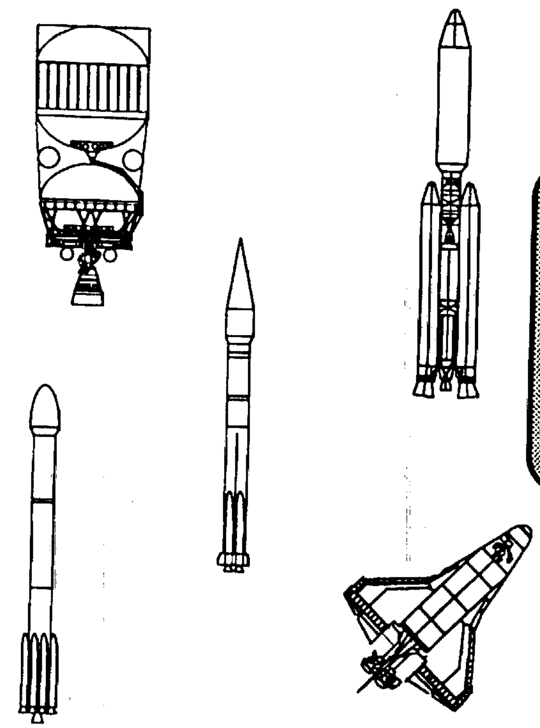

Upper Stage Requirements

MSFC

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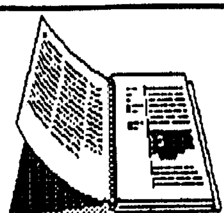
MARTIN MARIETTA

Lifecycle of the Technical Reqts Document

MSFC

Release Preliminary TRD to NASA



Preliminary TRD

- Preliminary TRD Based on Recent Upper Stage Studies and Marketing Analysis

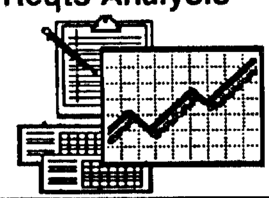
Validate Initial Req'ts

NASA	Internal
Military	External
Commercial	

- Ensure Complete Integration of All Customers' Reqts

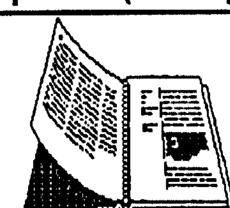
Distribute to Contractors

Reqts Analysis



- Identify Driving Reqts to Customer
- Integrate into Reqts Docs
- Provide Basis for Concept Development

Update (If Req'd)



Final TRD

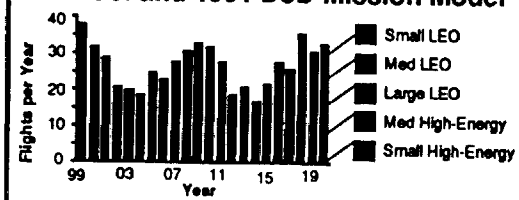
Key Upper Stage Requirements

Analyses Provide Understanding of Key Requirements

MSFC

Reqt: U/S Missions —
Small, Medium, & Large Leo P/L
Small, Medium, & Large High-Energy P/L

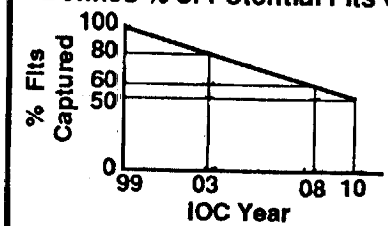
- 592 Missions Identified from HQ Code D Model and 1991 DoD Mission Model



- May Identify Need for a Family of U/S
- Missions Require KSC & VAFB Sites

Reqt: IOC — 2003

- Drives Technology Selection
- Defines % of Potential Flts Captured



- Drives Annual/Peak Funding Reqt

Reqt: Multi-Launch System Compatibility

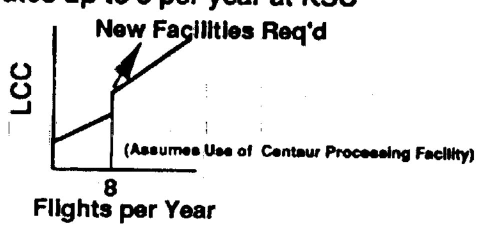
Potential Launch Systems Identified

Opt 1: STS, Titan IV, Spacelifter
Opt 2: STS, Titan IV, Spacelifter, CTV/ PLS, Explor Vehicle
Opt 3: STS, Titan IV, SSTO, Explor Vehicle

- Multi-system compatibility drives need for Std Interface
- Std I/F reduces (~50%) individual LV/Payload integration analysis and planning cost (Average Cost $25-30M)

Reqt: Launch Rate — 4 to TBD/year

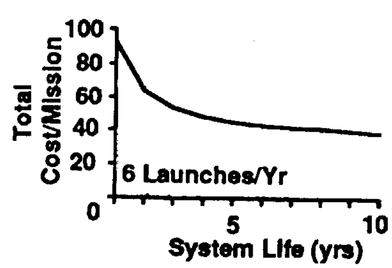
- Allows use of existing facilities for rates up to 8 per year at KSC



- High rates emphasize need for new infrastructure to improve reliability & efficiency while reducing manpower and cutting operating costs

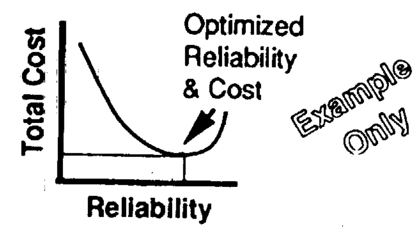
Reqt: System Life — 20 yrs

- Recent USAF studies suggest that a longer sytem life, and thus more vehicles, result in a lower overall cost/mission



Reqt: Reliability — 0.98

- Strong Influence on Total System Cost



- Defines Technology Development Req'd and/or System Redundancy

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LR934027-Key Reqt

Task Interactions

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Requirements Task & Marketing Plan Task

Marketing Plan Task Influences TRD Requirements

Marketing Info	→	Requirement
Economic Environment	→	IOC, Technology Which Balances Key Parameters (i.e. Performance, Operability) with Cost
Technological Environment		
Existing/Planned Launch Systems	→	LV/Upper Stage Compatability
Availability of Technology	→	Feasible Upper Stage Missions
Target Customers		
NASA	→	Safety
Air Force	→	Reliable Access
Commercial	→	Increased Availability

Requirements Task & Architecture/Concept Dev Task

Requirements Task Influences Architecture/Concept Development

Requirement	→	Architecture/Concept Impact
IOC	→	Technology Selection
Flight Rates	→	Facilities Required
Launch System(s)	→	Payload Capability
Missions	→	Number of Vehicles, Launch Site(s) Required
Operability	→	Payload Adapter Configuration
Mission Life	→	Power System Selection
Reliability	→	Hardware Redundancy

Analyses Summary

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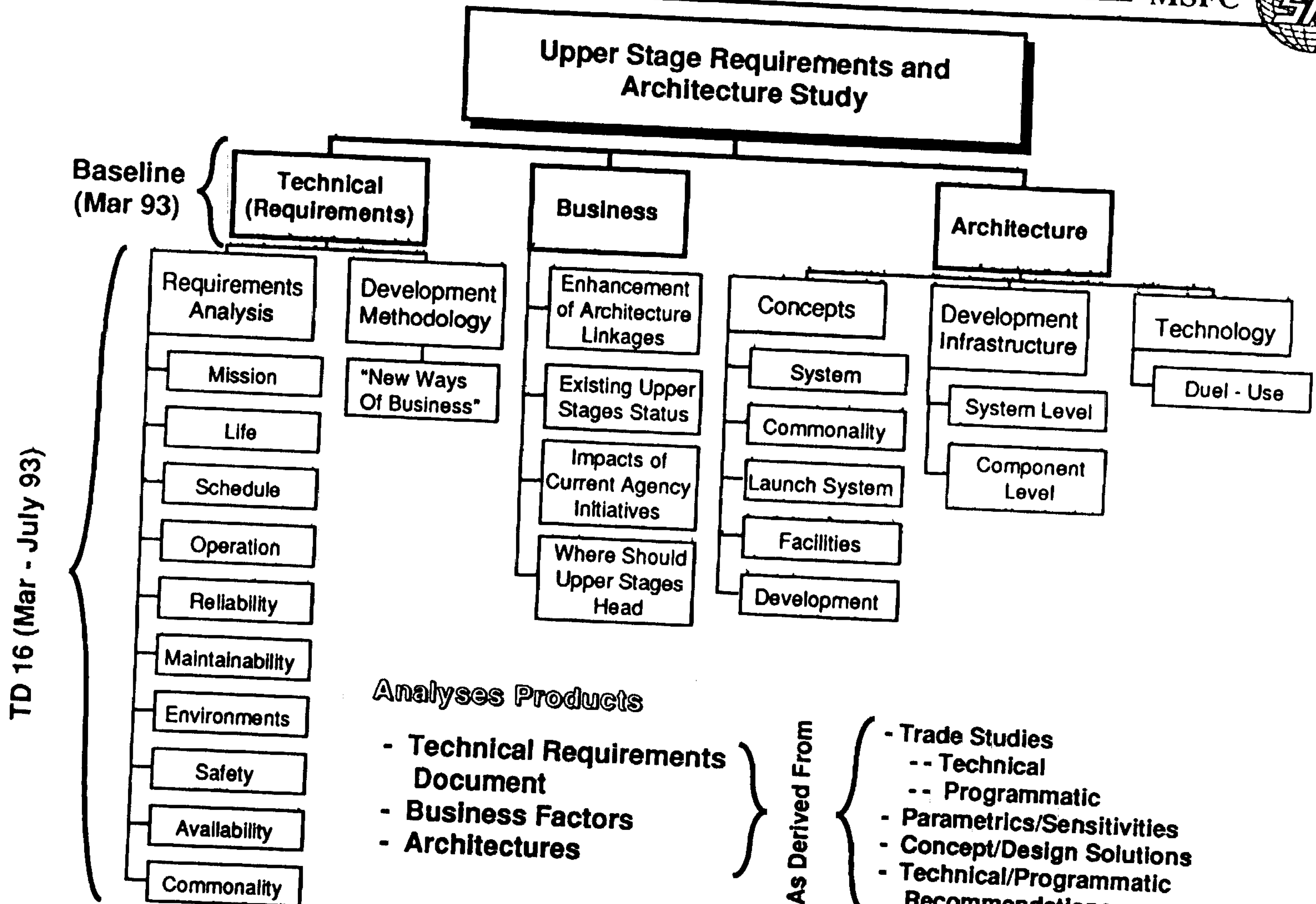
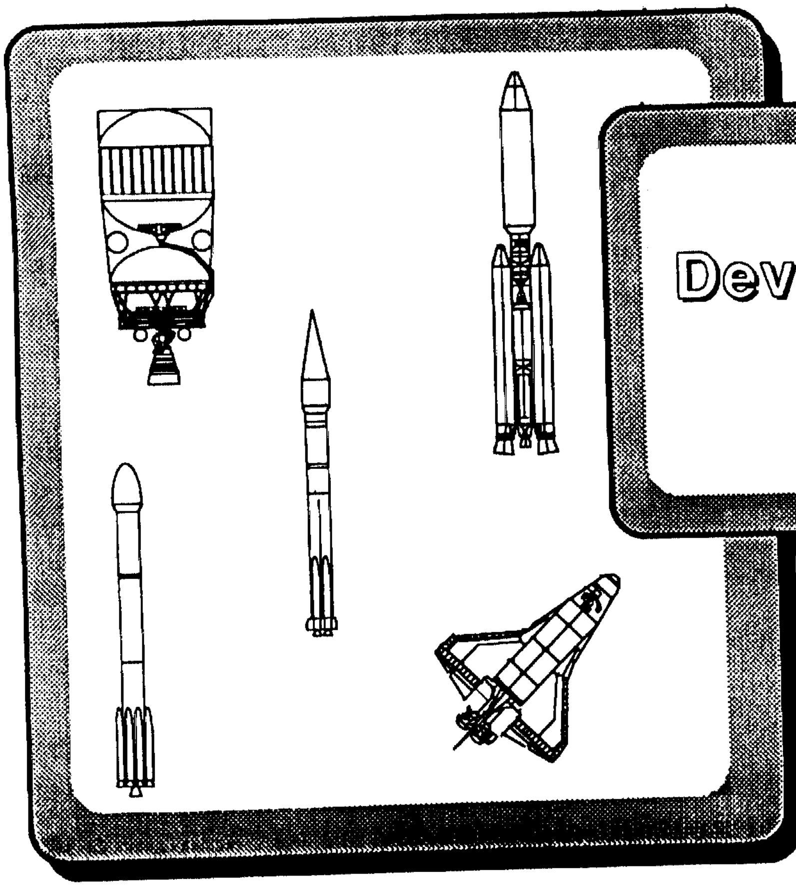


Analyses Products

- Technical Requirements Document
- Business Factors
- Architectures

As Derived From

- Trade Studies
  - -- Technical
  - -- Programmatic
- Parametrics/Sensitivities
- Concept/Design Solutions
- Technical/Programmatic Recommendations



Upper Stage Development/Evolutionary Architecture Review

MSFC

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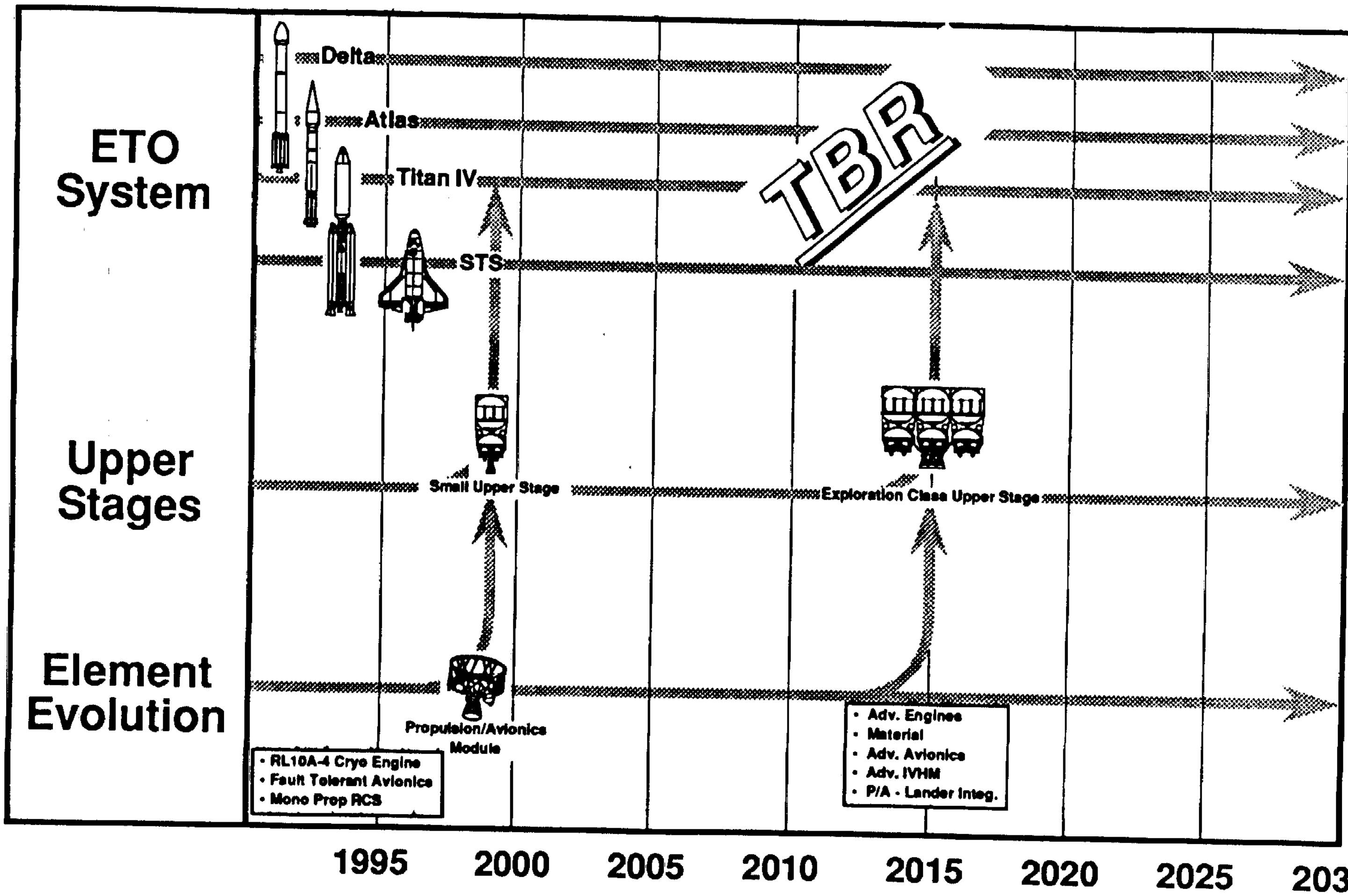
Upper Stage Dev./Evol. Arch. - Option #1

MSFC

- Shuttle Upgrades To Fly Through 2030 w/ Current ELV Fleet



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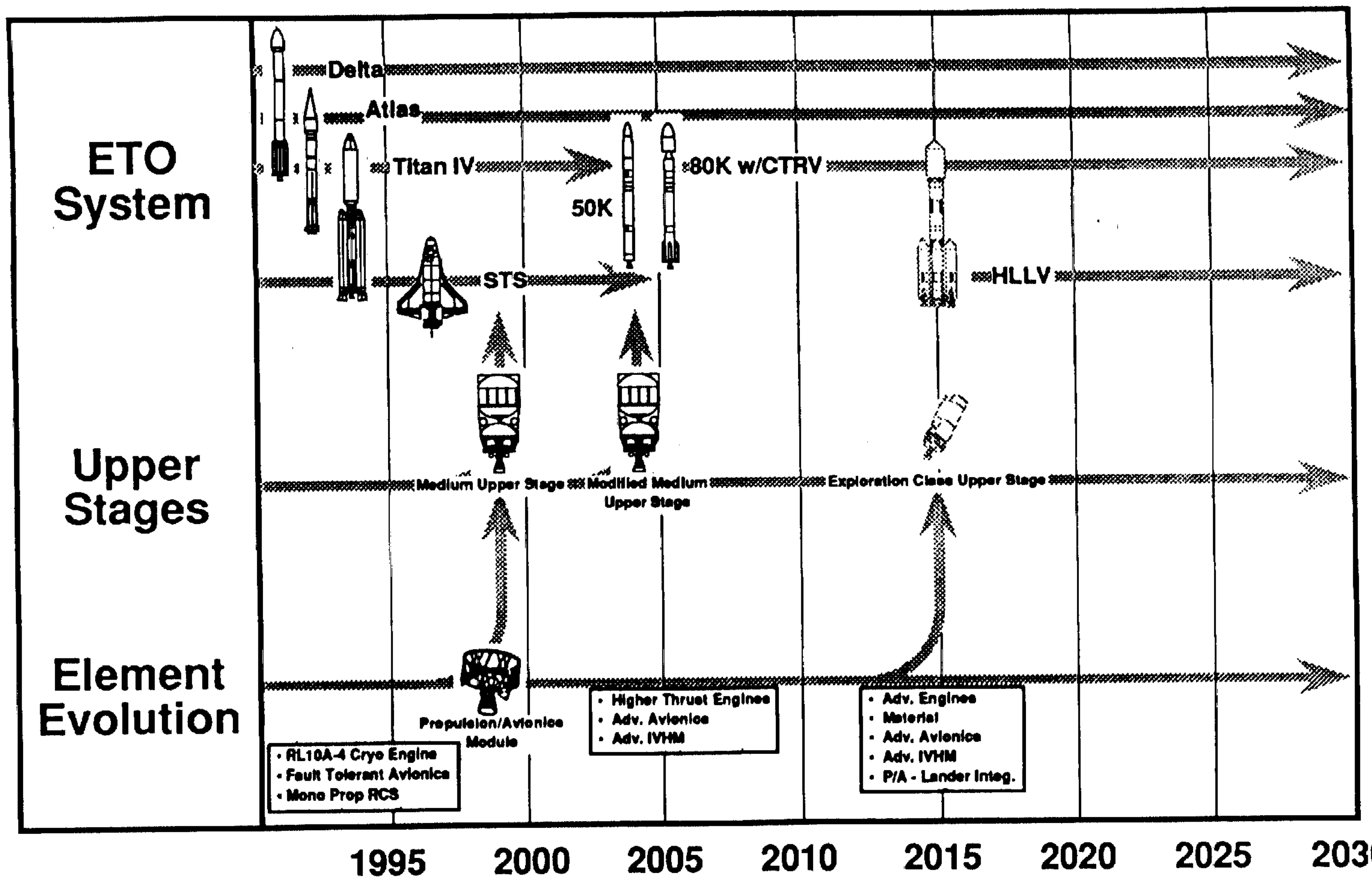
Upper Stage Dev./Evol. Arch. - Option #2

MSFC

- Shuttle Phaseout By 2005 (PLS/CTRV) w/Current ELV Fleet



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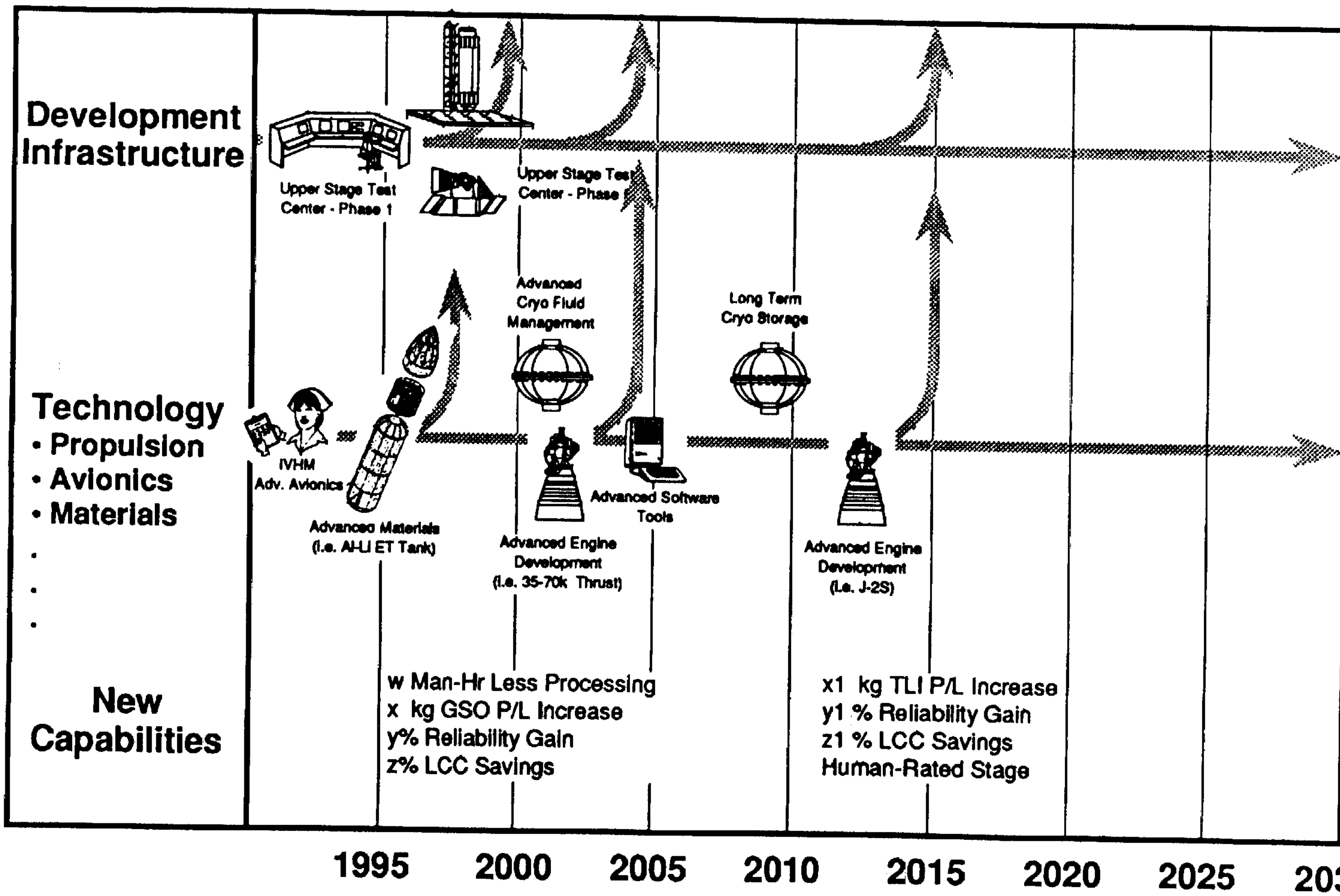
Upper Stage Dev. Infrastructure - Option #2

MSFC

- Shuttle Phaseout By 2005 (PLS/CTRV) w/Current ELV Fleet



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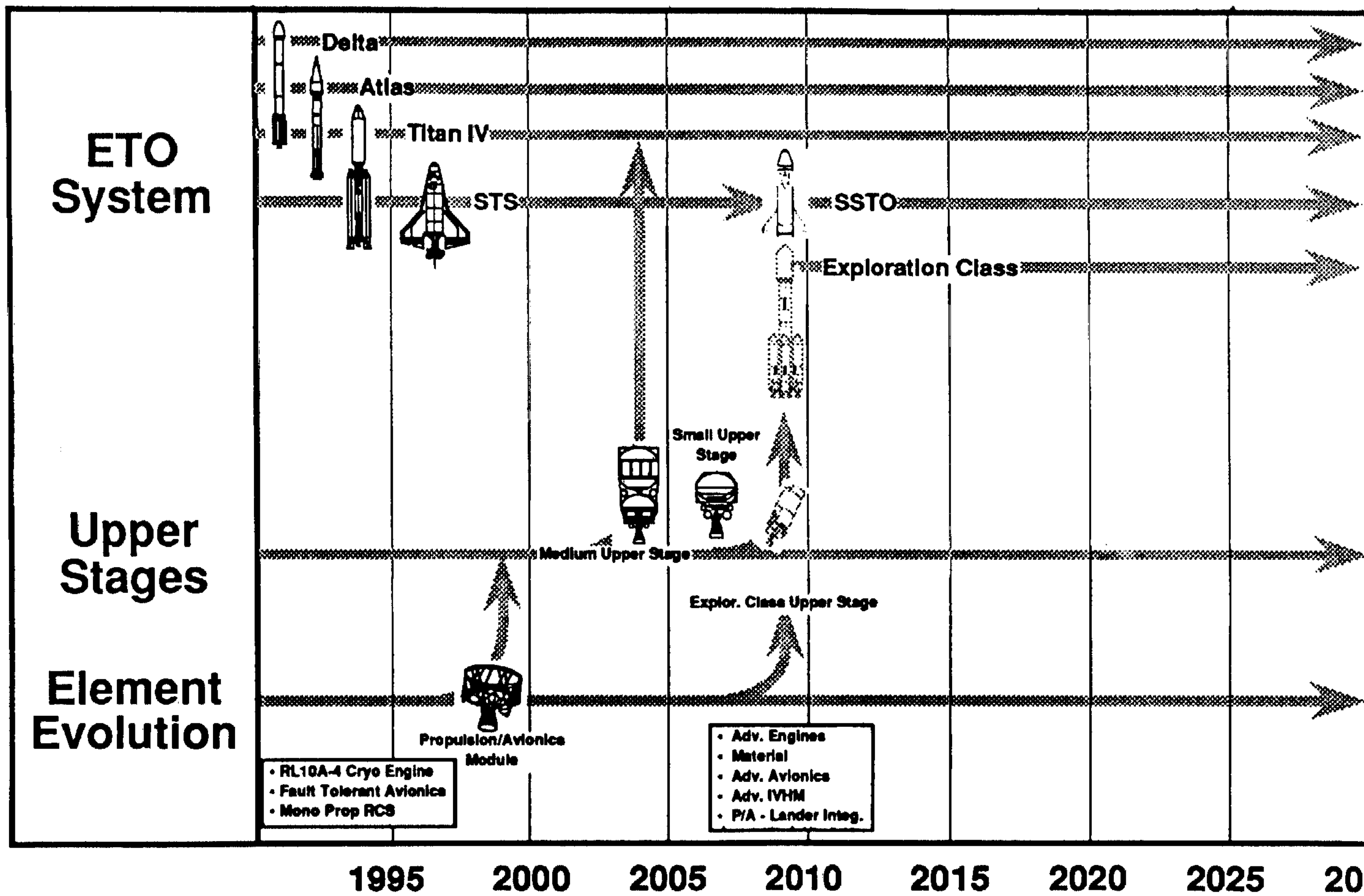
Upper Stage Dev./Evol. Arch. - Option #3

MSFC

- Shuttle Phaseout By 2008 (SSTO Vehicle) w/ Current 50k Fleet



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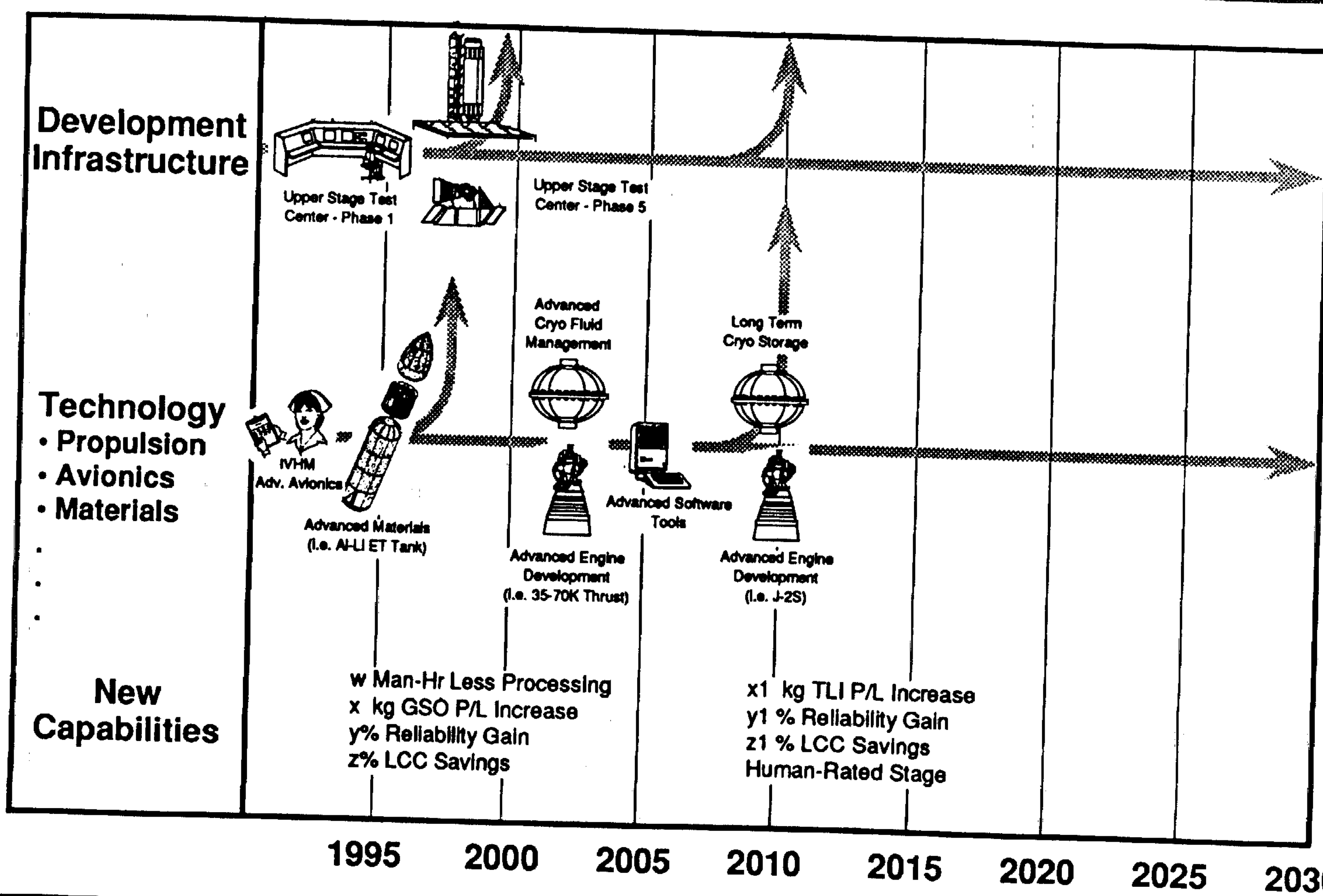
Upper Stage Dev. Infrastructure - Option #3

- Shuttle Phaseout By 2008 (SSTO Vehicle) w/ Current 50k Fleet

MSFC



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Access To Space Arch. Vehicle Options

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Launch Vehicles Description	Option #1				Option #2				Option #3			
	LEO lbs	P/L Dia.	P/L Lng.	G's A/L	LEO lbs	P/L Dia.	P/L Lng.	G's A/L	LEO lbs	P/L Dia.	P/L Lng.	G's A/L
STS	50k	15	60	3.2/2.5	50k	15	60	3.2/2.5	50k	15	60	3.2/2.5
STS Upgrades	?	15	60	?	?	15	60	?				
ELV's												
Delta 7920	11k	9.1	12	6/2	11k	9.1	12	6/2	11k	9.1	12	6/2
Atlas IIAS	18.5k	12	13.7	6/2	18.5k	12	13.7	6/2	18.5k	12	13.7	6/2
Titan IV	40k	15	66	6.5/1.5	40k	15	66	6.5/1.5	40k	15	66	6.5/1.5
ELV's Upgrades												
Titan IV/SRMU	48k	15	66	?	48k	15	66	?				
Spacelifter												
20K	20k	15	25	≤ 5	20k	15	25	≤ 5				
50K	50k	15	60	≤ 5	50k	15	60	≤ 5				
Vehicle/CTV/PLS												
50k					50k	?	?	?				
80k					80k	?	?	?				
SSTO/TSTO									45k	15	30	4.5/5.4
Exploration Vehicle					310k	33	60	4	310k	33	60	4

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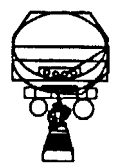
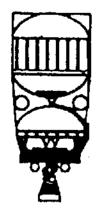
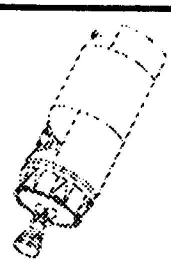
Access To Space Upper Stage Options

MSFC

Upper Stages	Mass Properties	Architecture Option	Poss. Launch Veh.
 Small US	Dry Mass 0 Propellant 0 RCS Prop 0 Total Mass 0 Contingency 20% Diameter 0 Length 0	Option #1 Option #2 Option #3	STS, Expendables, Spacelifter STS,Expendables, Spacelifter,CTV/PLS STS, Expend., SSTO
 Medium US	Dry Mass 3,349 Propellant 23,024 RCS Prop 136 Total Mass 26,509 Contingency 20% Diameter 4.3 m Length 10.3 m	Option #1 Option #2 Option #3	Expend., Spacelifter Expendables Spacelifter,CTV/PLS Expendables
 Exploration US	Dry Mass 26,600 Propellant 306,331 RCS Prop 1,257 Total Mass 334,188 Contingency 20% Diameter 8.4 m Length 24.7 m	Option #1 Option #2 Option #3	N/A Exploration Vehicle Exploration Vehicle

See Attached Sheets

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Preliminary Data Still In Work

020

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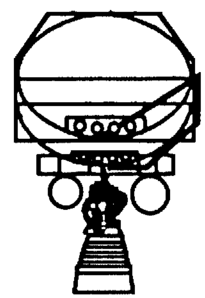
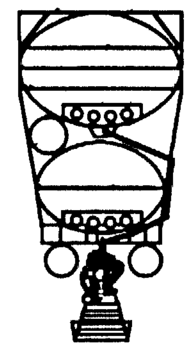
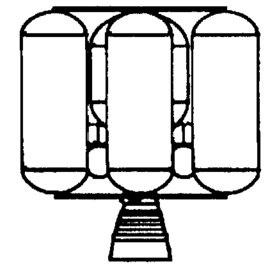
Small Upper Stage CryoTank Config. Options

MSFC

Configuration Options / Physical Characteristics	Common Dome	Conventional	Multi Tank
			
Dry Mass (lb)	3,378	3,243	3,555
Propellant (lb)	14,275	14,275	14,275
RCS Prop. (lb)	252	252	252
Contingency (lb)	20%	20%	20%
Total Mass (lb)	17,905	17,770	18,082
Diameter (ft)	11	10	14.5
Length (ft)	17.5	21.5	15.2
Thrust (lb)	20, 800	20, 800	20, 800
ISP (sec)	449	449	449
Mass Fraction	0.79	0.80	0.79

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RS930507-01B

Existing Upper Stage Configuration Data

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Upper Stages	Prop Type	Engine Type	Thrust (klb)	ISP (sec)	Prp Wgt (klbs)	Diam (ft)	Lng (ft)	Tot Wgt (lb)
PAM-DII	Solid	Star-63 D	17.6	281.7	7.14	5.25	5.8	7,695
Pam - D *(Delta 3rd Stage)*	Solid	Star-48 B	15.1	292.6	4.43	4.10	6.7	4,721
IUS	Solid	Orbus 21 / Orbus 6	45/18	292/300	27.46	9.5	17	32,560
TOS	Solid	SRM-1 Orbus 21	45.0	294.0	21.40	11.2	11.0	23,800
Centaur-IIA *(Atlas IIA)*	LOX/LH2	RL10A-4N	41.6	448.9	37.00	10.0	33.0	41,800
SSPS *(Delta 2nd Stage)*	N2O4/A-50	AJ10-118FJI	9.6	319.4	13.37	8.0	19.6	15,394
Transtage	N204/A-50	AJ10-138 (2x)	16.0	309.1	22.89	10.0	14.8	29,780
TDK-DM (Synthin) *(Proton Upper Stage)*	LOX/Sin	11D58M	19.4	361.0	33.20	12.1	20.6	40,572
H10 *(Ariane 4 - 3rd Stage)*	LOX/LH2	HM7B	14.1	444.2	23.80	8.5	32.5	26,700
L7 *(Ariane 5 - Upper Stage)*	N2O4/MMH	L7	6.1	316.0	15.80	17.7	14.8	20,000
H1 Stage 2 *(Japan H1 Launch Vehicle)*	LOX/LH2	LE5	23.2	447.8	19.40	8.17	33.9	23,400

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SSTO/Existing Upper Stage Compatibility

MSFC

- **Architecture 3 SSTO Upper Stage Analysis - Existing Systems**

LEO: 100 nm @ 28.5°
SSTO Capability: 45000 lbs
ASE Factor: 10% (=» Net SSTO Capability = 40500 lbs)

Stage	Mission	Capability (lbs)	Total Weight (lbs, no ASE)	Diameter (ft)	Length (ft)	Cost ($M)	Comments
PAM-D	GTO	2955	7675	4.0	6.7	15	
PAM-DII	GTO	4090	12203	5.3	5.8	17	
SSPS	GTO	9115	24509	8.0	19.6	14	
	GSO	2513	17907	"	"	11	
TDK-DM (Syn)	GTO	12775	40500	12.1	20.7	?	Prop Offload = 12848 lbs (39%)
	GSO	4640	"	"	"	?	Prop Offload = 4713 lbs (14%)
TDK-DM (Nap)	GTO	12420	"	"	"	?	Prop Offload = 12490 lbs (38%)
	GSO	4272	"	"	"	?	Prop Offload = 4347 lbs (13%)
IUS	GSO	5000	37560	9.5	17.0	70	
TOS	GTO	13395	37195	11.2	11.0	30	
TOS/AMS	GSO	5130	38572	"	16.0	40	
Transtage	GTO	11040	40490	10.0	14.8	25	Prop Offload = 330 lbs (1%)
Ariane 5 2nd Stg (L7 or EPS)	GTO	10960	28751	17.7	14.8	40-50	
	GSO	3260	21055	"	"	"	
Ariane 4 3rd Stg (H10)	GTO	20070	40489	8.5	32.5	35-40	Prop Offload = 6281 lbs (26%)
	GSO	11240	37929	"	"	"	
H-1 2nd Stage	GTO	19080	40498	8.2	33.9	35	Prop Offload = 1982 lbs (10%)
	GSO	7680	31083	"	"	"	
Centaur I	GTO	18680	40500	10.0	30.0	30	Prop Offload = 12484 lbs (42%)
	GSO	10805	"	"	"	"	Prop Offload = 4605 lbs (15%)
Centaur IIA	GTO	18316	"	"	33.0	35	Prop Offload = 19616 lbs (53%)
	GSO	10458	"	"	"	"	Prop Offload = 11759 lbs (32%)

Note: - Cost Numbers Do Not Reflect Payload Integration Costs
- Total Weight includes Upper Stage & Upper Stage Payload

Of The Detailed Existing Upper Stage Assessment Only Two Options are Potential Candidates For The SSTO Vehicle, The IUS and TOS/AMS. The IUS Has a Prohibitive Cost of $70 M and The TOS/AMS is a Paper Stage

SSTO Upper Stage Configuration Matrix

MSFC

- Architecture 3 SSTO Upper Stage Analysis - Optimized Systems

LEO: 100 nm @ 28.5°
SSTO Capability: 45000 lbs
ASE Factor: 10% (=» Net SSTO Capability = 40500 lbs)
Stage MF: 0.85 For "Launch Vehicle Through Capability - Optimized" ONLY!
Payload Bay Dimension: 15ft Dia. X 30ft Lng.

Launch Vehicle Through Capability - Optimized

Engine	Crnt Appl.	Thrust (lbs)	Isp (sec)	Estimates					Oxidizer/Fuel
				Stg Inert (lbs)	Prop. (lbs)	GSO P/L (lbs)	Dia (ft)	Lng (ft)	
RL10A-4	Centaur	20,800	449	4,450	25,220	10,820	13.5	25.5	LOX/LH2
AJ-10-118K	SSPS	9,645	319	5,350	30,330	4,810	8.0	22.6	N2O4/A-50
11D58M (Syn)	TDK-DM	19,400	361	5,020	28,469	7,000	10.0	19.1	LOX/Synthin
11D58M (NAP)	TDK-DM	18,740	352	5,090	28,840	6,570	10.0	19.1	LOX/Naphtyl

Payload Fixed @ 5000lb - Optimized

Engine	Crnt Appl.	Thrust (lbs)	Isp (sec)	Estimates					General Comments	Lnch Veh. P/L Margin
				Stg Inert (lbs)	Prop. (lbs)	GSO P/L (lbs)	Dia (ft)	Lng (ft)		
RL10A-4	Centaur	20,800	449	3,605	14,210	5,000	11.0	17.5	Common Dome	17,685*
"	"	"	"	3,783	14,504	5,000	14.5	15.2	1 LOX & 4 LH2	17,213*
"	"	"	"	3,470	13,987	5,000	10.0	21.5	Conventional Tanks	18,043*
[redacted]										
11D58M (Syn)	TDK-DM	19,400	361	3,590	20,330	5,000	8.0	18.2	Conventional Tanks	11,580*
11D58M (NAP)	TDK-DM	18,740	352	3,870	21,950	5,000	8.0	18.2	Conventional Tanks	9,680*

* Launch Vehicle P/L Margin Accounts For 4,500 lb ASE

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RS930503-02B

Option #3 Small Upper Stage - Summary

- Sizing Groundrules For The Small Upper Stage on The SSTO Vehicle are:
  - Payload Diameter = 15ft
  - Payload Length = 30 ft
  - Required Payload to GSO = 5000 lb
  - Non-Conventional Orientation For Launch

- Of The Detailed Existing Upper Stage Assessment Only Two Options are Potential Candidates For The SSTO Vehicle, The IUS and TOS/AMS. The IUS Has a Prohibitive Cost of $70 M and The TOS/AMS is a Paper Stage

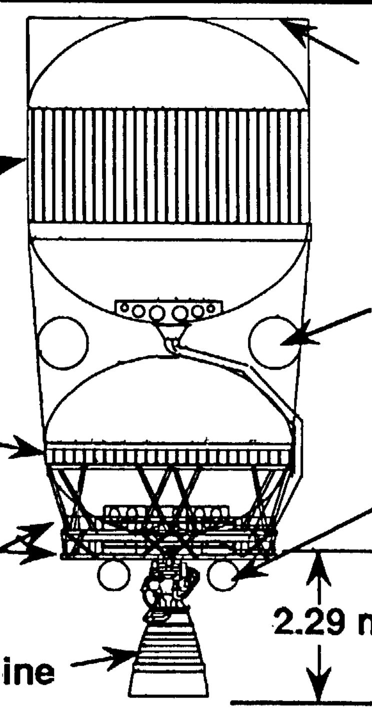
- The Only Two Candidates For a New Upper Stage are The LOX/LH2 & LOX/RP Type Stages. Both are Viable Options With the LOX/LH2 Type Stages Having a Larger Potential For Growth Based on Performance

- This Small Upper Stage Has a Natural Outgrowth into The Other Launch Vehicles (i.e. Delta, Atlas, Titan, STS, etc...)

Medium Upper Stage Configuration

MSFC

Payload (Titan IV (U))	9.0 t
Payload Position	TLI
Structure	1,956
Thermal Manag.	54
Engine & Hardware	297
Pressurant Sys.	121
RCS System	33
Avionics	
GN&C / Miss. Man.	133
Comm.	15
Power	136
RSS	46
Contingency (20%)	558
Dry Mass (kg)	3,349
Propellant (LOX/LH2)	23,024
RCS Prop. (N2H4)	136
STAGE GLOW (kg)	26,509
Eff. Mass Fract.	0.860
Engine Type/#	RL10A-4/1
Vac. Thrust	20.8 klbf
Vac. ISP	449 sec
Exit Diameter	1.17 m



Stage Length = 10.3 m

Material:

- Structure: AL2219
- Avionics Deck: AL HoneyComb

Notes:

Payload (t)	LEO	GSO	TLI
NLS 2	36.6	8.4	13.0
NLS 3 (2x RL10A-4)	9.9	1.0	2.5

Additional ELV Analysis Will Yield Subsequent Configuration Changes
Config. Shown Sized For Titan IV SRMU Launch Vehicle

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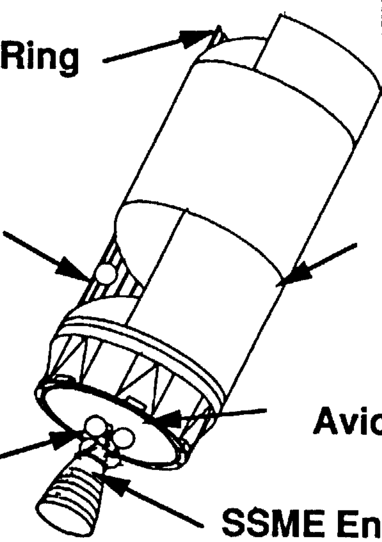
RS930303-02C

NLS HLLV TLI Upper Stage Configuration

MSFC

Payload	103 t
Payload Position	TLI
Structure	16,636
Thermal Manag.	211
Engine & Hardware	4,705
Pressurant Sys.	181
RCS System	94
Avionics	
GN&C / Miss. Man.	134
Comm.	14
Power	146
RSS	46
Contingency (20%)	4,433
Dry Mass (kg)	26,600
Propellant (LOX/LH2)	306,331
RCS Prop. (N2H4)	1,257
STAGE GLOW (kg)	334,188
Eff. Mass Fract.	0.901
Engine Type/#	SSME/1
Vac. Thrust	470 klbf
Vac. ISP	453 sec
Exit Diameter	2.29 m



Stage Length = 24.7 m

Material:	
Structure	AL2219
Avionics Deck	AL HoneyComb
Notes:	
Max G = 4.0	Max Q = 900 psf
Sub Orbital Burn	1x
Engine Out	None

See NLS Derived Heavy Lift Launch Vehicle Data Sheet for Booster Data

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Technology Integration

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MARTIN MARIETTA

C-8

Matrix Indicates Technology's Applicability to Each Space Transportation Option

	Option 1				Option 2						Option 3			
Advanced Propulsion	STS	ELV	SL	ELVU	STS	50K	EXP	ELV	SL	ELVU	STS	ELV	SSTO	EXP
Robust Main Engine:														
35-70K lbs Thrust Cryogenic		X	X	X		X	X	X	X	X		X	X	X
Integrated Modular Engine		X	X	X		X	X							
Enhanced Throttling Range														
SSME Restart Capability														
Adv. Pressurization Techniques		X	X	X										
Improved EMA Valves	X	X	X	X										
Laser Initiation	X	X	X	X										
Reaction Control:														
RCS - GH2/GO2		X	X											
Advanced Tank Material		X	X											
Thrust Vector Control:														
Electro-Mechanical Actuation	X	X	X											
Electro-Hydrostatic Valves														
Data Handling & Control:														
Standard Access Interfaces	X	X												
Failure ID Algorithms		X												
Sensors:														
Plume Spectroscopy		X												
Holographic/IR Leak Detection	X	X												
Propellant Management:														
Advanced Thermal Insulation	X													
Long Term Cryo Storage														
Adv. Fluid Transfer & Instr.	X													

Matrices Developed to Document Applicability of Technology to Upper Stages for Each of the Space Transportation Program Options. A Future Analysis will Determine Which Technologies are Critical or Provide Cost Effective Enhancements

Assumptions:

- No New Upper Stages Developed for STS Missions - Only Upgrades Considered for Existing Vehicles (i.e. TOS, IUS, etc.)
- 35-70K Engine Thrust Upper Stage Assumed for ELV, SpaceLifter, and ELV Upgrade Options
- TLI-Class Upper Stage Assumed for Exploration Vehicle
- Small High-Performance Upper Stage Assumed for SSTO/TSTO Vehicles

STS - Shuttle
ELV - Existing Expendable Launch Vehicle
SL - Spacelifter
SSTO - Single Stage to Orbit
ELVU - Upgrades to Existing ELVs
50K - 50K Vehicle/CTV/PLS
EXP - Large Vehicle for Exploration (TLI Class)

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ORIGINAL PAGE IS OF POOR QUALITY

Advanced Development Technologies

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Discipline/Areas	Technology	Benefits
• Propulsion	50K lbs Thrust Cryogenic Integrated Modular Engine Enhanced Throttling Range SSME Restart Capability Advanced Pressurization Tech.	Larger Payloads Scalability/Reliability Performance Flexibility Improved Reusability Eliminates Large Inflight GH2 Requirement
• Avionics	Open Architecture Fault Tolerance/Redundancy Mgmt. Integrated Test Capability Standardized Interfaces	Low Cost Integration & Dev. High Reliability/Safety Low Cost Integration & Dev. Low Cost Integration & Dev.
• IVHM	Automated Checkout & Test Onboard Processing Distributed Fault Detection/Isolation Direct Access Test Interface	Low Cost Integration Low Cost Integration Increased Visibility Low Cost Integration
• Structures/Materials	Metal Matrix Composites Composite Isogrids Embedded Sensors	Environmental Tolerance Improved Strength to Weight Improved Reliability/Safety
• Operations	Laser Ordnance Processing Automated Checkout & Test Launch decision Support	Low Cost/Increased Safety Low Cost Operations Low Cost Operations

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Technology Development Infrastructure

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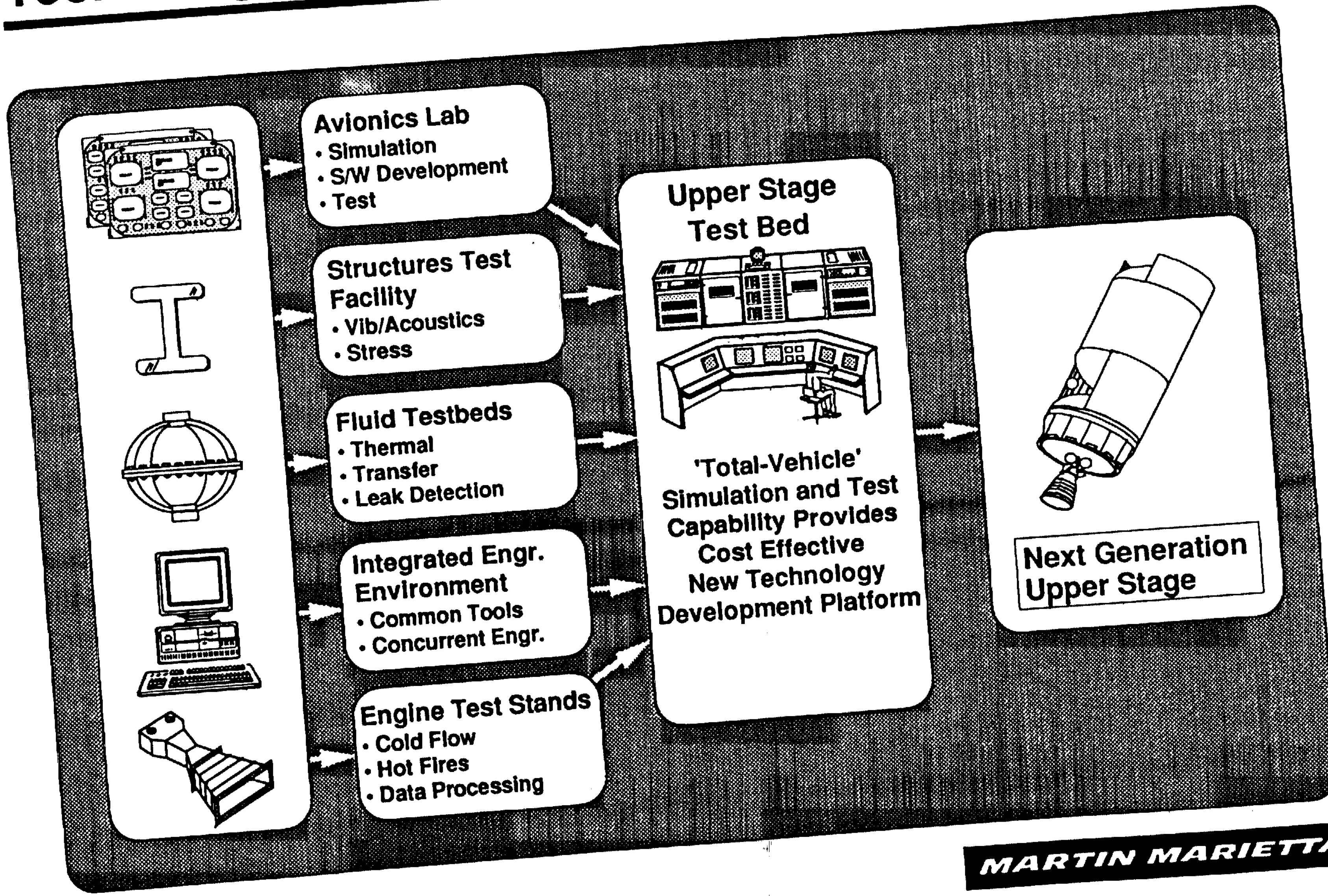

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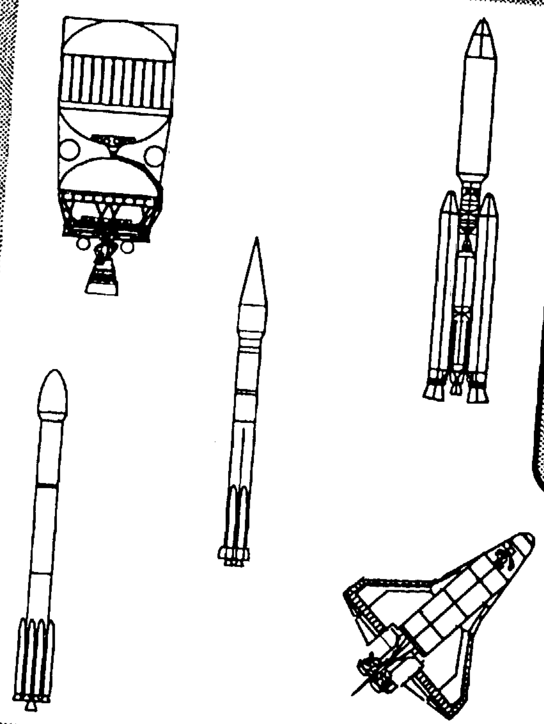

Significant Business Factors

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Sidney M. Earley
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Upper Stage Planning Triad

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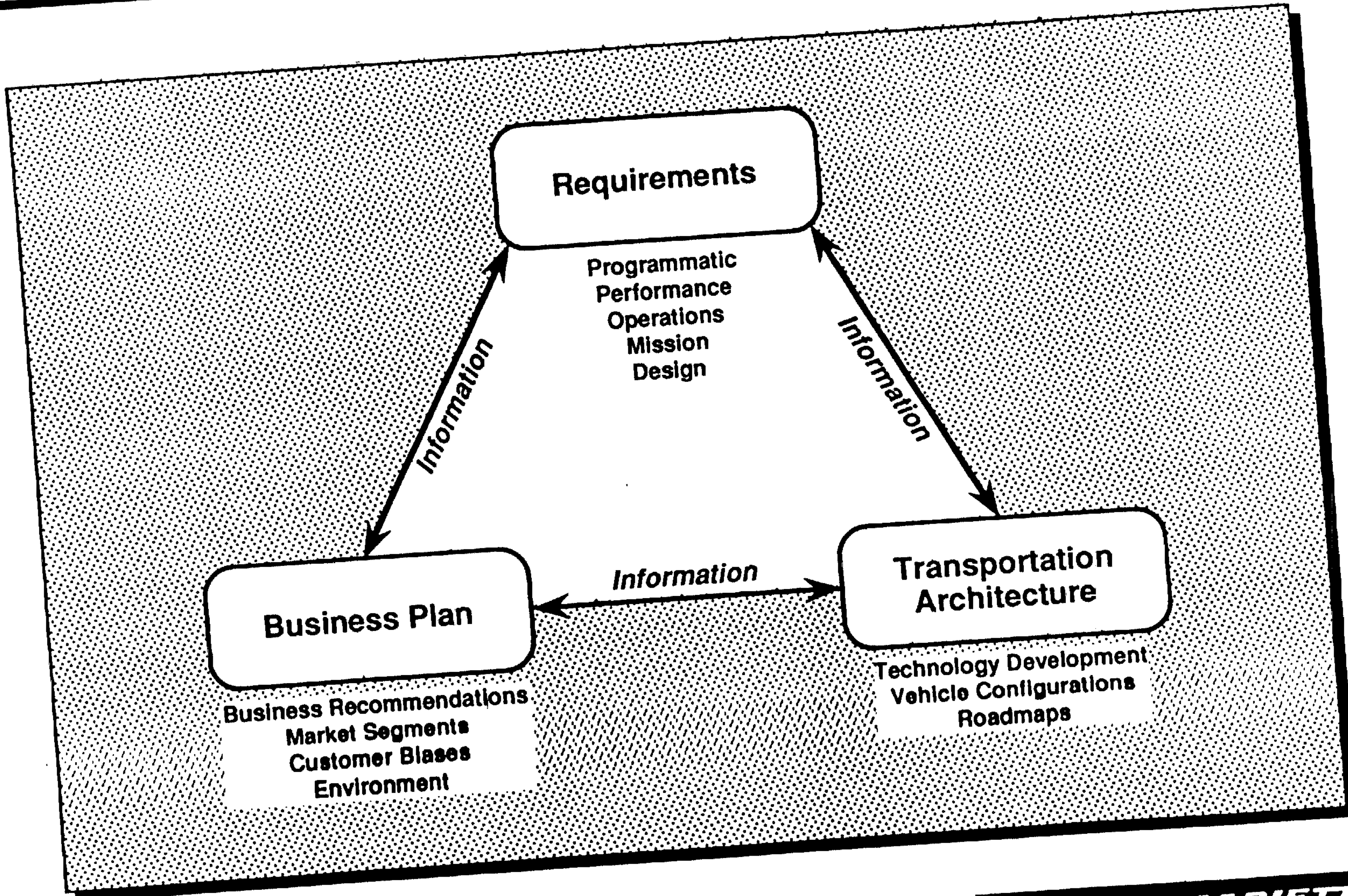

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SE930430-01A

Status of Existing Domestic Upper Stages

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Upper Stage	Payload (t)	Cost ($M)	Comments
Centaur Centaur G'	2.7 - 3.5 to GTO 4.5 - 5.0 to GSO	30 60	• Only Operational Cryogenic Upper Stage in the U.S. • Relatively High Operational Complexity • Compatible with Atlas and Titan IV (G') • Failures of Common Subsystems Cause Standdown of both Launch Systems • Manufactured by General Dynamics
IUS	2.4 to GSO	70	• Two Stage (PKM/AKM) All Solid System • High Cost-to-Payload Mass Ratio • Compatible with Shuttle, Titan III & IV • Production Line Will Soon Be Shutdown • Manufactured by Boeing
TOS	5.0 to GTO	30	• Single Stage All Solid System • Relatively New, Only One Flight to Date • Compatible with Shuttle, Titan III & IV • Manufactured by Martin Marietta
PAM-D	1.8 to GTO	15	• Single Stage All Solid System • Vast Flight History, Very Reliable • Compatible with Delta and Shuttle • Manufactured by McDonnell Douglas

Upper Stage Customers & Their Interests

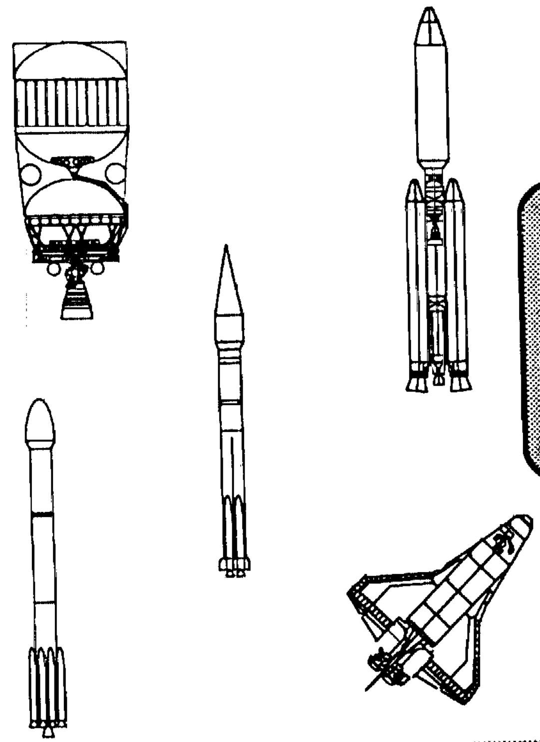
Customer	Interests	
NASA	• Support Missions • Lower Cost & Risk • Technology Development • International Cooperation	• Leverage National Assets • Unify Support from the Field Centers• Safety • Human Rated Vehicle
Air Force	• Reliable Access to Space • Support Missions • Lower Cost & Risk	• Responsiveness • Maintain Capabilities
Commercial Users	• Reliable Access to Space • Improved Availability • Lower Cost & Risk	
Congress	• Cut Spending (≈ Lower Cost) • Increase Employment • Improve Competitive Position • Constituents' Interests	• Leverage National Assets • Maintaining Budget Constraints
Administration	• Cut Spending (≈ Lower Cost) • Increase Employment • Improve Competitive Position	• Increase Efficiency • Technology Transfer

Development of a New Upper Stage Requires a Thorough Understanding of the Key Interests of the Customer(s)

Reasons for a New Upper Stage

- Increased U.S. Competitiveness
- Technology Advancement
- Lower Life Cycle Costs
- Improved Performance Capabilities (i.e., Payload, Reliablity, Operability)
- Synergistically Unite Different Government Agencies
- Employment to the Aerospace Community
- Help to Bring About a Change in the Way Industry Operates (e.g., The Use of an Innovative Development Schedule)

These Points Need to Be Publicized to Sell the Program



Future Vision

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Dan O'Neil
Sidney M. Earley
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Upper Stage Study Issues

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- NASA Must Define a Long Term Transportation Plan to Provide a Framework for Future Studies
- Upper Stage Concepts Should Drive Requirements for the Launch Vehicle
- Studies Should Answer Specific Questions and Close Issues
- Study Products Should Provide a Market Context for Upper Stage Concepts
- NASA Should Define an End-To-End Process for Developing and Selling Upper Stage Programs
- Studies Should Produce Program, Performance, and Market Data Identified by the End-To-End Program Development Process
- NASA's Project Life Cycle Should Emphasize the Use of Study Products

Recommended Study Process

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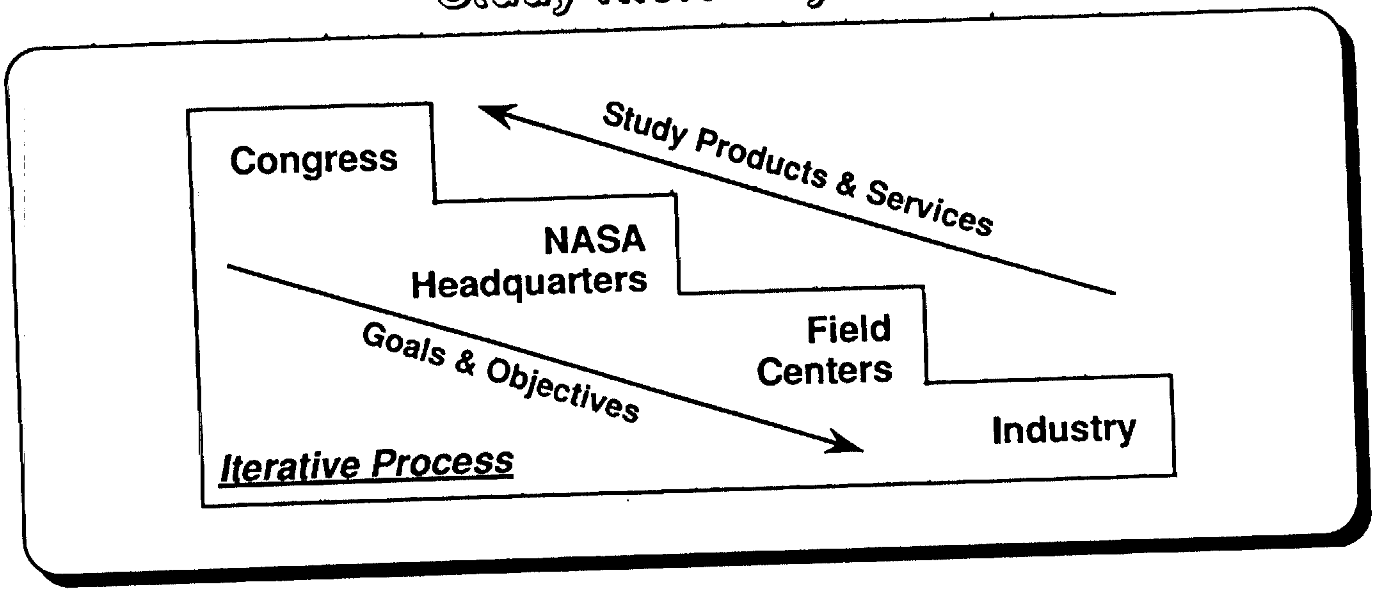
Study Hierarchy



- Specify Study Products & Services to Support the Decision Making Process
  - Answer Specific Questions Relating to NASA's Strategic Plan
  - Develop Requirements & Provide Technical Data
  - Perform Analysis on the Scope of the Long Range Plan

- Studies Should Determine Feasibility
  - Provide a Decision Point
  - Influence Strategic Direction

Factoring Making A Successful Program

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- Each Play Must Commit Resources - Government Should Team with Industry
- Each of the Team Members' Benefits from the Program Must Be Proportional to the Amount of Resources Contributed (much like the ESA)
- An Individual Organization Must Be Identified as a Team Lead and Held Accountable
- The Upper Stage Team Could Attempt to Secure Multi-Year Funding
- The Program Must Have Identifiable Products Throughout the Life Cycle to Demonstrate that Progress Is Being Made
- A Constituency Must Be Built in Congress and the Administration that Will Promote the Idea of Developing a New Upper Stage
- Support Must Be Sought from Multiple Contractors to Reinforce and Add Credibility to the Assertion that a New Upper Stage Is Needed

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Recommended Direction

- Conduct Trade Studies for Upper Stage Concepts Based on a 50K Thrust Conceptual Engine and Existing Upper Stage Engines
- Define a System Development Program Based on Requirements Derived from the Winning Concept
- Establish an Engineering Environment that Captures Corporate Knowledge from All Phases of an Upper Stage Project Life Cycle
- Define an End-To-End Program Development Process and Use the Next Generation Upper Stage as a Path Finder

Upper Stage
Technical Requirements Document

Upper Stage Technical Requirements Document

Note: Changes from previous version are underlined.
Comments and/or references are contained in italics following the requirement.

1.1 Mission

a. The upper stage(s) will support a wide range of missions including those defined in the table below. *(Initial missions derived from analysis of HQ Code D Mission Model and the 1991 DoD National Mission Model. Backup data for the mission classes will be provided in Appendix A.)*

Missions	Operational Apogee (nmi)	Delivered Mass (lbs)
Small LEO Payloads	≤500	15,000-24,999
Medium LEO Payloads	≤500	25,000-39,999
Large LEO Payloads	≤500	≥40,000
Small-High Energy Payloads	≥5,000	4,000-7,999
Med-High Energy Paylaods	≥5,000	8,000-15,000
Large-High Energy Payloads	≥5,000	≥60,000

1.2 System Life

a. The system will be operational over a lifetime of at least 20 (TBR) years.

1.3 Mission Life

a. The minimum mission life (from first ignition through disposal) of the upper stage shall be TBD.

1.4 IOC

a. The system shall have an initial operating capability in 2003 (TBR). *(2003-Approximate date in current "Access to Space" Option 2, 2005-Input from J. Green)*

1.5 Launch

a. The system shall have the capability to be launched from both ETR and WTR.

b. The minimum nominal launch rate shall be 4 (TBR) per year with growth to accommodate up to a maximum of TBD flights per year by TBD. *(Derived from analysis of CNDB 1991. Four flights/year also appears compatible with 50K & 80K spacelifters described in Access to Space architecture analysis.)*

1.6 Reliability

a. The system shall deploy payloads to intended orbits with 0.98 probability of success.*(FLO system flight success = .96 FLO PRD Vol 1 #882 ; ALS HLLV & Upper stage =0.98 AFSPACECOM SORD 4.1.1.2.A, Current value used in architecture analysis at MSFC)*

b. Hardware shall be designed such that the effects of single-point failures shall not cause loss of mission. *(USRS SRD 6.6.1)*

1.7 Facilities

a. Operations and Processing facilities shall be coordinated/designed in parallel with upper stage to achieve more efficient, reliable operations involving fewer people and shorter launch schedules *(Derived from recommendations in Earth to Orbit* and the *10 Year Tech Plan):*

1.8 Environments

a. The upper stage shall be designed to operate in and survive the environments described in RECON 89N22638 "Orbital Debris Environment for Spacecraft Design to Operate in Low Earth Orbit - NASA TM 100471, Sept. 1, 1988", NASA-SP-8030 "Meteoroid Environment Model, 1970 - Interplanetary and Planetary. NASA Space Vehicle Design Criteria Environment. Oct., 1970", and EXPO-T2-920021-EXPO, "Lunar Engineering Models: General and Site-Specific Data". *(FLO PRD Vol 1 #813, #814, #815)*

b. The upper stage must be designed to withstand the launch system acceleration of 4-6 g (TBR). *(Values accepted in recent NLS studies).*

c. Maximum acceleration of the upper stage shall not exceed 4-6 g (TBR).

1.8+ Environmental Impact

a. New facility development will be constrained by environmental limitations. Site selection must consider flora, fauna, cultural, and historic sites.

b. Upper stage toxic emissions and other hazardous effects must be minimized and precluded if possible.

1.9 Safety

a. The upper stage program will include a system safety and personnel safety program which has been developed in compliance with mission, launch and processing site specific requirements (e.g., ESMCR 127-1 for the ETR launch site, and KSC 1098 for KSC processing).

1.10 Disposal

a. After separation from the payload, the upper stage shall provide a controlled disposal into a disposal orbit, a broad ocean area (BOA), or deep space (TBR). *(FLO PRD Vol 3 #1616)*

1.11 Piloted Flights

a. The system shall have the capability to support piloted flights in 2003. *(Date consistent with IOC).*

1.12 GN&C

a. The upper stage shall provide the following accuracies:

Mission	Apogee altitude (nmi)	Perigee Altitude (nmi)	Inclination (°)
			0.1
Small LEO Payloads	5	5	0.1
Medium LEO Payloads	5	5	0.1
Large LEO Payloads	5	5	0.1-0.2
Small-High Energy Payloads	100-115	100-115	0.1-0.2
Medium-High Energy Payloads	100-115	100-115	0.1-0.2
Large-High Energy Payloads	100-115	100-115	

1.13 Communication

a. The system must provide for communication with the range, the pad, the LCC, the relay network (if used), and the tracking network.

1.14 Operability

a. The upper stage shall implement the integrate-encapsulate-launch ground operational process. *(Recent STV studies have demonstrated benefits associated with process. Also, referenced in From Earth to Orbit as effective method to provide robust, reliable, low cost launch infrastructure.)*

b. The system shall be designed with simple, standard payload interfaces. Payload unique requirements must be addressed by use of adapter systems and self-contained servicing support. *(AFSPACECOM SORD 4.1.1.1.C.3)*

c. The upper stage shall allow for payload substitution (within a given payload class) up to 5 days prior to launch *(AFSPACECOM SORD 4.1.1.1.C.2)]*

d. The upper stage will be compatible with TBD launch systems.

1.15 Maintainability

a. The system shall detect and isolate 90 - 95% of failures to a specific component within established time constraints using internal automatic or semiautomatic health monitoring, external support equipment, technical orders, and training. *(AFSPACECOM SORD 4.1.1.3.A.)*

b. Routine maintenance shall not be performed on the pad (unless shown to be operationally beneficial). *(AFSPACECOM SORD 4.1.2.A)*

c. Failed Line Replaceable Units (LRUs) shall be removed and replaced, packaged, and shipped back to the vendor or supplier for repair or replacement. *(AFSPACECOM SORD 4.1.2.A)*

d. Maintenance personnel shall work in a "paper-less" environment using automated, user-friendly systems to reduce the workload and simplify procedures. *(AFSPACECOM SORD 4.1.2.A)*

1.16 Transportation

a. Vehicle components and propellants will meet all federal, state, and local transportation requirements. This includes safety, size, weight, and security. Transportation will be accomplished by the most practical and economical means. *(AFSPACECOM SORD 4.1.2.4.B)*

b. Conventional, non-specialized commercial transports shall be used to deliver finished materials from the manufacturer to the site, whenever possible. Military or Government vehicles should be used whenever practical for transportation of vehicle components between on-site facilities. Military airlift may be used for component transport between sites, where economical. Transportation of components will not require overly complex loading, housing, or transportation equipment. (Note: Reference to Military Vehicles applicable pending incorporation of DoD missions.) *(AFSPACECOM SORD 4.1.2.4.B)*

1.17 Security

a. The system must be capable of providing security appropriate for payload classification (up to and including Top Secret(TS)/Sensitive Compartmental Information (SCI). This includes operations security, communication security, and information security. (Note: Security requirement applicable pending incorporation of DoD missions.)

1.18 Availability

a. The system shall sustain system availability of 0.90 over the life cycle. Availability is a measure of the degree to which an item is in an operable and commitable state a the start of a mission when the mission is called for at an unknown time. *(AFSPACECOM SORD 4.1.1.4.A)*

b. Stand-down time of longer than 3 months shall have a probability of less than 0.05. *(AFSPACECOM SORD 4.1.1.4.D)*

1.19 Dependability

a. System dependability must be at least 0.95. Dependability is the ability to maintain flight schedule. It is the pre-ascent reliability of the overall system. This includes external factors such as weather and internal factors such as production, assembly, and payload integration anomalies. *(AFSPACECOM SORD 4.1.1.4.1.a)*

1.20 Proximity Operations

a. The system must be capable of supporting proximity operations.

1.21 Commonality

a. Commonality among hardware, software, and operations must be emphasized in the event that a family of concepts is needed to fulfill the mission requirements.

1.22 Technology

a. Technology advances should be pursued as required to ensure a balance among operability, affordability, performance, supportability, producibility, and schedule. Such advances shall contribute to and/or be compatible with other requirements in this document.

Reference Documents

Civil Needs Data Base FY 91 Version, NASA Headquarters, March 1992.

Air Force Space Command System Operational Requirements Document for Military Advanced Launch Systems, Department of the Air Force Headquarters, AFSPACECOM/XRSD, 14 August 1990.

Upper State Responsiveness Study/Titan Upper Stage Systems Requirements Document, USRS RFP, AFSD, December 1988.

First Lunar Outpost Program Requirements Document, Volumes 1 and 3, Johnson Space Center, 26 January 1993.

10 Year Space Launch Technology Plan, Federal Agencies (DoD, DoE, NASA) and several Industry members, November 1992.

From Earth to Orbit - An Assessment of Transportation Options, National Research Council, 1992.

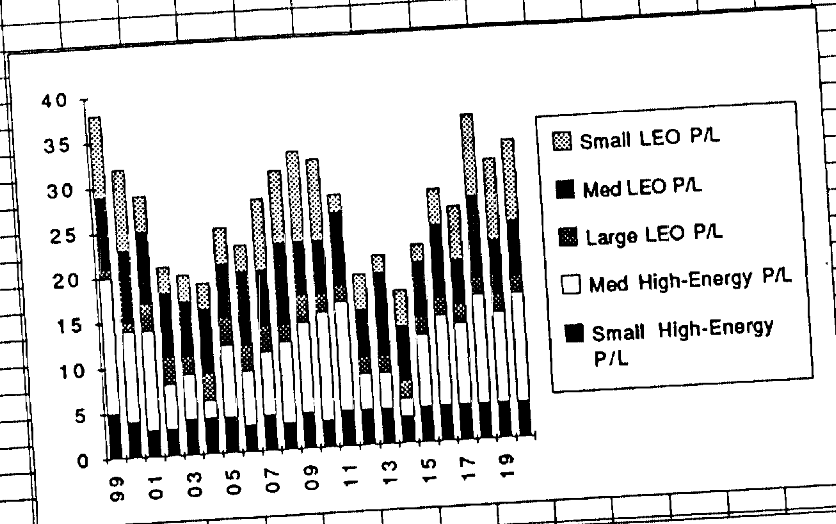
HQ Code D Mission Model

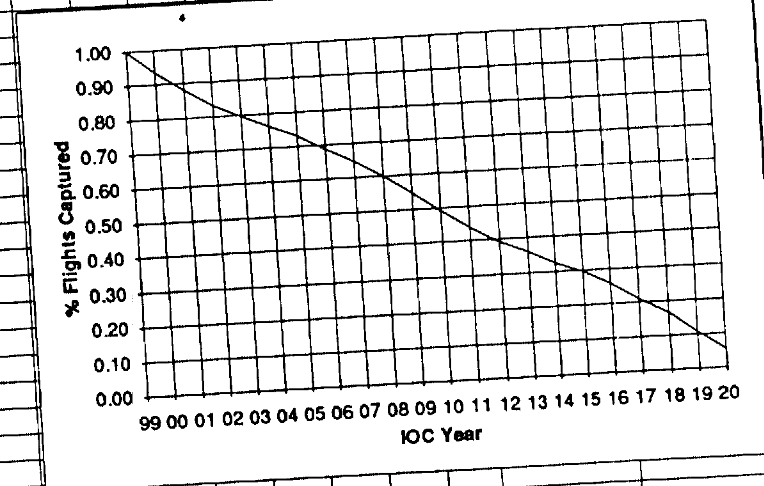
DoD National Mission Model 1991

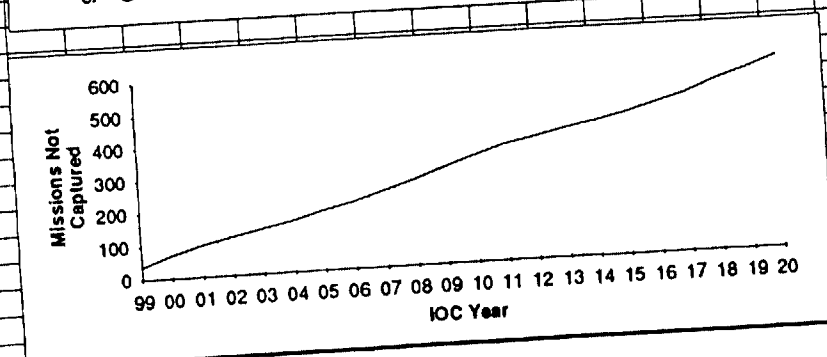
TD16-001
Upper Stage Technical Requirements Document
Source Analysis for Paragraph:
1.1a Mission

Potential Missions by Year

Year	99	00	01	02	03	04	05	06	07	08	09	10	11	12	13	14	15	16	17	18	19	20	
Sm High-Energy	5	4	3	3	4	4	4	3	4	3	4	3	4	4	4	3	4	4	4	4	4	4	
Med High-Energy	15	10	11	5	5	2	8	6	7	9	10	12	12	4	4	2	8	10	9	12	10	12	
Large LEO	1	1	3	3	2	3	3	3	3	2	3	2	2	2	2	2	2	2	2	2	2	2	
Med LEO	8	8	8	7	6	7	6	8	6	9	6	6	8	5	9	6	6	8	5	9	6	6	
Sm LEO	9	9	4	3	3	3	4	3	8	8	10	9	2	4	2	4	2	4	6	9	9	9	
Total Flights	38	32	29	21	20	19	25	23	28	31	33	32	28	19	21	17	22	28	26	36	31	33	592
% Captured	1.00	0.94	0.88	0.83	0.80	0.76	0.73	0.69	0.65	0.60	0.55	0.49	0.44	0.39	0.36	0.33	0.30	0.26	0.21	0.17	0.11	0.06	
Number Missions Nc	38	70	99	120	140	159	184	207	235	266	299	331	359	378	399	416	438	466	492	528	559	592	







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Potential Upper Stage Missions

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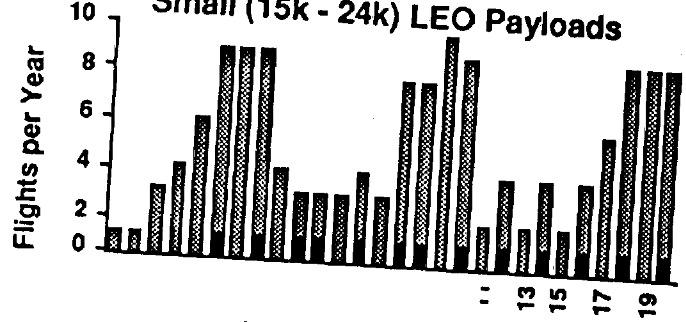
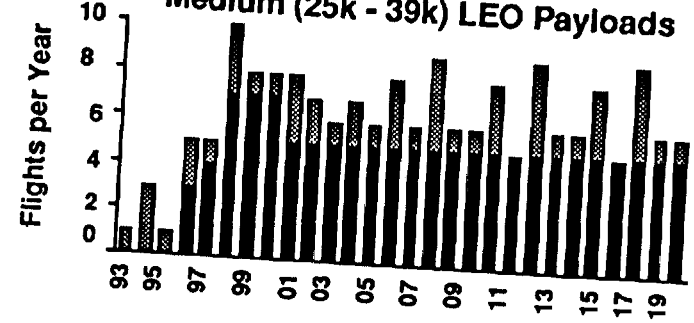
Small (15k - 24k) LEO Payloads

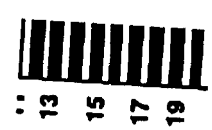


Medium (25k - 39k) LEO Payloads

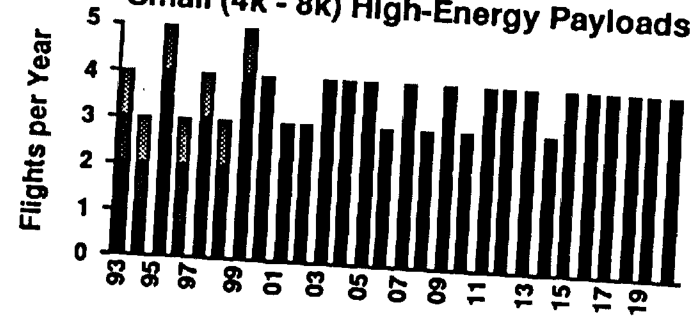


TD 1C Folder

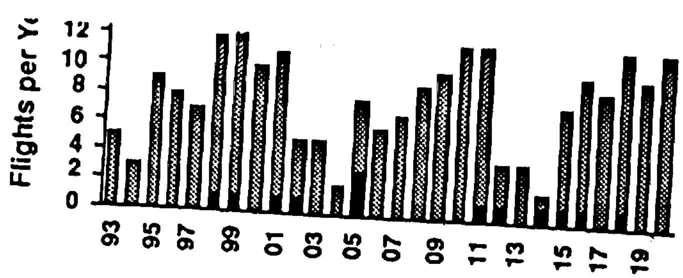
yloads



Small (4k - 8k) High-Energy Payloads



nergy Payloads



Legend

- NASA Missions (HQ Code D Mission Model)
- DoD Missions (1991 National Mission Model)
- Commercial Missions (HQ Code D Mission Model)

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LR930514-Mission Reqt Analysis

Small LEO Payloads - Class 5

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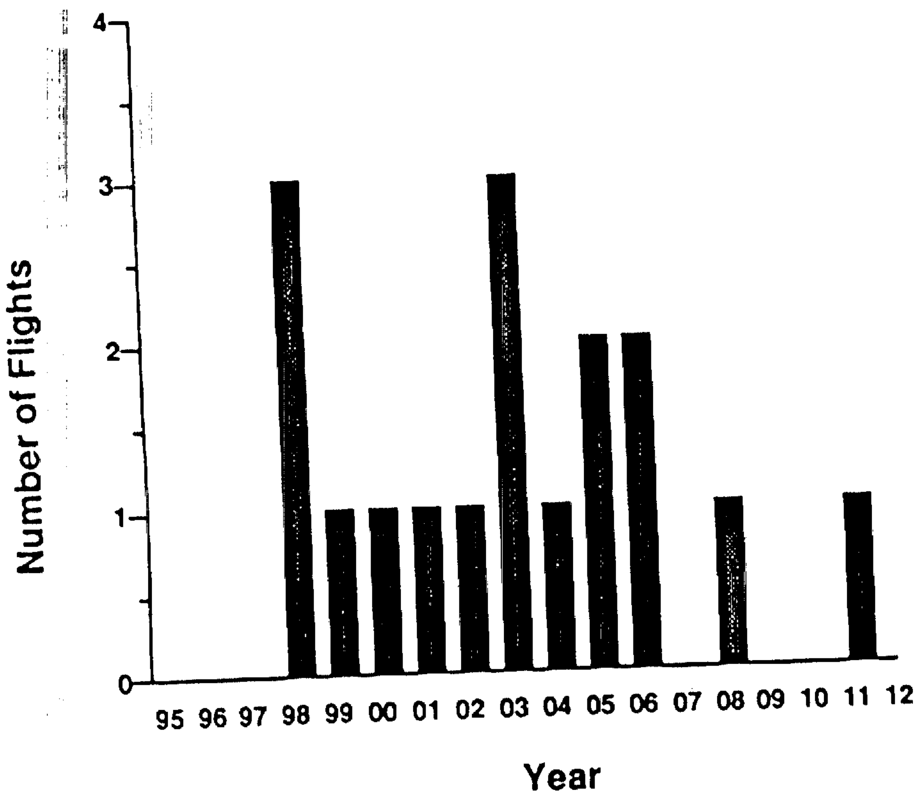

<u>Payload Names</u>/(Number of Flts)
CELSS SS Mod Project (1)
EOS Polar Orbiting Platforms (6)
Industrial Space Facility - Aux mod (1)
Hard X-Ray Imaging Facility (1)
Base Mission to SSF (2)
Growth Missions to SSF (5)
Variable Gravity Large Centrifuge Facility (1)

- Class 5
  - Op Apogee ≤ 500 nmi
  - Deliver Mass ≥ 20,000-29,999 lbs
- 17 Missions Possible Through Year 2021

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Large LEO Payloads - Class 3

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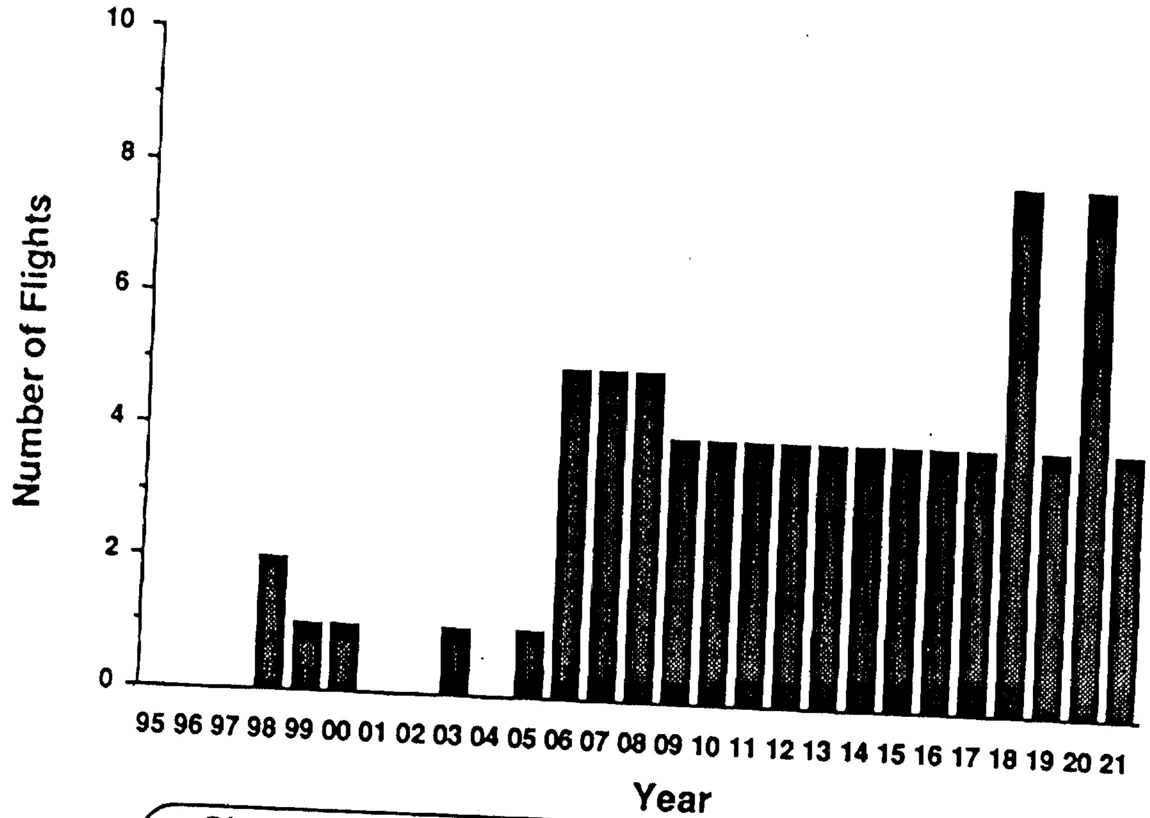

<u>Payload Names/(Number of Flts)</u>
Large Deployable Reflector (1)
Lunar - Piloted & Cargo (65)
Mars - Piloted & Cargo (8)
Base Mission to SSF (3)
Growth Missions to SSF (4)

- Class 3
  - Op Apogee $\leq$ 500 nmi
  - Deliver Mass $\geq$ 40,000 lbs
- 81Missions Possible Through Year 2021

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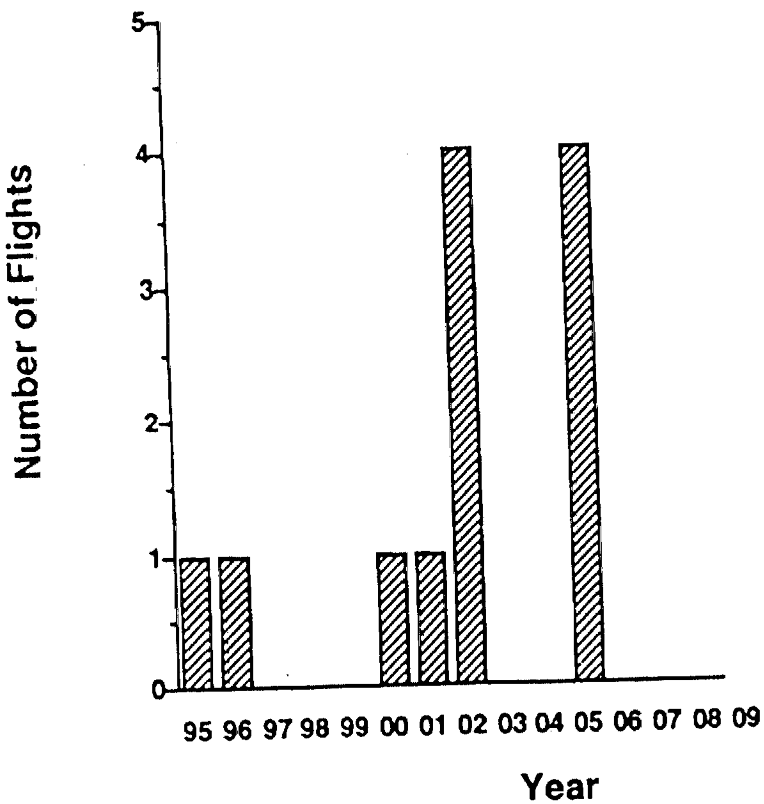
Large High Energy Payloads - Class 1

MSFC



Payload Names/(Number of Flts)
Cassini (1)
Submillimeter Mission (1)
Space Infrared Telescope Facility (1)
GEO Platform (1)
Comment Rendezvous/Asteroid Flyby (1)
Mars Rover/Sample Return (2)
Comet Nucleus Sample Return (1)
Pluto Flyby (1)
Neptune Orbiter/Probe (1)
Uranus Orbiter/Probe (1)
Mercury Orbiter (1)

- Class 1
  - Op Apogee ≥5,000 nmi
  - Delivered Mass ≥6,000 lbs
- 12 Missions Possible Through Year 2021

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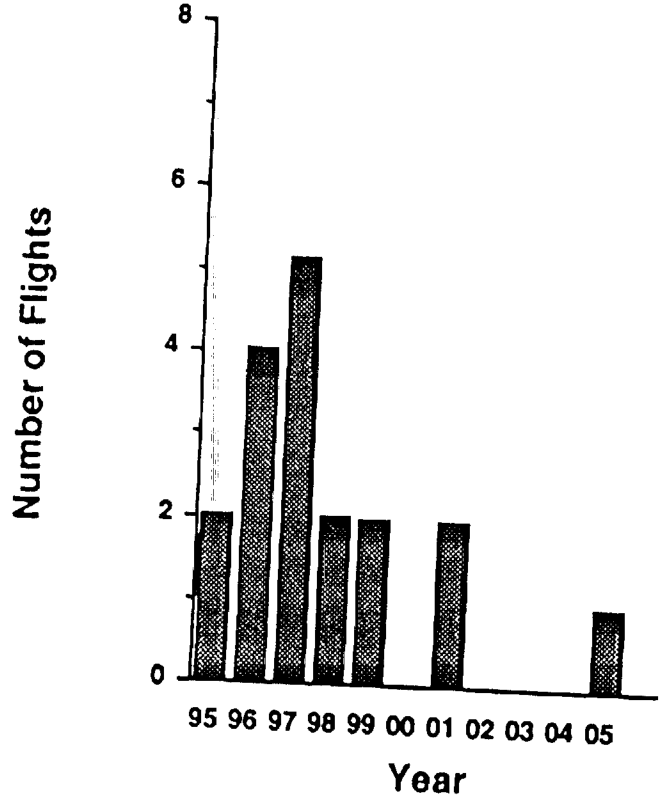
N/A

Medium LEO Payloads - Class 4

MSFC



Payload Names/(Number of Flts)
Advanced X-Ray Astrophysics Facility (1)
Animal/Paint Vivarium (1)
Industrial Space Facility - Mod 1 (2)
Base Mission to SSF (12)
Growth Missions to SSF (2)

- Class 4
  - Op Apogee ≤ 500 nmi
  - Deliver Mass ≥ 30,000-39,999 lbs
- 18 Missions Possible Through Year 2021

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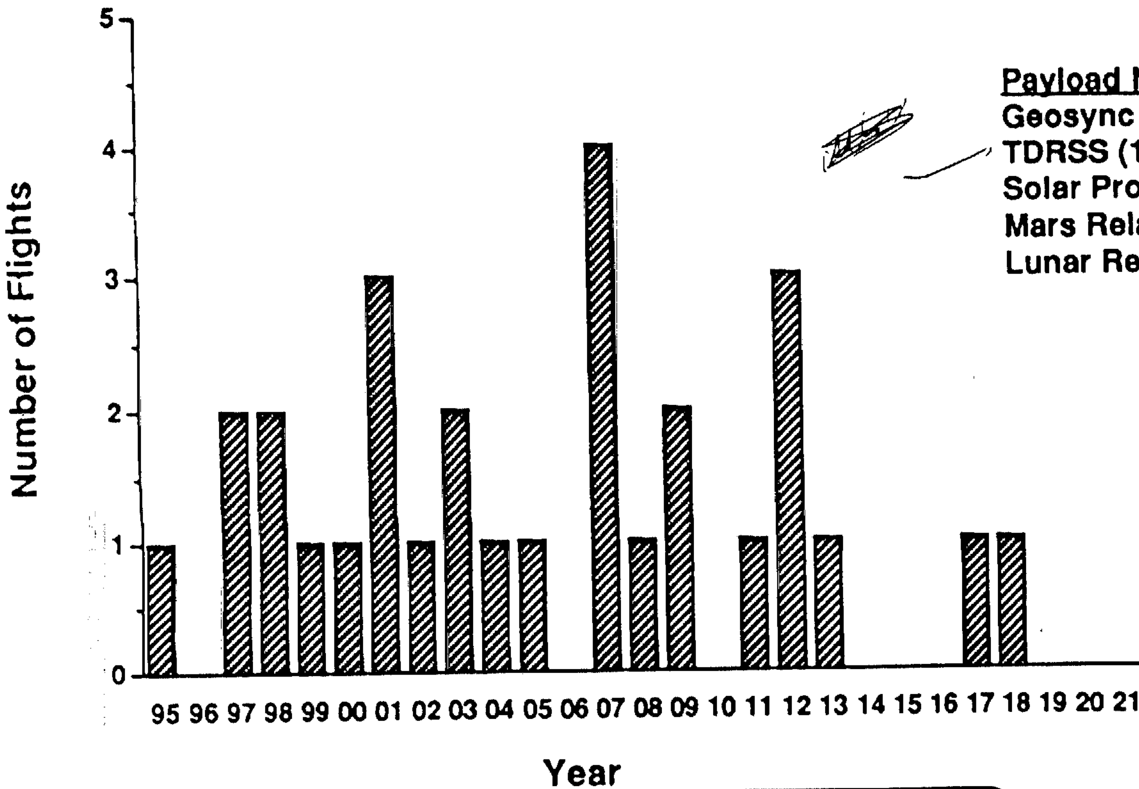
LR930216-04

Medium High Energy Payloads - Class 2

MSFC



Payload Names/(Number of Flts)
Geosync Operational Environmental Satellite (10)
TDRSS (11)
Solar Probe (1)
Mars Relay - Phase I,II,III (6)
Lunar Relay (1)

- Class 2
  - Op Apogee $\geq$ 5,000 nmi
  - Delivered Mass 2,000 - 6,000 lbs
- 29 Missions Possible Through Year 2021

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LR930216-01

NAME	FY	Launch			MASS TO LEO	PLD MASS	LENGTH	DIAMETER	DEST	APO	PER	INC	DELTA V
ACTS	1993	Jun-93 STS/TOS	STS	Planned	30580	6026	29.5	14.1	GEO	19310	19310	0.0	38267
NOAA-I	1993	Jun-93 Atlas (VAFB)	MC	Planned	2883	2266	14	6	LEO SYN	460	460	98.7	26702
Other Shuttle P/L	1993		STS	Potential	40000	40000			LEO	160	160	28.5	24579
Other Shuttle P/L	1993		STS	Potential	40000	40000			LEO	160	160	28.5	24579
Other Shuttle P/L	1993		STS	Potential	40000	40000			LEO	160	160	28.5	24579
Other Shuttle P/L	1993		STS	Potential	40000	40000			LEO	160	160	28.5	24579
TDRS-F	1993	Jan-93 STS/IUS	STS	Planned	27580	4905	19	9.8	GEO	19310	19310	0.0	38267
GOES-I	1994	Apr-94 Atlas I	IC	Planned	16470	2160	9	9	GEO	19310	19310	0.0	38267
NOAA-J	1994	May-94 Atlas (VAFB)	MC	Planned	2883	2255	14	6	LEO SYN	460	460	98.7	26702
Other Shuttle P/L	1994		STS	Potential	40000	40000			LEO	160	160	28.5	24579
Other Shuttle P/L	1994		STS	Potential	40000	40000			LEO	160	160	28.5	24579
Other Shuttle P/L	1994		STS	Potential	40000	40000			LEO	160	160	28.5	24579
Other Shuttle P/L	1994		STS	Potential	40000	40000			LEO	160	160	28.5	24579
Other Shuttle P/L	1994		STS	Potential	40000	40000			LEO	160	160	28.5	24579
Other Shuttle P/L	1994		STS	Potential	40000	40000			LEO	160	160	28.5	24579
Other Shuttle P/L	1994		STS	Potential	40000	40000			LEO	160	160	28.5	24579
POLAR	1994	May-94 Delta	MC	Planned	10119	2645	7	9	EAR OTH	30996	6199	90.0	36254
WIND	1994	Dec-93 Delta	MC	Planned	9956	2645	6.6	9	LUN ORB	826560	41328	0.0	34768
GOES-J	1995	Apr-95 Atlas I	IC	Planned	16470	2160	9	9	GEO	19310	19310	0.0	38267
Other Shuttle P/L	1995		STS	Potential	40000	40000			LEO	160	160	28.5	24579
Other Shuttle P/L	1995		STS	Potential	40000	40000			LEO	160	160	28.5	24579
Other Shuttle P/L	1995		STS	Potential	40000	40000			LEO	160	160	28.5	24579
Other Shuttle P/L	1995		STS	Potential	40000	40000			LEO	160	160	28.5	24579
Other Shuttle P/L	1995		STS	Potential	40000	40000			LEO	160	160	28.5	24579
RADARSAT	1995	Dec-94 Delta	MC	Planned	8167	7040	14	9	LEO SYN	411	411	98.5	26548
SOHO	1995	Jul-95 Atlas IIAS	IC	Planned	17085	4078	11.8	12	DS SOL	-	-	-	34768
TDRS-G	1995	Jul-95 STS/IUS	STS	Planned	27580	4905	19	9.8	GEO	19310	19310	0.0	38267
LAGEOS-III	1996	TBD Delta	MC	Planned	2424	893	2	2	LEO OTH	3240	3240	52.0	30940
NOAA-K	1996	Jan-96 Titan II	MC	Planned	2883	2255	14	6	LEO SYN	460	460	98.7	26702
Other Shuttle P/L	1996		STS	Potential	40000	40000			LEO	160	160	28.5	24579
Other Shuttle P/L	1996		STS	Potential	40000	40000			LEO	160	160	28.5	24579
Other Shuttle P/L	1996		STS	Potential	40000	40000			LEO	160	160	28.5	24579
Other Shuttle P/L	1996		STS	Potential	40000	40000			LEO	160	160	28.5	24579
Other Shuttle P/L	1996		STS	Potential	40000	40000			LEO	160	160	28.5	24579
SCOUT-1	1996	Mar-96 Delta	MC	Planned	4585	1200	23	10	LUN ORB	-	-	-	34768
SSF Assembly	1996		STS	Potential	36300	36300			LEO SSF	220	220	28.5	24579
SSF Assembly	1996		STS	Potential	36300	36300			LEO SSF	220	220	28.5	24579
SSF Assembly	1996		STS	Potential	36300	36300			LEO SSF	220	220	28.5	24579
XTE	1995	Aug-95 Delta	MC	Planned	7209	6505	19	6	LEO OTH	270	270	28.5	24748
ACE	1997	Aug-97 Delta	MC	Planned	7123	1433	8.2	6.6	EAR OTH	826543	826543	0.0	36228
ARTEMIS F-1	1997	Sep-97 Delta	MC	Planned	11187	3000	9.8	9.8	LUN SUR	-	-	-	34768
NOAA-L	1997	May-97 Titan II	MC	Planned	2883	2255	14	6	LEO SYN	460	460	98.7	26702
TDRS-H	1997	Oct-97 STS/IUS	STS	Planned	27580	4905	19	9.8	GEO	19310	19310	0.0	38267
Other Shuttle P/L	1997		STS	Potential	40000	40000			LEO	160	160	28.5	24579
Other Shuttle P/L	1997		STS	Potential	40000	40000			LEO	160	160	28.5	24579
Other Shuttle P/L	1997		STS	Potential	40000	40000			LEO	160	160	28.5	24579
SCOUT-2	1997	Mar-97 Delta	MC	Planned	4585	1200	23	10	LUN ORB	-	-	-	34768
SSF Assembly	1997		STS	Potential	36300	36300			LEO SSF	220	220	28.5	24579
SSF Assembly	1997		STS	Potential	36300	36300			LEO SSF	220	220	28.5	24579
SSF Assembly	1997		STS	Potential	36300	36300			LEO SSF	220	220	28.5	24579
SSF Assembly	1997		STS	Potential	36300	36300			LEO SSF	220	220	28.5	24579
ARISTOTELES	1998		MC	Potential	5443	4840	23	6.6	LEO POL	270	270	95.0	26088
ARTEMIS F-2	1998	Sep-98 Delta	MC	Planned	11187	3000	9.8	9.8	LUN SUR	-	-	-	34768
CASSINI	1998	Oct-97 Titan IV/Cent	LC	Planned	18294	4510	11	12.97	DS SAT	-	-	-	34768

NAME	FY	Launch			MASS TO LEO	PLD MASS	LENGTH	DIAMETER	DEST	APO	PER	INC	DELTA V
EOS-AM 1	1998	Jun-98	IC	Planned	13554	12000	0	0	LEO POL	381	381	98.2	26452
NEAR	1998	Jan-98 Delta	MC	Planned	5710	1566	5.9	6.2	DS C/A	-	-	-	34768
PLUTO FLYBY	1998		LC	Potential	32132	9934	11	13	DS PLU	-	-	-	34768
SSF Assembly	1998		STS	Potential	36300	36300			LEO SSF	220	220	28.5	24579
SSF Assembly	1998		STS	Potential	36300	36300			LEO SSF	220	220	28.5	24579
SSF Assembly	1998		STS	Potential	36300	36300			LEO SSF	220	220	28.5	24579
SSF Assembly	1998		STS	Potential	36300	36300			LEO SSF	220	220	28.5	24579
SSF Utilization	1998		STS	Potential	36300	36300			LEO SSF	220	220	28.5	24579
SSF Utilization	1998		STS	Potential	36300	36300			LEO SSF	220	220	28.5	24579
SSF Utilization	1998		STS	Potential	36300	36300			LEO SSF	220	220	28.5	24579
STEP	1998		MC	Potential									
TDRS II-F1	1998	Apr-99	IC	Planned	27580	4905	19	9.8	GEO	19310	19310	0.0	38267
ARTEMIS	1999		IC	Potential	16867	4000	10	8.33	LUN SUR	-	-	-	34768
AXAF-S	1999	Sep-99 Delta	MC	Planned	8628	7400	44	14	LEO OTH	320	320	28.5	26614
GOES-K	1999	Apr-99 Atlas I	IC	Planned	16470	2160	9	9	GEO	19310	19310	0.0	38267
HESP	1999		MC	Potential	5786	5060	20	7	EAR OTH	324	324	90.0	26267
MESUR	1999		MC	Potential	4941	1316	0	5.6	DS MAR SUR	-	-	-	34768
MESUR	1999		MC	Potential	4941	1316	0	5.6	DS MAR SUR	-	-	-	34768
NOAA-M	1999	Jan-99 Titan II	MC	Planned	2883	2255	14	6	LEO SYN	460	460	98.7	26702
PLUTO FLYBY	1999		LC	Potential	32132	9934	11	13	DS PLU	-	-	-	34768
SSF Assembly	1999		STS	Potential	36300	36300			LEO SSF	220	220	28.5	24579
SSF Assembly	1999		STS	Potential	36300	36300			LEO SSF	220	220	28.5	24579
SSF Assembly	1999		STS	Potential	36300	36300			LEO SSF	220	220	28.5	24579
SSF Assembly	1999		STS	Potential	36300	36300			LEO SSF	220	220	28.5	24579
SSF Utilization	1999		STS	Potential	36300	36300			LEO SSF	220	220	28.5	24579
SSF Utilization	1999		STS	Potential	36300	36300			LEO SSF	220	220	28.5	24579
SSF Utilization	1999		STS	Potential	36300	36300			LEO SSF	220	220	28.5	24579
TIMED H, L	1999		MC	Potential	11217	2750	7.5	6.6	LEO SYN	3238	81	0.0	35463
TIMED H, L	1999		MC	Potential	11217	2750	7.5	6.6	LEO SYN	3238	81	0.0	35463
ARTEMIS	2000		IC	Potential	16867	4000	10	8.33	LUN SUR	-	-	-	34768
EOS-PM 1	2000		IC	Potential	13885	12300	0	0	LEO POL	381	381	98.2	26452
EOS/SAR	2000		MC	Potential	3428	2867	35.8	8.5	LEO SYN	335	335	97.5	26303
FUSE	2000		MC	Potential	2867	2867	23	6.9	LEO OTH				
GOES-L	2000		IC	Potential	16470	2160	9	9	GEO	19310	19310	0.0	38267
IMI	2000		MC	Potential	5185	1870	23	8	EAR OTH	20663	216	90.0	33772
INTEGRAL	2000		IC	Potential	3063	2645	10	5	LEO OTH	189	189	28.0	24592
MIDEX	2000		MC	Potential	3000	3000	9	6	LEO OTH				
NOAA-N	2000		MC	Potential	2883	2255	14	6	LEO SYN	460	460	98.7	26702
SSF Assembly	2000		STS	Potential	36300	36300			LEO SSF	220	220	28.5	24579
SSF Assembly	2000		STS	Potential	36300	36300			LEO SSF	220	220	28.5	24579
SSF Assembly	2000		STS	Potential	36300	36300			LEO SSF	220	220	28.5	24579
SSF Assembly	2000		STS	Potential	36300	36300			LEO SSF	220	220	28.5	24579
SSF Utilization	2000		STS	Potential	36300	36300			LEO SSF	220	220	28.5	24579
SSF Utilization	2000		STS	Potential	36300	36300			LEO SSF	220	220	28.5	24579
SSF Utilization	2000		STS	Potential	36300	36300			LEO SSF	220	220	28.5	24579
TDRS II-F2	2000		IC	Planned	27580	4905	19	9.8	GEO	19310	19310	0.0	38267
ARTEMIS	2001		IC	Potential	16867	4000	10	8.33	LUN SUR	-	-	-	34768
ASTROMAG	2001		IC	Potential	25566	12566	13	12	LEO SS RM	220	220	28.5	24678
GTC	2001		IC	Potential	9866	9866	0	0	EAR OTH			90.0	
LUNAR SCOUT	2001		MC	Potential	7885	2000	0	6	LUN ORB	-	-	-	34768
MESUR	2001		MC	Potential	4941	1316	0	5.6	DS MAR SUR	-	-	-	34768
MESUR	2001		MC	Potential	4941	1316	0	5.6	DS MAR SUR	-	-	-	34768
MIDEX	2001		MC	Potential	3000	3000	9	6	LEO OTH				
NOAA-N	2001		MC	Potential	2883	2255	14	6	LEO SYN	460	460	98.7	26702
Other Shuttle P/L	2001		STS	Potential	40000	40000			LEO	160	160	28.5	24579

NAME	FY	Launch		MASS TO LEO	PLD MASS	LENGTH	DIAMETER	DEST	APO	PER	INC	DELTA V
Other Shuttle P/L	2001	STS	Potential	40000	40000			LEO	160	160	28.5	24579
PROBE (M)	2001	MC	Potential	7000	7000	9	6	EAR OTH				
SMM	2001	MC	Potential	6135	2000	10	10	LEO SYN	37797	540	90.0	34806
SSF Logistics/Crew R	2001	STS	Potential	36300	36300			LEO SSF	220	220	28.5	24579
SSF Logistics/Crew R	2001	STS	Potential	36300	36300			LEO SSF	220	220	28.5	24579
SSF Logistics/Crew R	2001	STS	Potential	36300	36300			LEO SSF	220	220	28.5	24579
SSF Logistics/Crew R	2001	STS	Potential	36300	36300			LEO SSF	220	220	28.5	24579
SSF Logistics/Crew R	2001	STS	Potential	36300	36300			LEO SSF	220	220	28.5	24579
TDRS II-F3	2001	IC	Potential	27580	4905	19	9.8	GEO	19310	19310	0.0	38267
ALT	2002	MC	Potential	6000	6000	0	0	LEO OTH				
ARTEMIS	2002	IC	Potential	16867	4000	10	8.33	LUN SUR	.	.	.	34768
CHEM	2002	IC	Potential	12000	12000	0	0	LEO POL	400	400	90.0	26452
CNM	2002	LC	Potential	32132	9934	11	13	DS C/A	.	.	.	34768
IAM	2002	IC	Potential									
MIDEX	2002	MC	Potential	3000	3000	9	6	LEO OTH		•		
OSL	2002	MC	Potential	4361	3813	15	9.2	LEO SYN	275	275	97.4	26104
Other Shuttle P/L	2002	STS	Potential	40000	40000			LEO	160	160	28.5	24579
Other Shuttle P/L	2002	STS	Potential	40000	40000			LEO	160	160	28.5	24579
SSF Logistics/Crew R	2002	STS	Potential	36300	36300			LEO SSF	220	220	28.5	24579
SSF Logistics/Crew R	2002	STS	Potential	36300	36300			LEO SSF	220	220	28.5	24579
SSF Logistics/Crew R	2002	STS	Potential	36300	36300			LEO SSF	220	220	28.5	24579
SSF Logistics/Crew R	2002	STS	Potential	36300	36300			LEO SSF	220	220	28.5	24579
SSF Logistics/Crew R	2002	STS	Potential	36300	36300			LEO SSF	220	220	28.5	24579
TDRS II-F4	2002	IC	Potential	27580	4905	19	9.8	GEO	19310	19310	0.0	38267
ARTEMIS	2003	IC	Potential	16867	4000	10	8.33	LUN SUR	.	.	.	34768
EOS-AM 2	2003	IC	Potential	13554	12000	0	0	LEO POL	381	381	98.2	26452
MIDEX	2003	MC	Potential	3000	3000	9	6	LEO OTH				
NOAA-O	2003	MC	Potential	2883	2255	14	6	LEO SYN	460	460	98.7	26702
Other Shuttle P/L	2003	STS	Potential	40000	40000			LEO	160	160	28.5	24579
Other Shuttle P/L	2003	STS	Potential	40000	40000			LEO	160	160	28.5	24579
PROBE (M)	2003	MC	Potential	7000	7000	9	6	EAR OTH				
SSF Logistics/Crew R	2003	STS	Potential	36300	36300			LEO SSF	220	220	28.5	24579
SSF Logistics/Crew R	2003	STS	Potential	36300	36300			LEO SSF	220	220	28.5	24579
SSF Logistics/Crew R	2003	STS	Potential	36300	36300			LEO SSF	220	220	28.5	24579
SSF Logistics/Crew R	2003	STS	Potential	36300	36300			LEO SSF	220	220	28.5	24579
SSF Logistics/Crew R	2003	STS	Potential	36300	36300			LEO SSF	220	220	28.5	24579
TDRS II-F5	2003	IC	Potential	27580	4905	19	9.8	GEO	19310	19310	0.0	38267
AIM	2004	MC	Potential			0	10					
ARTEMIS	2004	IC	Potential	16867	4000	10	8.33	LUN SUR	.	.	-	34768
GOES-M	2005	IC	Potential	16470	2160	9	9	GEO	19310	19310	0.0	38267
MIDEX	2004	MC	Potential	3000	3000	9	6	LEO OTH				
NAE	2004	IC	Potential	3063	2645	10	5	LEO OTH	189	189	28.0	24592
Other Shuttle P/L	2004	STS	Potential	40000	40000			LEO	160	160	28.5	24579
Other Shuttle P/L	2004	STS	Potential	40000	40000			LEO	160	160	28.5	24579
SSF Logistics/Crew R	2004	STS	Potential	36300	36300			LEO SSF	220	220	28.5	24579
SSF Logistics/Crew R	2004	STS	Potential	36300	36300			LEO SSF	220	220	28.5	24579
SSF Logistics/Crew R	2004	STS	Potential	36300	36300			LEO SSF	220	220	28.5	24579
SSF Logistics/Crew R	2004	STS	Potential	36300	36300			LEO SSF	220	220	28.5	24579
SSF Logistics/Crew R	2004	STS	Potential	36300	36300			LEO SSF	220	220	28.5	24579
TDRS II-F6	2004	IC	Potential	27580	4905	19	9.8	GEO	19310	19310	0.0	38267
ARTEMIS	2005	IC	Potential	16867	4000	10	8.33	LUN SUR	.	.	.	34768
EOS-PM 2	2005	IC	Potential	13885	12300	0	0	LEO POL	381	381	98.2	26452
GEOPLAT	2005	LC	Potential	48458	12700	20	15	GEO	19310	19310	0.0	38267
GOES-N	2006	IC	Potential	16470	2160	9	9	GEO	19310	19310	0.0	38267

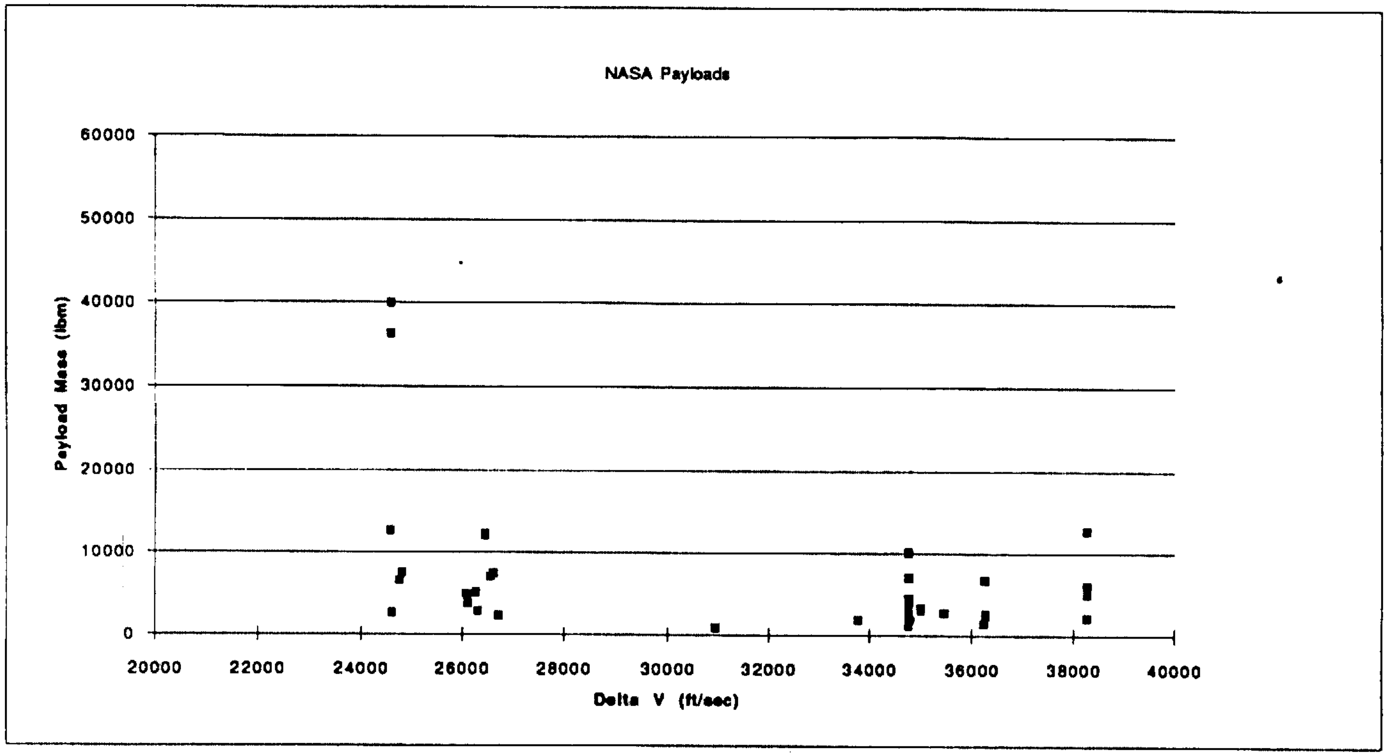
NAME		FY	Launch		MASS TO LEO	PLD MASS	LENGTH	DIAMETER	DEST	APO	PER	INC	DELTA V
LTT		2005	LC	Potential	8539	2204	6.56	6.56	LUN SUR	-	-	-	34768
MIDEX		2005	MC	Potential	3000	3000	9	6	LEO OTH				
MSR		2005	IC	Potential	32734	10221	0	0	DS MAR SUR	-	-	-	34768
MSR		2005	IC	Potential	32734	10221	0	0	DS MAR SUR	-	-	-	34768
Other Shuttle P/L		2005	STS	Potential	40000	40000			LEO	160	160	28.5	24579
Other Shuttle P/L		2005	STS	Potential	40000	40000			LEO	160	160	28.5	24579
PROBE (M)		2005	MC	Potential	7000	7000	9	6	EAR OTH				
SSF Logistics/Crew	R	2005	STS	Potential	36300	36300			LEO SSF	220	220	28.5	24579
SSF Logistics/Crew	R	2005	STS	Potential	36300	36300			LEO SSF	220	220	28.5	24579
SSF Logistics/Crew	R	2005	STS	Potential	36300	36300			LEO SSF	220	220	28.5	24579
SSF Logistics/Crew	R	2005	STS	Potential	36300	36300			LEO SSF	220	220	28.5	24579
SSF Logistics/Crew	R	2005	STS	Potential	36300	36300			LEO SSF	220	220	28.5	24579
TDRS II-F7		2005	IC	Potential	27580	4905	19	9.8	GEO	19310	19310	0.0	38267
ARTEMIS		2006	IC	Potential	16867	4000	10	8.33	LUN SUR	-	-	-	34768
MIDEX		2006	MC	Potential	3000	3000	9	6	LEO OTH				
NOAA-P		2006	MC	Potential	2883	2255	14	6	LEO SYN	460	460	98.7	26702
Other Shuttle P/L		2006	STS	Potential	40000	40000			LEO	160	160	28.5	24579
Other Shuttle P/L		2006	STS	Potential	40000	40000			LEO	160	160	28.5	24579
SSF Logistics/Crew	R	2006	STS	Potential	36300	36300			LEO SSF	220	220	28.5	24579
SSF Logistics/Crew	R	2006	STS	Potential	36300	36300			LEO SSF	220	220	28.5	24579
SSF Logistics/Crew	R	2006	STS	Potential	36300	36300			LEO SSF	220	220	28.5	24579
SSF Logistics/Crew	R	2006	STS	Potential	36300	36300			LEO SSF	220	220	28.5	24579
SSF Logistics/Crew	R	2006	STS	Potential	36300	36300			LEO SSF	220	220	28.5	24579
ALT-2		2007	MC	Potential	6000	6000	0	0	LEO OTH				
ARTEMIS		2007	IC	Potential	16867	4000	10	8.33	LUN SUR	-	-	-	34768
CHEM 2		2007	IC	Potential	12000	12000	0	0	LEO POL	400	400	90.0	26452
MIDEX		2007	MC	Potential	3000	3000	9	6	LEO OTH				
Other Shuttle P/L		2007	STS	Potential	40000	40000			LEO	160	160	28.5	24579
Other Shuttle P/L		2007	STS	Potential	40000	40000			LEO	160	160	28.5	24579
PROBE (M)		2007	MC	Potential	7000	7000	9	6	EAR OTH				
SSF Logistics/Crew	R	2007	STS	Potential	36300	36300			LEO SSF	220	220	28.5	24579
SSF Logistics/Crew	R	2007	STS	Potential	36300	36300			LEO SSF	220	220	28.5	24579
SSF Logistics/Crew	R	2007	STS	Potential	36300	36300			LEO SSF	220	220	28.5	24579
SSF Logistics/Crew	R	2007	STS	Potential	36300	36300			LEO SSF	220	220	28.5	24579
SSF Logistics/Crew	R	2007	STS	Potential	36300	36300			LEO SSF	220	220	28.5	24579
TDRS II-F8		2007	IC	Potential	27580	4905	19	9.8	GEO	19310	19310	0.0	38267
ARTEMIS		2008	IC	Potential	16867	4000	10	8.33	LUN SUR	-	-	-	34768
EOS-AM 3		2008	IC	Potential	13554	12000	0	0	LEO POL	381	381	98.2	26452
MIDEX		2008	MC	Potential	3000	3000	9	6	LEO OTH				
Other Shuttle P/L		2008	STS	Potential	40000	40000			LEO	160	160	28.5	24579
Other Shuttle P/L		2008	STS	Potential	40000	40000			LEO	160	160	28.5	24579
SSF Logistics/Crew	R	2008	STS	Potential	36300	36300			LEO SSF	220	220	28.5	24579
SSF Logistics/Crew	R	2008	STS	Potential	36300	36300			LEO SSF	220	220	28.5	24579
SSF Logistics/Crew	R	2008	STS	Potential	36300	36300			LEO SSF	220	220	28.5	24579
SSF Logistics/Crew	R	2008	STS	Potential	36300	36300			LEO SSF	220	220	28.5	24579
SSF Logistics/Crew	R	2008	STS	Potential	36300	36300			LEO SSF	220	220	28.5	24579
ARTEMIS		2009	IC	Potential	16867	4000	10	8.33	LUN SUR	-	-	-	34768
MIDEX		2009	MC	Potential	3000	3000	9	6	LEO OTH				
NOAA-Q		2009	MC	Potential	2883	2255	14	6	LEO SYN	460	460	98.7	26702
Other Shuttle P/L		2009	STS	Potential	40000	40000			LEO	160	160	28.5	24579
Other Shuttle P/L		2009	STS	Potential	40000	40000			LEO	160	160	28.5	24579
PROBE (M)		2009	MC	Potential	7000	7000	9	6	EAR OTH				
SSF Logistics/Crew	R	2009	STS	Potential	36300	36300			LEO SSF	220	220	28.5	24579
SSF Logistics/Crew	R	2009	STS	Potential	36300	36300			LEO SSF	220	220	28.5	24579

NAME		FY	Launch		MASS TO LEO	PLD MASS	LENGTH	DIAMETER	DEST	APO	PER	INC	DELTA V
SSF Logistics/Crew	R	2009	STS	Potential	36300	36300			LEO SSF	220	220	28.5	24579
SSF Logistics/Crew	R	2009	STS	Potential	36300	36300			LEO SSF	220	220	28.5	24579
SSF Logistics/Crew	R	2009	STS	Potential	36300	36300			LEO SSF	220	220	28.5	24579
TDRS II-F9		2009	IC	Potential	27580	4905	19	9.8	GEO	19310	19310	0.0	38267
ARTEMIS		2010	IC	Potential	16867	4000	10	8.33	LUN SUR	-	-	-	34768
EOS-PM 3		2010	IC	Potential	13885	12300	0	0	LEO POL	381	381	98.2	26452
MIDEX		2010	MC	Potential	3000	3000	9	6	LEO OTH				
Other Shuttle P/L		2010	STS	Potential	40000	40000			LEO	160	160	28.5	24579
Other Shuttle P/L		2010	STS	Potential	40000	40000			LEO	160	160	28.5	24579
SOLAR PROBE		2010	LC	Potential	8543	2205	65.6	13	DS SOL	-	-	-	34768
SSF Logistics/Crew	R	2010	STS	Potential	36300	36300			LEO SSF	220	220	28.5	24579
SSF Logistics/Crew	R	2010	STS	Potential	36300	36300			LEO SSF	220	220	28.5	24579
SSF Logistics/Crew	R	2010	STS	Potential	36300	36300			LEO SSF	220	220	28.5	24579
SSF Logistics/Crew	R	2010	STS	Potential	36300	36300			LEO SSF	220	220	28.5	24579
SSF Logistics/Crew	R	2010	STS	Potential	36300	36300			LEO SSF	220	220	28.5	24579
Generic DS-2		2011	IC	Extrapolatec	28752	7000			DS Planetary	-	-	-	34768
Generic EO-1		2011	MC	Extrapolatec	10943	2400			EAR OTH	31000	6000	90.0	36259
Generic GEO		2011	IC	Extrapolatec	27833	5000			GEO	19310	19310	0.0	38267
Generic LEO OTH		2011	MC	Extrapolatec	8310	7500			LEO OTH	270	270	28.5	24800
Generic LEO SYN		2011	MC	Extrapolatec	11462	3000			LEO SYN	450	450	90.0	35000
Generic LUN-2		2011	IC	Extrapolatec	16866	4000			LUNAR	-	-	-	34768
Other Shuttle P/L		2011	STS	Extrapolatec	40000	40000			LEO	160	160	28.5	24579
Other Shuttle P/L		2011	STS	Extrapolatec	40000	40000			LEO	160	160	28.5	24579
SSF Logistics/Crew	R	2011	STS	Extrapolatec	36300	36300			LEO SSF	220	220	28.5	24579
SSF Logistics/Crew	R	2011	STS	Extrapolatec	36300	36300			LEO SSF	220	220	28.5	24579
SSF Logistics/Crew	R	2011	STS	Extrapolatec	36300	36300			LEO SSF	220	220	28.5	24579
SSF Logistics/Crew	R	2011	STS	Extrapolatec	36300	36300			LEO SSF	220	220	28.5	24579
SSF Logistics/Crew	R	2011	STS	Extrapolatec	36300	36300			LEO SSF	220	220	28.5	24579
Generic DS-3		2012	LC	Extrapolatec	32270	10000			DS Planetary	-	-	-	34768
Generic GEO		2012	IC	Extrapolatec	27833	5000			GEO	19310	19310	0.0	38267
Generic LEO OTH		2012	MC	Extrapolatec	8310	7500			LEO OTH	270	270	28.5	24800
Generic LEO POL		2012	IC	Extrapolatec	13885	12300			LEO POL	381	381	98.2	26453
Generic LEO SYN		2012	MC	Extrapolatec	11462	3400			LEO SYN	450	450	90.0	35000
Generic LUN-2		2012	IC	Extrapolatec	16866	4000			LUNAR	-	-	-	34768
Other Shuttle P/L		2012	STS	Extrapolatec	40000	40000			LEO	160	160	28.5	24579
Other Shuttle P/L		2012	STS	Extrapolatec	40000	40000			LEO	160	160	28.5	24579
SSF Logistics/Crew	R	2012	STS	Extrapolatec	36300	36300			LEO SSF	220	220	28.5	24579
SSF Logistics/Crew	R	2012	STS	Extrapolatec	36300	36300			LEO SSF	220	220	28.5	24579
SSF Logistics/Crew	R	2012	STS	Extrapolatec	36300	36300			LEO SSF	220	220	28.5	24579
SSF Logistics/Crew	R	2012	STS	Extrapolatec	36300	36300			LEO SSF	220	220	28.5	24579
SSF Logistics/Crew	R	2012	STS	Extrapolatec	36300	36300			LEO SSF	220	220	28.5	24579
Generic DS-1		2013	MC	Extrapolatec	5815	1600			DS Planetary	-	-	-	34768
Generic EO-1		2013	MC	Extrapolatec	10943	2400			EAR OTH	31000	6000	90.0	36259
Generic GEO		2013	IC	Extrapolatec	27833	5000			GEO	19310	19310	0.0	38267
Generic LEO OTH		2013	MC	Extrapolatec	8310	7500			LEO OTH	270	270	28.5	24800
Generic LEO SYN		2013	MC	Extrapolatec	11462	3400			LEO SYN	450	450	90.0	35000
Generic LUN-2		2013	IC	Extrapolatec	16866	4000			LUNAR	-	-	-	34768
Other Shuttle P/L		2013	STS	Extrapolatec	40000	40000			LEO	160	160	28.5	24579
Other Shuttle P/L		2013	STS	Extrapolatec	40000	40000			LEO	160	160	28.5	24579
SSF Logistics/Crew	R	2013	STS	Extrapolatec	36300	36300			LEO SSF	220	220	28.5	24579
SSF Logistics/Crew	R	2013	STS	Extrapolatec	36300	36300			LEO SSF	220	220	28.5	24579
SSF Logistics/Crew	R	2013	STS	Extrapolatec	36300	36300			LEO SSF	220	220	28.5	24579
SSF Logistics/Crew	R	2013	STS	Extrapolatec	36300	36300			LEO SSF	220	220	28.5	24579
SSF Logistics/Crew	R	2013	STS	Extrapolatec	36300	36300			LEO SSF	220	220	28.5	24579

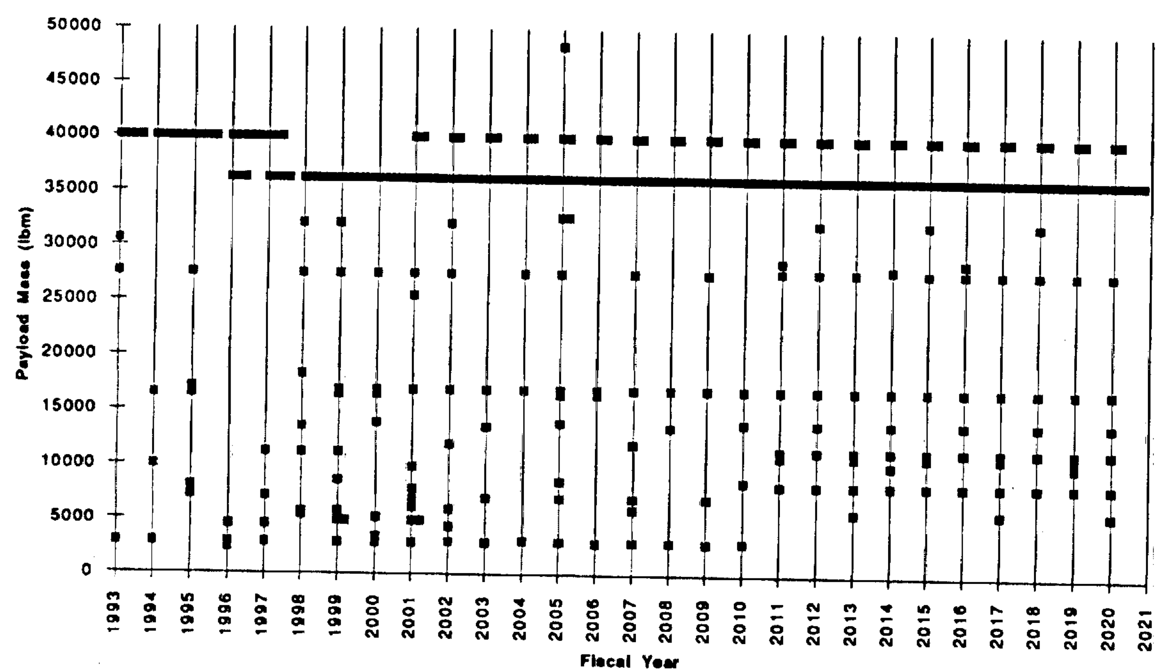
NAME		FY	Launch			MASS TO LEO	PLD MASS	LENGTH	DIAMETER	DEST	APO	PER	INC	DELTA V
Generic EO-2		2014		IC	Extrapolatec	28123	6700			EAR OTH	31000	6000	90.0	36250
Generic LEO OTH		2014		MC	Extrapolatec	8310	7500			LEO OTH	270	270	28.5	24800
Generic LEO POL		2014		IC	Extrapolatec	13885	12300			LEO POL	381	381	98.2	26453
Generic LEO SYN		2014		MC	Extrapolatec	11462	3400			LEO SYN	450	450	90.0	35000
Generic LUN-1		2014		MC	Extrapolatec	10,132	2700			LUNAR	-	-	-	34768
Generic LUN-2		2014		IC	Extrapolatec	16866	4000			LUNAR	-	-	-	34768
Other Shuttle P/L		2014		STS	Extrapolatec	40000	40000			LEO	160	160	28.5	24579
Other Shuttle P/L		2014		STS	Extrapolatec	40000	40000			LEO	160	160	28.5	24579
SSF Logistics/Crew	R	2014		STS	Extrapolatec	36300	36300			LEO SSF	220	220	28.5	24579
SSF Logistics/Crew	R	2014		STS	Extrapolatec	36300	36300			LEO SSF	220	220	28.5	24579
SSF Logistics/Crew	R	2014		STS	Extrapolatec	36300	36300			LEO SSF	220	220	28.5	24579
SSF Logistics/Crew	R	2014		STS	Extrapolatec	36300	36300			LEO SSF	220	220	28.5	24579
SSF Logistics/Crew	R	2014		STS	Extrapolatec	36300	36300			LEO SSF	220	220	28.5	24579
Generic DS-3		2015		LC	Extrapolatec	32270	10000			DS Planetary	-	-	-	34768
Generic EO-1		2015		MC	Extrapolatec	10943	2400			EAR OTH	31000	6000	90.0	36259
Generic GEO		2015		IC	Extrapolatec	27833	5000			GEO	19310	19310	0.0	38267
Generic LEO OTH		2015		MC	Extrapolatec	8310	7500			LEO OTH	270	270	28.5	24800
Generic LEO SYN		2015		MC	Extrapolatec	11462	3400			LEO SYN	450	450	90.0	35000
Generic LUN-2		2015		IC	Extrapolatec	16866	4000			LUNAR	-	-	-	34768
Other Shuttle P/L		2015		STS	Extrapolatec	40000	40000			LEO	160	160	28.5	24579
Other Shuttle P/L		2015		STS	Extrapolatec	40000	40000			LEO	160	160	28.5	24579
SSF Logistics/Crew	R	2015		STS	Extrapolatec	36300	36300			LEO SSF	220	220	28.5	24579
SSF Logistics/Crew	R	2015		STS	Extrapolatec	36300	36300			LEO SSF	220	220	28.5	24579
SSF Logistics/Crew	R	2015		STS	Extrapolatec	36300	36300			LEO SSF	220	220	28.5	24579
SSF Logistics/Crew	R	2015		STS	Extrapolatec	36300	36300			LEO SSF	220	220	28.5	24579
SSF Logistics/Crew	R	2015		STS	Extrapolatec	36300	36300			LEO SSF	220	220	28.5	24579
Generic DS-2		2016		IC	Extrapolatec	28752	7000			DS Planetary	-	-	-	34768
Generic GEO		2016		IC	Extrapolatec	27833	5000			GEO	19310	19310	0.0	38267
Generic LEO OTH		2016		MC	Extrapolatec	8310	7500			LEO OTH	270	270	28.5	24800
Generic LEO POL		2016		IC	Extrapolatec	13885	12300			LEO POL	381	381	98.2	26453
Generic LEO SYN		2016		MC	Extrapolatec	11462	3400			LEO SYN	450	450	90.0	35000
Generic LUN-2		2016		IC	Extrapolatec	16866	4000			LUNAR	-	-	-	34768
Other Shuttle P/L		2016		STS	Extrapolatec	40000	40000			LEO	160	160	28.5	24579
Other Shuttle P/L		2016		STS	Extrapolatec	40000	40000			LEO	160	160	28.5	24579
SSF Logistics/Crew	R	2016		STS	Extrapolatec	36300	36300			LEO SSF	220	220	28.5	24579
SSF Logistics/Crew	R	2016		STS	Extrapolatec	36300	36300			LEO SSF	220	220	28.5	24579
SSF Logistics/Crew	R	2016		STS	Extrapolatec	36300	36300			LEO SSF	220	220	28.5	24579
SSF Logistics/Crew	R	2016		STS	Extrapolatec	36300	36300			LEO SSF	220	220	28.5	24579
SSF Logistics/Crew	R	2016		STS	Extrapolatec	36300	36300			LEO SSF	220	220	28.5	24579
Generic DS-1		2017		MC	Extrapolatec	5815	1600			DS Planetary	-	-	-	34768
Generic EO-1		2017		MC	Extrapolatec	10943	2400			EAR OTH	31000	6000	90.0	36259
Generic GEO		2017		IC	Extrapolatec	27833	5000			GEO	19310	19310	0.0	38267
Generic LEO OTH		2017		MC	Extrapolatec	8310	7500			LEO OTH	270	270	28.5	24800
Generic LEO SYN		2017		MC	Extrapolatec	11462	3400			LEO SYN	450	450	90.0	35000
Generic LUN-2		2017		IC	Extrapolatec	16866	4000			LUNAR	-	-	-	34768
Other Shuttle P/L		2017		STS	Extrapolatec	40000	40000			LEO	160	160	28.5	24579
Other Shuttle P/L		2017		STS	Extrapolatec	40000	40000			LEO	160	160	28.5	24579
SSF Logistics/Crew	R	2017		STS	Extrapolatec	36300	36300			LEO SSF	220	220	28.5	24579
SSF Logistics/Crew	R	2017		STS	Extrapolatec	36300	36300			LEO SSF	220	220	28.5	24579
SSF Logistics/Crew	R	2017		STS	Extrapolatec	36300	36300			LEO SSF	220	220	28.5	24579
SSF Logistics/Crew	R	2017		STS	Extrapolatec	36300	36300			LEO SSF	220	220	28.5	24579
SSF Logistics/Crew	R	2017		STS	Extrapolatec	36300	36300			LEO SSF	220	220	28.5	24579
Generic DS-3		2018		LC	Extrapolatec	32270	10000			DS Planetary	-	-	-	34768
Generic GEO		2018		IC	Extrapolatec	27833	5000			GEO	19310	19310	0.0	38267

NAME		FY	Launch		MASS TO LEO	PLD MASS	LENGTH	DIAMETER	DEST	APO	PER	INC	DELTA V
Generic LEO OTH		2018	MC	Extrapolatec	8310	7500			LEO OTH	270	270	28.5	24800
Generic LEO POL		2018	IC	Extrapolatec	13885	12300			LEO POL	381	381	98.2	26453
Generic LEO SYN		2018	MC	Extrapolatec	11462	3400			LEO SYN	450	450	90.0	35000
Generic LUN-2		2018	IC	Extrapolatec	16866	4000			LUNAR	-	-	-	34768
Other Shuttle P/L		2018	STS	Extrapolatec	40000	40000			LEO	160	160	28.5	24579
Other Shuttle P/L		2018	STS	Extrapolatec	40000	40000			LEO	160	160	28.5	24579
SSF Logistics/Crew	R	2018	STS	Extrapolatec	36300	36300			LEO SSF	220	220	28.5	24579
SSF Logistics/Crew	R	2018	STS	Extrapolatec	36300	36300			LEO SSF	220	220	28.5	24579
SSF Logistics/Crew	R	2018	STS	Extrapolatec	36300	36300			LEO SSF	220	220	28.5	24579
SSF Logistics/Crew	R	2018	STS	Extrapolatec	36300	36300			LEO SSF	220	220	28.5	24579
SSF Logistics/Crew	R	2018	STS	Extrapolatec	36300	36300			LEO SSF	220	220	28.5	24579
Generic EO-1		2019	MC	Extrapolatec	10943	2400			EAR OTH	31000	6000	90.0	36259
Generic GEO		2019	IC	Extrapolatec	27833	5000			GEO	19310	19310	0.0	38267
Generic LEO OTH		2019	MC	Extrapolatec	8310	7500			LEO OTH	270	270	28.5	24800
Generic LEO SYN		2019	MC	Extrapolatec	11462	3400			LEO SYN	450	450	90.0	35000
Generic LUN-1		2019	MC	Extrapolatec	10132	2700			LUNAR	-	-	-	34768
Generic LUN-2		2019	IC	Extrapolatec	16866	4000			LUNAR	-	-	-	34768
Other Shuttle P/L		2019	STS	Extrapolatec	40000	40000			LEO	160	160	28.5	24579
Other Shuttle P/L		2019	STS	Extrapolatec	40000	40000			LEO	160	160	28.5	24579
SSF Logistics/Crew	R	2019	STS	Extrapolatec	36300	36300			LEO SSF	220	220	28.5	24579
SSF Logistics/Crew	R	2019	STS	Extrapolatec	36300	36300			LEO SSF	220	220	28.5	24579
SSF Logistics/Crew	R	2019	STS	Extrapolatec	36300	36300			LEO SSF	220	220	28.5	24579
SSF Logistics/Crew	R	2019	STS	Extrapolatec	36300	36300			LEO SSF	220	220	28.5	24579
SSF Logistics/Crew	R	2019	STS	Extrapolatec	36300	36300			LEO SSF	220	220	28.5	24579
Generic DS-1		2020	MC	Extrapolatec	5815	1600			DS Planetary	-	-	-	34768
Generic GEO		2020	IC	Extrapolatec	27833	5000			GEO	19310	19310	0.0	38267
Generic LEO OTH		2020	MC	Extrapolatec	8310	7500			LEO OTH	270	270	28.5	24800
Generic LEO POL		2020	IC	Extrapolatec	13885	12300			LEO POL	381	381	98.2	26453
Generic LEO SYN		2020	MC	Extrapolatec	11462	3400			LEO SYN	450	450	90.0	35000
Generic LUN-2		2020	IC	Extrapolatec	16866	4000			LUNAR	-	-	-	34768
Other Shuttle P/L		2020	STS	Extrapolatec	40000	40000			LEO	160	160	28.5	24579
Other Shuttle P/L		2020	STS	Extrapolatec	40000	40000			LEO	160	160	28.5	24579
SSF Logistics/Crew	R	2020	STS	Extrapolatec	36300	36300			LEO SSF	220	220	28.5	24579
SSF Logistics/Crew	R	2020	STS	Extrapolatec	36300	36300			LEO SSF	220	220	28.5	24579
SSF Logistics/Crew	R	2020	STS	Extrapolatec	36300	36300			LEO SSF	220	220	28.5	24579
SSF Logistics/Crew	R	2020	STS	Extrapolatec	36300	36300			LEO SSF	220	220	28.5	24579
SSF Logistics/Crew	R	2020	STS	Extrapolatec	36300	36300			LEO SSF	220	220	28.5	24579

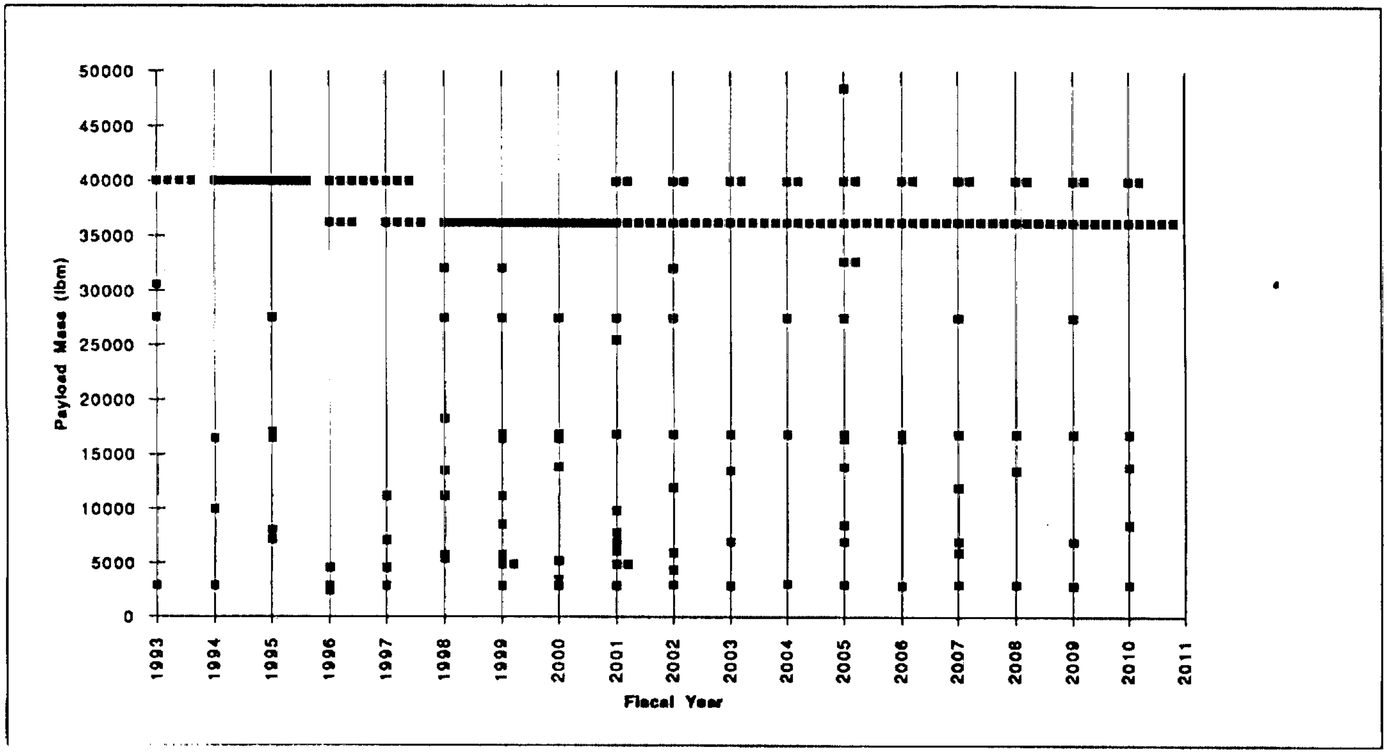
NAME	FY	Launch	MASS TO LEO	PLD MASS	LENGTH	DIAMETER	DEST	APO	PER	INC	DELTA V



NAME	FY	Launch	MASS TO LEO	PLD MASS	LENGTH	DIAMETER	DEST	APO	PER	INC	DELTA V



NAME	FY	Launch	MASS TO LEO	PLD MASS	LENGTH	DIAMETER	DEST	APO	PER	INC	DELTA V



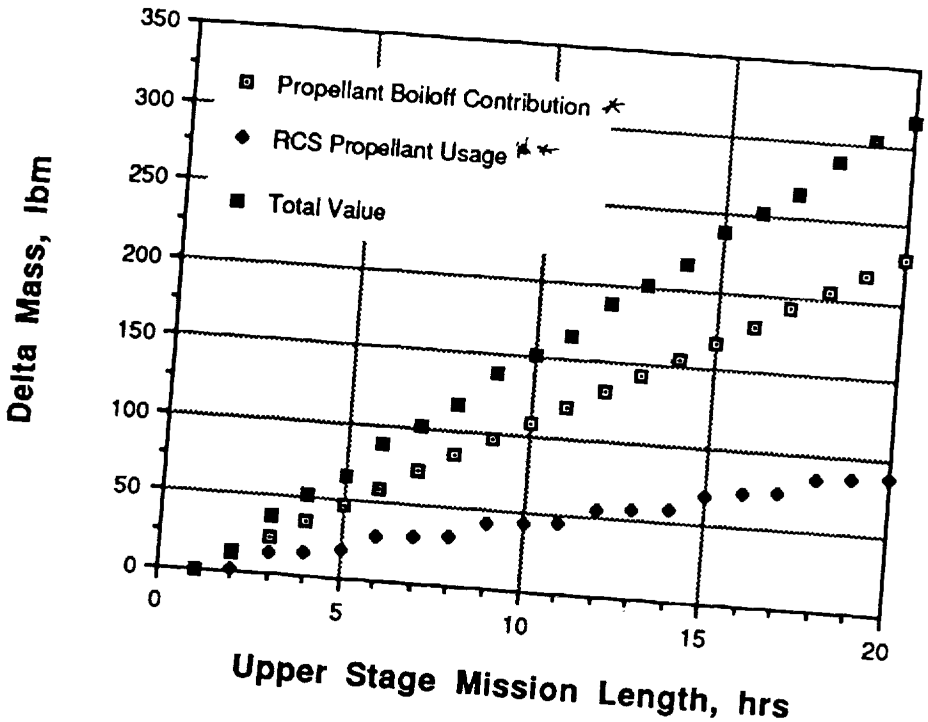
TD16-002
Upper Stage Technical Requirements Document
Source Analysis for Paragraph:
1.2a System Life

The 20 year system life requirement was based on engineering judgement of how long facilities will last without major refurbishment, and on the economic maintenance life of GSE.

TD16-003
Upper Stage Technical Requirements Document
Source Analysis for Paragraph:
1.3a Mission Life

Current baseline is to support missions of up to 20 hrs. total length. Propulsion system impacts of mission length are detailed in the attached graph for a medium (Centaur equivalent) Upper Stage. Avionics system impacts are almost exclusively in batteries: impacts are similar to propulsion system impacts, but about one-half the magnitude, based on past mission time-line / power profile analysis, worst case thermal profile.

Is this close to what you wanted?



* Assumes 5 layers of MLI, for both tanks, approximate propellant load of 45,000 lb

** Assume RCS settling required to vent tank every 3 hrs plus a thermal roll maneuver once an hour.

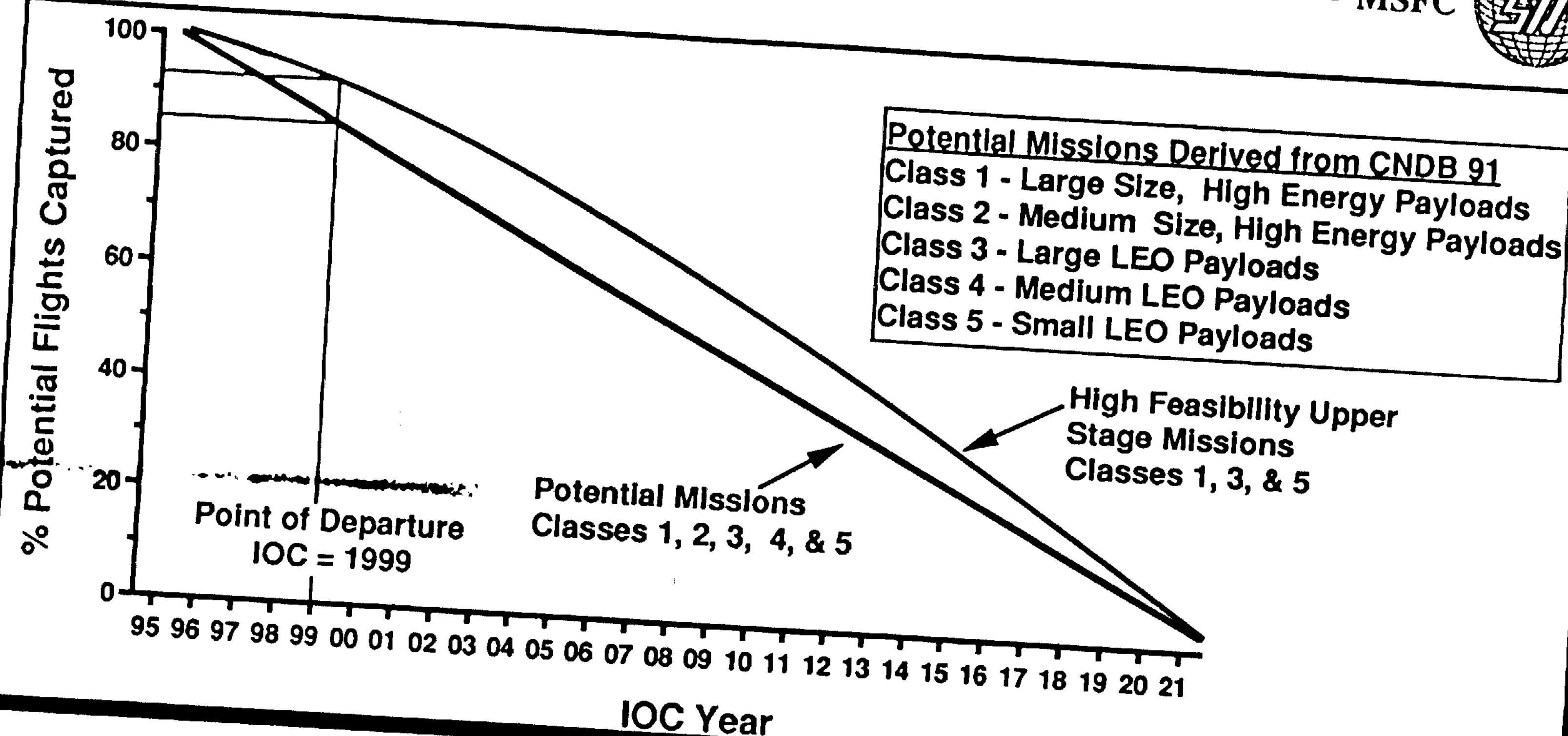
• Assessed subsystem insulation and heater requirement (Contribution determined to be in the noise level)

TD16-004
Upper Stage Technical Requirements Document
Source Analysis for Paragraph:
1.4a Initial Operational Capability

1.4a - Needs to be updated based on "new" mission model analysis

Upper Stage IOC Selection

MSFC



- Initial IOC Selection Attempts to Maximize Potential Flights Captured and Provide Credible Development Cycle for Program New Start

- Adverse Impacts May Result If IOC Slips
  - Payloads Must Be Slipped, Canceled, or Remanifested on Another System
  - Reduced System Cost Effectiveness (Less Flts to Bear Burden of Nonrecurring Cost)

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LR930218-IOC Vs Flts

Schedule Delay - Cost Sensitivity

MSFC

Also need to show ...
Also need to show funding ...

Assumptions:

Most cost effective development ... at ... 0.

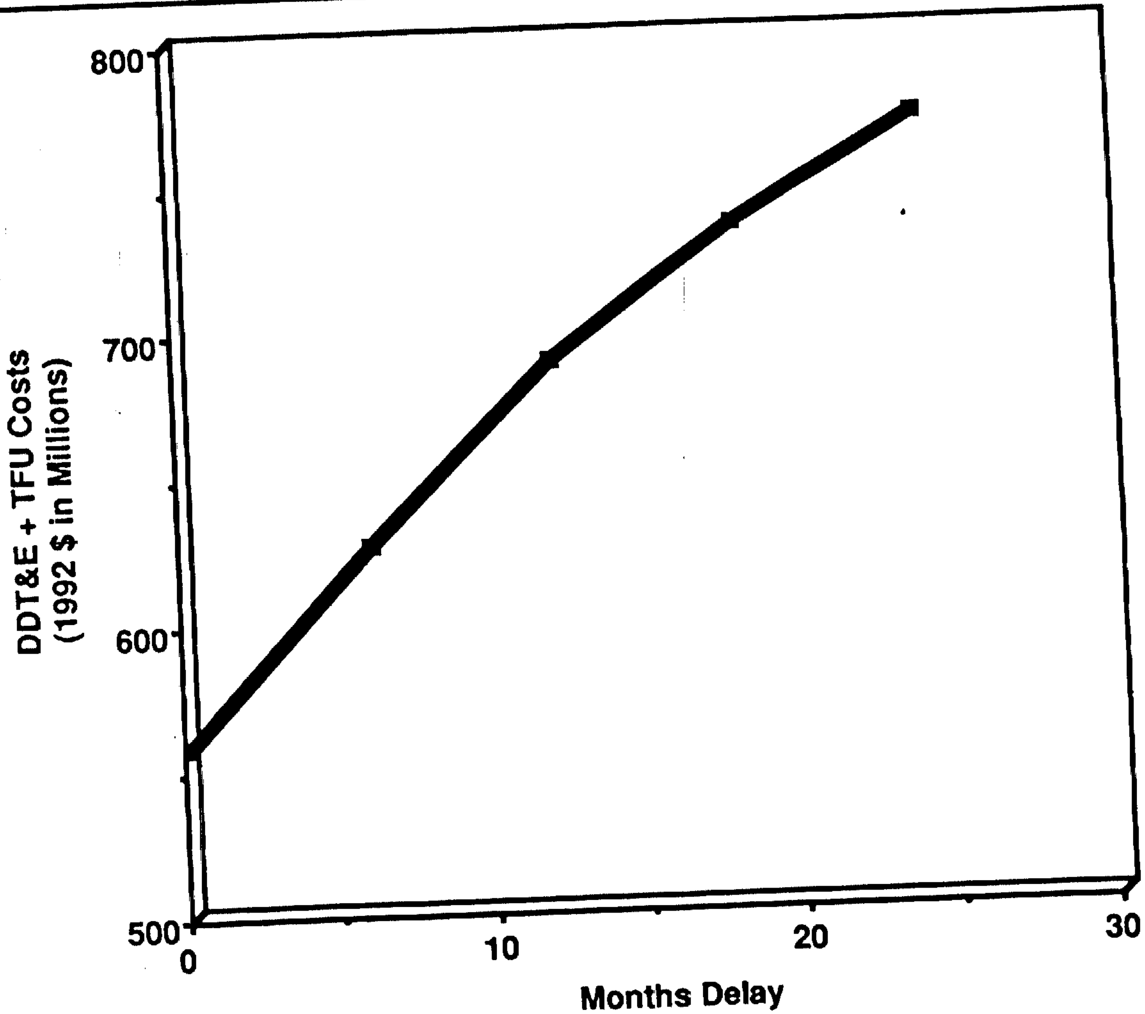
Factors Contributing to Cost Increase:

Manpower carryover

Additional T&E

Increased subcontractor cost

Additional launch site ...



Conclusions:

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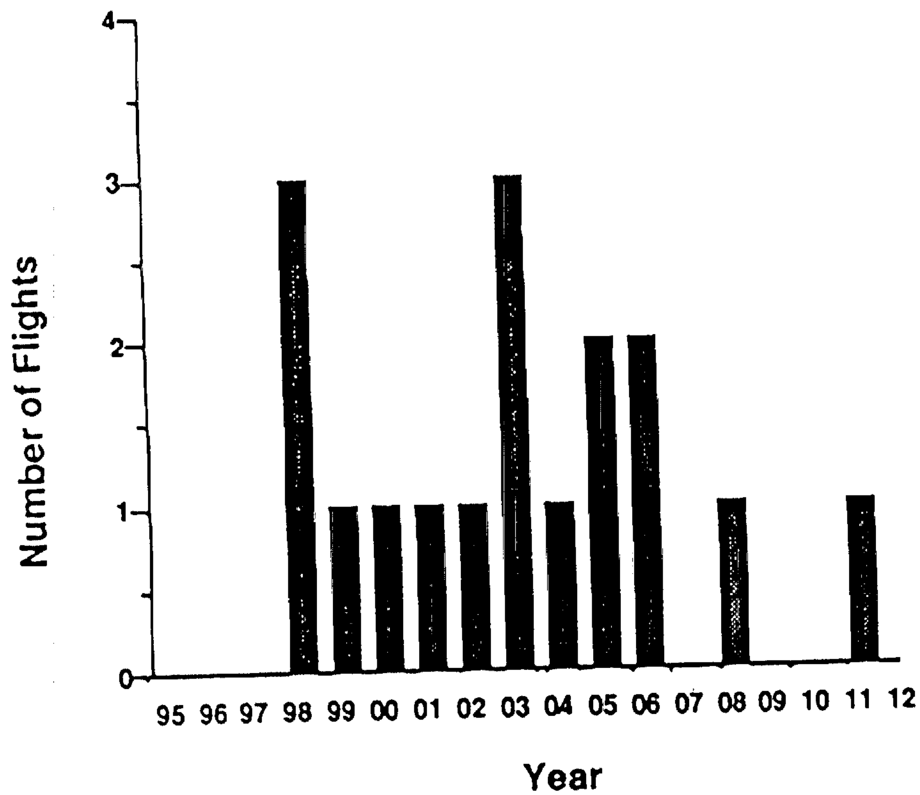
JC930111-04A

TD16-005
Upper Stage Technical Requirements Document
Source Analysis for Paragraph:
1.5a Launch - Site Capability

Small LEO Payloads - Class 5

MSFC



Payload Names/(Number of Flts)
CELSS SS Mod Project (1)
EOS Polar Orbiting Platforms (6)
Industrial Space Facility - Aux mod (1)
Hard X-Ray Imaging Facility (1)
Base Mission to SSF (2)
Growth Missions to SSF (5)
Variable Gravity Large Centrifuge Facility (1)

- Class 5
  - Op Apogee $\leq$ 500 nmi
  - Deliver Mass $\geq$ 20,000-29,999 lbs
- 17 Missions Possible Through Year 2021

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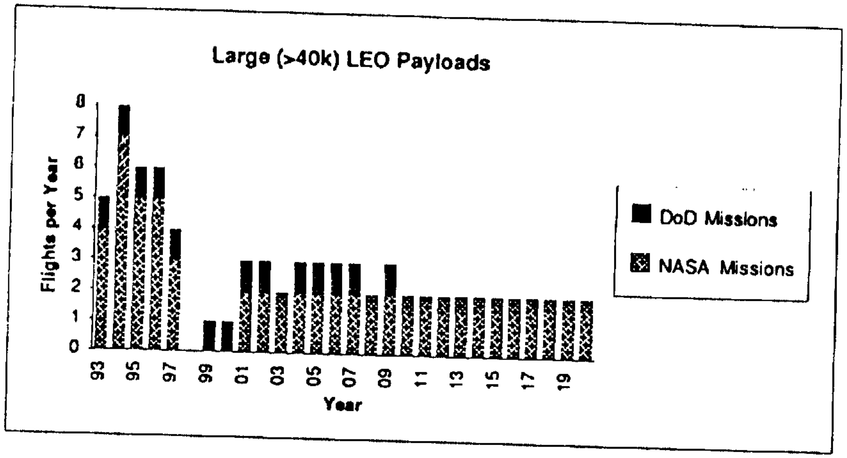
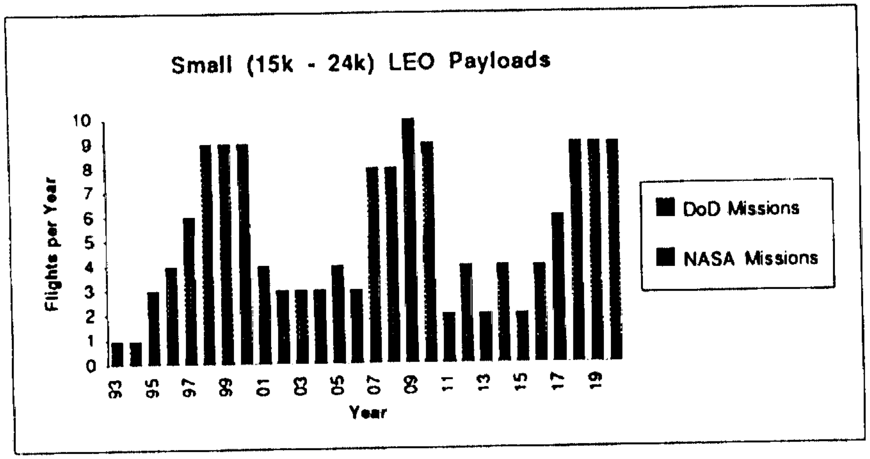
LR930216-05

LEO - Large (≥ 40k)

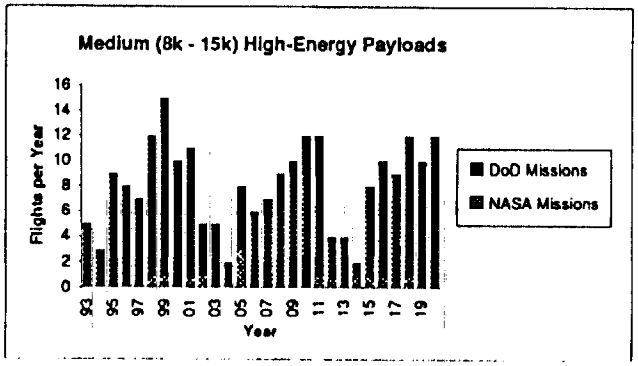
	93	94	95	96	97	98	99	00	01	02	03	04	05	06	07	08	09	10	11	12	13	14	15	16	17	18	19	20	1993 - 2010	2010 - 2020	1993 - 2020
NASA Total	4	7	5	5	3	0	0	0	2	2	2	2	2	2	2	2	2	2	2	2	2	2	2	2	2	2	2	2	44	20	64
Commercial Total	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
DoD Total	1	1	1	1	1	0	1	1	1	1	0	1	1	1	1	0	1	0	0	0	0	0	0	0	0	0	0	0	14	0	14
Cumulative Total	5	8	6	6	4	0	1	1	3	3	2	3	3	3	3	2	3	2	2	2	2	2	2	2	2	2	2	2	58	20	78



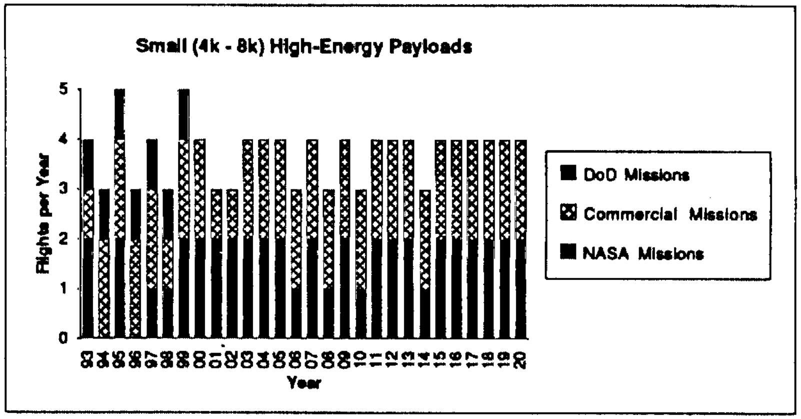
	93	94	95	96	97	98	99	00	01	02	03	04	05	06	07	08	09	10	11	12	13	14	15	16	17	18	19	20	1993 - 2010	2010 - 2020	1993 - 2020
NASA Total	0	0	0	0	0	1	0	1	0	1	1	0	1	0	1	1	0	1	0	1	0	1	0	1	0	1	0	1	8	5	13
Commercial Total	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
DoD Total	1	1	3	4	6	8	9	8	4	2	2	3	3	3	7	7	10	8	2	3	2	3	2	3	6	8	9	8	89	46	135
Cumulative Total	1	1	3	4	6	9	9	9	4	3	3	3	4	3	8	8	10	9	2	4	2	4	2	4	6	9	9	9	97	51	148



NAME	DELTA V	Year	93	94	95	96	97	98	99	00	01	02	03	04	05	06	07	08	09	10	11	12	13	14	15	16	17	18	19	20	1993 - 2010	2010 - 2020	1993 - 2020
GTC	35000*										1																				1	0	1
MSR	34768														2																2	0	2
GENERIC DS-2	34768																				1					1					0	2	2
GENERIC EO-2	36250																							1							0	1	1
PLUTO FLYBY	34768							1	1																						2	0	2
CNM	34768											1																			1	0	1
GEOPLAT	38267														1																1	0	1
GENERIC DS-3	34768																					1			1			1			0	3	3
	NASA Total		0	0	0	0	0	1	1	0	1	1	0	0	3	0	0	0	0	0	1	1	0	1	1	1	0	1	0	0	7	6	13
	Commercial Total		0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
	DoD Total		5	3	9	8	7	11	14	10	10	4	5	2	5	6	7	9	10	12	11	3	4	1	7	9	9	11	10	12	137	77	214
	Cumulative Total		5	3	9	8	7	12	15	10	11	5	5	2	8	6	7	9	10	12	12	4	4	2	8	10	9	12	10	12	144	83	227

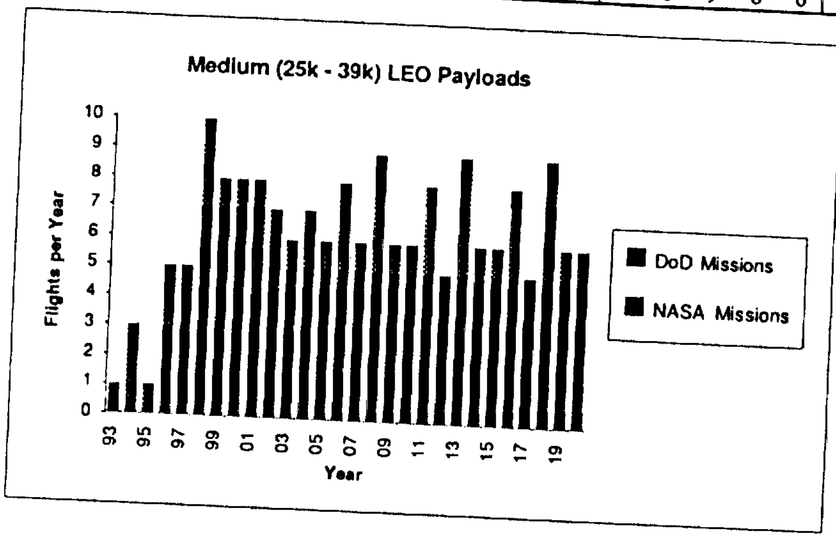


NAME	DELTA V	93	94	95	96	97	98	99	00	01	02	03	04	05	06	07	08	09	10	11	12	13	14	15	16	17	18	19	20	1993 - 2010	2010 - 2020	TOTAL
SOHO	34768			1																										1	0	1
ARTEMIS	34768							1	1	1	1	1	1	1	1	1	1	1	1											12	0	12
TDRS II	38267							1	1	1	1	1	1	1		1		1												9	0	9
GENERIC GEO	38267																			1	1	1		1	1	1	1	1	1	0	9	9
GENERIC LUN-2	34768																			1	1	1	1	1	1	1	1	1	1	0	10	10
CASSINI	34768						1																							1	0	1
ACTS	38267	1																												1	0	1
TDRS	38267	1		1		1																								3	0	3
	NASA Total	2	0	2	0	1	1	2	2	2	2	2	2	2	1	2	1	2	1	2	2	2	1	2	2	2	2	2	2	27	19	NASA Total
	Commercial Total	2	7	6	5	5	3	5	5	3	3	5	5	5	5	5	5	5	5	5	5	5	5	5	5	5	5	5	5	84	50	Commercial Total
	35% of commercia	0.7	2.5	2.1	1.8	1.8	1.1	1.8	1.8	1.1	1.1	1.8	1.8	1.8	1.8	1.8	1.8	1.8	1.8	1.8	1.8	1.8	1.8	1.8	1.8	1.8	1.8	1.8	1.8	29.4		
	35% of Commerci	1	2	2	2	2	1	2	2	1	1	2	2	2	2	2	2	2	2	2	2	2	2	2	2	2	2	2	2	32		
	DoD Total	1	1	1	1	1	1	1	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	7	0	DoD Total
	Cumulative Total	4	3	5	3	4	3	5	4	3	3	4	4	4	3	4	3	4	3	4	4	4	3	4	4	4	4	4	4	179.4	69	Cumulative Total



LEO - Medium (25k - 39k)

	93	94	95	96	97	98	99	00	01	02	03	04	05	06	07	08	09	10	11	12	13	14	15	16	17	18	19	20	1993 - 2010	2010 - 2020	1993 - 2020
NASA Total	0	0	0	3	4	7	7	7	5	5	5	5	5	5	5	5	5	5	5	5	5	5	5	5	5	5	5	5	78	50	128
Commercial Total	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
DoD Total	1	3	1	2	1	3	1	1	3	2	1	2	1	3	1	4	1	1	3	0	4	1	1	3	0	4	1	1	32	18	50
Cumulative Total	1	3	1	5	5	10	8	8	8	7	6	7	6	8	6	9	6	6	8	5	9	6	6	8	5	9	6	6	110	68	178



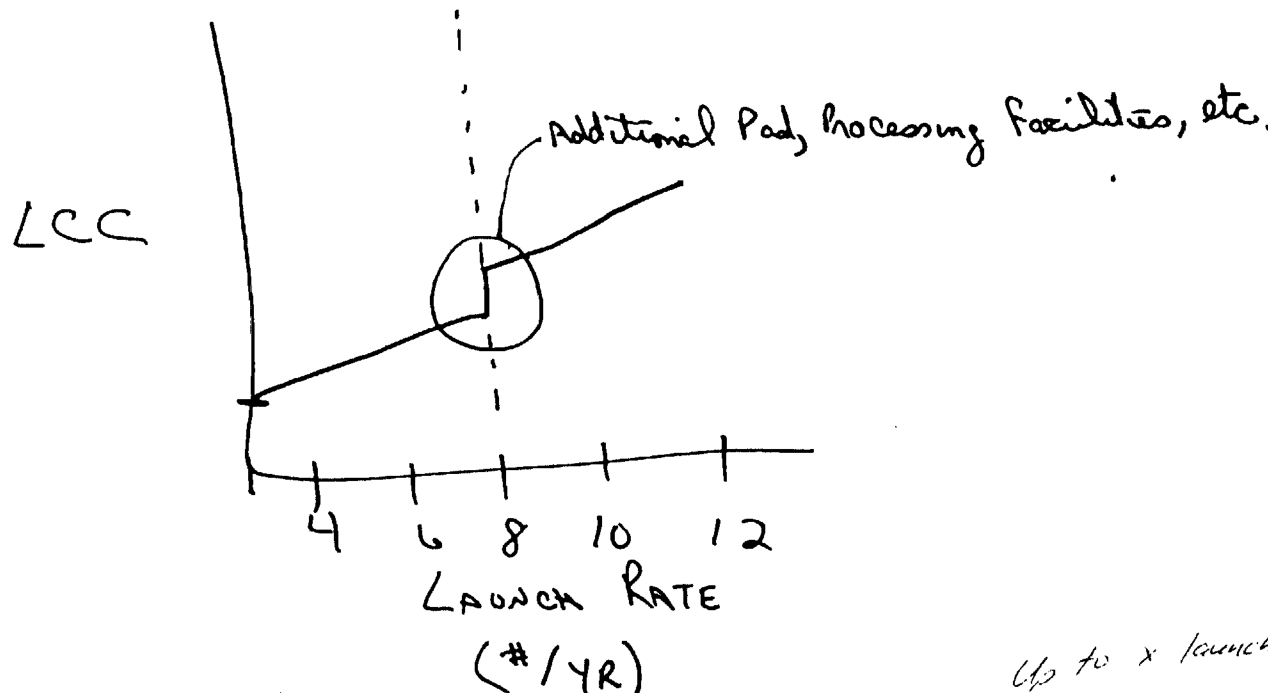
Choices of launch sites were limited to ETR & WTR.

Sufficient polar flights inclination > 57° with high energy requirements are planned to justify launch compatability with WTR requirements.

The remaining portions (majority) of the high energy market would be addressed via the ETR.

TD16-006
Upper Stage Technical Requirements Document
Source Analysis for Paragraph:
1.5b Launch Rate

1.5 b LAUNCH RATE



Assumption
This applies to "small" upper stage. Up to x launches using Centaur [illegible] Facility
Probably have a chart for [illegible] upper stage. [illegible] a few for the LVs
[illegible].

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TD16-007
Upper Stage Technical Requirements Document
Source Analysis for Paragraph:
1.6a Reliability - Mission Success

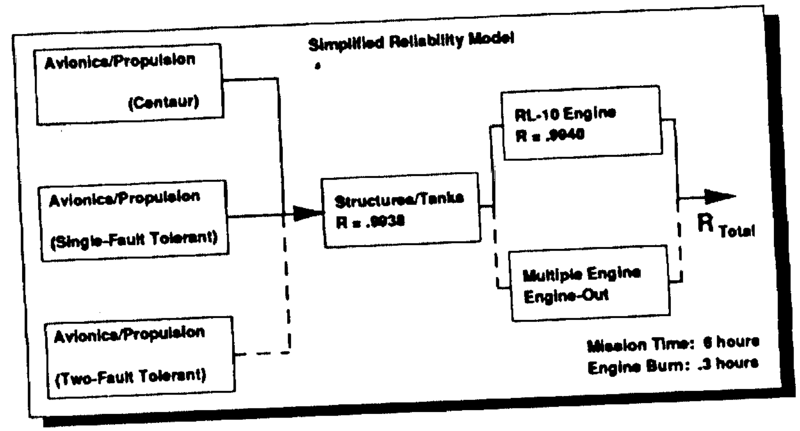
16a

US - Reliability Requirements Analysis

MSFC

Configurations:

Baseline Vehicle - Single Engine Centaur (Zero-Fault Tolerant)



Configuration Options:

1) High Reliability Avionics

2) Single-Fault Tolerant Avionics

3) Dual-Fault Tolerant Avionics

4) Single-Fault Tolerant Avionics w/Multiple Engine/Engine-out Capability

5) All Subsystems Single-Fault Tolerant

Probability of Mission Success Reliability Requirement is .98

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RW930329-01A

16a cont.

US - Reliability Requirements Analysis

MSFC

- Upper Stage System Level Reliability Predictions
- Baseline and All Options Meet Reliability Requirement of .98

Subsystem	Baseline	Option 1	Option 2	Option 3	Option 4	Option 5
Avionics/RCS (Mission/Mission Loss)	.9932 (147)	.9950 (200)	.9984 (625)	.9999 (10,000)	.9984 (625)	.9984 (625)
Primary Propulsion	.9940 (166)	—	—	—	.9999 (10,000)	.9999 (10,000)
Structures/Tanks Thermal/Ordnance	.9938 (161)	—	—	—	—	.9982 (556)
Single String Subsystems have Greatest Influence on Total Mission Reliability Results						
Mission Reliability	.9811	.9829	.9863	.9877	.9922	.9965
Missions per Mission Loss	53	58	73	82	128	286

- Engine Reliability Numbers Based Upon Demonstrated RL-10 Reliability as of 11/91 with 90% Confidence
- Missions per Mission Loss = 1/(1-R)

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RW930329-02A

TD16-008
Upper Stage Technical Requirements Document
Source Analysis for Paragraph:
1.6b Reliability - Fault Tolerance

Persuit of AF Launch services market appears to require "No single point failures" may cause loss of mission" approach.

NASA market is more concentrated on fault tolerance of safety critical concerns; "Fail safe", "Fail Op/fail safe" type criteria.

USRS SRD 6.6.1 was taken as the more-stringent requirement

TD16-009
Upper Stage Technical Requirements Document
Source Analysis for Paragraph:
1.7a Facilities - Coordination and Design

1.7a

Operations Technologies at the USTC

MSFC

- Improved Cryogenic Propellant Loading System
  - USTC can be used as a simulation base for the NGUS to verify the manpower and timeline reductions as well as the rapid loading and unloading of propellants for contingency operations.

- Automated Launch Operations Management
  - Ideally the Launch Ops management tool would include the launch vehicle, payload, and upper stage. The USTC could be used to develop and verify the upper stage portion of the tool since it will act as an integration point for the NGUS subsystems.

- Improved Mechanical AGE
  - Depending on the extent of BIT/VHM used in the vehicle, improved mechanical AGE can help to reduce the overall processing timeline. The USTC can be used to enhance the design of the AGE by providing simulation capabilities to define the requirements and to verify the operational procedures by using the breadboard/brassboard upper stage subsystems in actual practice of a launch operations process.

/ 7a

Operations Technologies at the USTC (cont)

MSFC

- Improved Electrical AGE (may be redundant with VHM)
  - Depending on the extent of BIT/VHM used in the vehicle, improved electrical AGE can help to reduce the overall processing timeline. The USTC can be used to enhance the design of the AGE by providing simulation capabilities to define the requirements and to verify the operational procedures by using the breadboard/brassboard upper stage subsystems in actual practice of a launch operations process

- Automated Payload Integration System
  - An automated payload integration system would be used to automate the mission management and engineering functions required to verify that the interfaces and environments are within the acceptable limits. The USTC can be used to develop the specific requirements for the NGUS and payload and also used to verify and troubleshoot procedures for ~~use in~~ the actual launch processing.

- Electromechanical Actuators
  - The USTC role in the use of the EMAs can be twofold. First, it can be used to integrate the EMAs with the rest of the NGUS subsystems during the development cycle and verify the expected flight performance. Second, it can be used to develop and verify the reductions expected in the manpower requirements and the processing timelines.

1.7a FACILITIES

- GATHERING DATA ON CENTAUR PROCESSING FACILITY
- SURVEY OF EXISTING FACILITIES WITH "LARGE" CAPABILITIES
- TECHNOLOGIES AT USTC FOR DEM/VAL

1.14a OPERABILITY: IEL

- DEPENDENT ON FACILITIES USED
- GENERIC TASKS DESCRIBED
- TIMELINES + MANPOWER TO BE GENERATED

TD16-010
Upper Stage Technical Requirements Document
Source Analysis for Paragraph:
1.8a Environments - Orbital Debris

Requirements Analysis Task Plan

LYLE BAREISS

Requirement: 1.8a The upper stage shall be designed to operate in and survive the environments described in RECON 89N22638 "Orbital Debris Environment for Spacecraft Design to Operate in Low Earth Orbit - NASA TM 100471, Sept. 1, 1988", NASA-SP-8030 "Meteoroid Environment Model, 1970 - Interplanetary and Planetary. NASA Space Vehicle Design Criteria Environment. Oct., 1970", and EXPO-T2-920021-EXPO, "Lunar Engineering Models: General and Site-Specific Data". (FLO PRD Vol 1 #813, #814, #815)

Responsible Individual/Supporting Individual(s): Bob Spencer

? LEVEL OF DETAIL !

Summary of Approach:

- Generate Environments database for each mission profile.
- Compile list of candidate materials as shielding options.
  - Candidate material
  - Candidate construction techniques
- Analyze different shield options against Environments for vehicle mass impacts.
- Determine cost per pound of different shield options

Task Description:

Analyze different shield options in differnet environments for mass to dose sensativity. Determine cost per pound of different shield options

Interfaces:
- Environment Engineering Specialists
- Cost analyst

3-2D ...

Inputs
- Derived Environments
- Mission Profiles & Timelines
- Candidate material for shielding
- Identify different shielding applications per vehicle.

Outputs
- Graphical summary of shielding mass vs dose and time
- Graphical summary of Cost per po...
- L...
loc...

NEEDS: UNFA...
1.) LIST OF TYPI... MAT. + THICK... + DENSITIES
2.) CONST...

1.8a

CHARTS
2-23 CONTAM.
...T/TIME

Schedule

	March		April		May
Program Milestones	Δ Start TD 8	Δ 17 Prelim Concept Description	Δ 12 Internal Progress Review		
Task Milestones					

ENVIRONMENTS:
1.) ATOMIC OXYGEN
2.) MICROMETEROID
3.) DEBRIS
4.) UV

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Req. 1.8a Envir. Analysis - Lessons Learned

MSFC

- A Contamination Free System is Not Possible With Today"s Technology
  - Adverse Environments DO Affect System Instrument Performance
- Analysis Must Be Worked From Systems Viewpoint Crossing All Interfaces
  - Must Be Addressed Early In Any Space System Program
  - Should Be Treated as Major Design/Systems Discipline (e.g. Thermal, Structures, Power, etc.)
- Must First Be Able to Quantify Environments (i.e. Contamination, Atomic Oxygen, Radiation, etc.) Before Attempting to Control Their Effects
- System Contamination Can Be Minimized by Proper Control of:
  - Materials
  - Engines/Vents
  - Geometry (Viewing)
  - Ground Facility Operations
  - Mission Operations

Note: Data From Systems Engineering Course D-1A214 / EN214 "Engineering Specialties"- Section "Non-Nuclear Survivability" By Lyle Bareiss

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Req. 1.8a Environments Analysis

Compliance With Environment Documents:

RECON 89N22638 "Orbital Debris Environments for Spacecraft Design to Operate in Low Earth Orbit - NASA TM 100471, Sept. 1,1988"

NASA-SP-8030 "Meteoroid Environment Model, 1970 - Interplanetary and Planetary. NASA Space Vehicle Design Criteria Environment. Oct.,1970"

EXPO-T2-920021-EXPO "Lunar Engineering Models: General and Site-Specific Data."

Potential Environments:

Environment	Possible Effects Mitigation
• Atomic Oxygen	TBD
• Micro Meteoroid	TBD
• Debris	TBD
• Ultra Violet	TBD
• Thermal Cycling	TBD

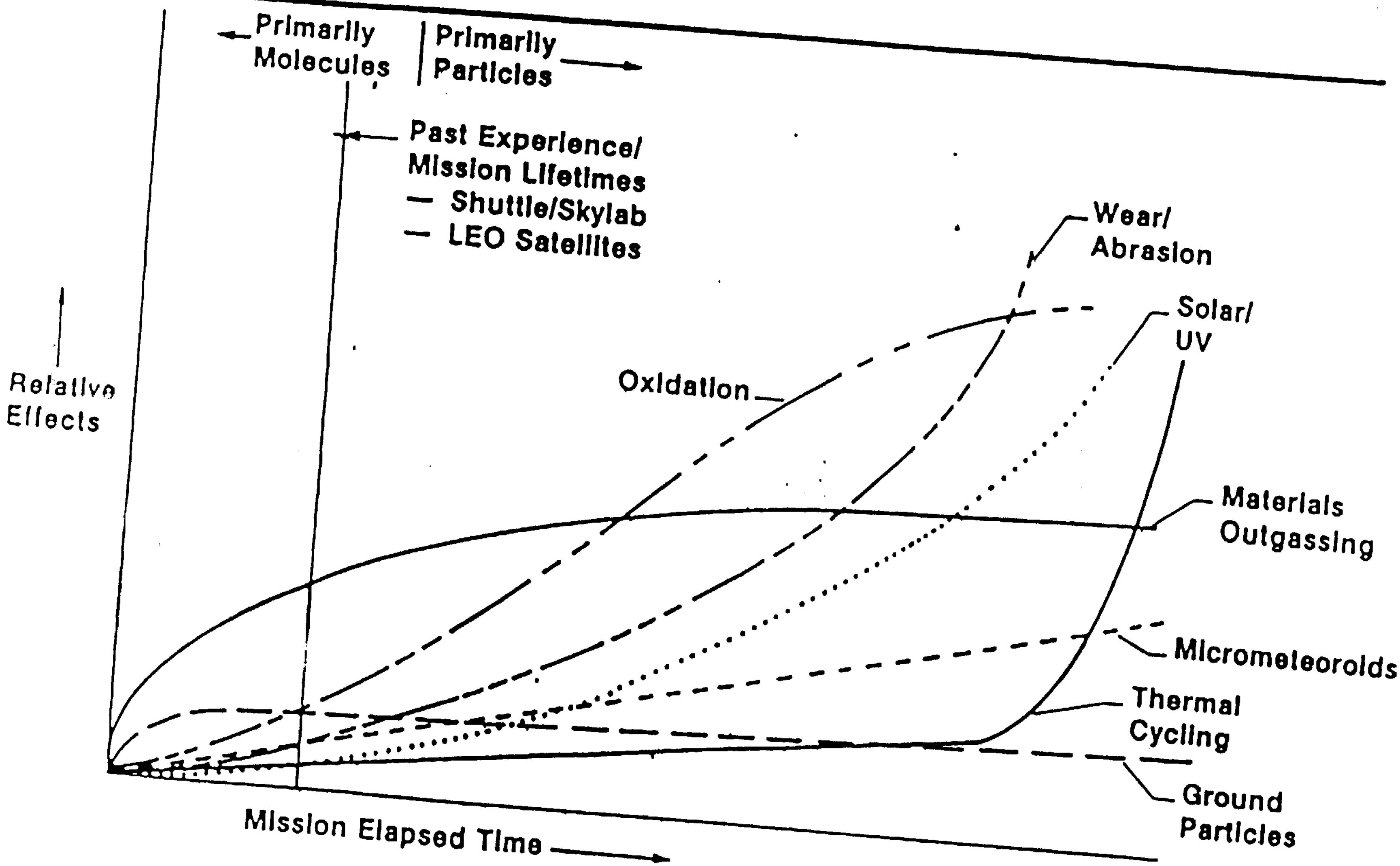
Req. 1.8a Environments Analysis - Products

MSFC

Desired Products:

- List of Candidate Materials Used as typical Shielding For Different Environments Encountered in Space
- Material Properties for Those Materials Listed Above. (i.e. Density, Thickness, etc.)
- Typical Construction Techniques Used to Protect Against Identified Environments
- Cost per Pound of The Materials Used for Protection - To Assess Variation Impacts to the Baseline Vehicle Configuration

New Challenges – Long Term Satellite Contamination



MARTIN MARIETTA

Interoffice Memo

MARTIN MARIETTA

Post-it™ brand fax transmittal memo 7671	# of pages ▸ 4
To Bob Spencer	From Rick Hjelm
Co. 7-8150	Co.
Dept.	Phone # 1-9131
Fax # 7-7031	Fax # 1-9085

DATE: 21 May 1993

TO: Bob Spencer

cc: Lyle Bareiss

FROM: Rick Hjelm (x1-9131)

SUBJECT: STV Micrometeoroid and Space Debris Penetration Vulnerability Assessment

---

The objective of this analysis was to perform a first-cut assessment of the Space Transfer Vehicle (STV) vulnerability to penetration from micrometeoroids and space debris. (Note that this memo completely supersedes the previous memo addressed to Bob Spencer dated 14 May 1993.)

The analysis approach was as follows:

1) Select worst case mission from the eight reference missions;

2) Compute minimum particle diameter to penetrate tank skin for four selected tank material layups;

3) Compute micrometeoroid and space debris fluxes of particles of diameter greater than the minimum diameter to penetrate; and,

4) Compute the probability of no penetration for the selected STV tank material layups, for each of the two STV design options (exposed areas).

Because this was a minimum effort, first-cut analysis a number of simplifying assumptions were employed. These were as follows:

- All impacts were normal to the surface.
- Space debris and micrometeoroid fluxes were isotropic.
- Space debris velocity for all particles was 10 km/sec.
- Space debris particle density was 2.8 gm/cm^3.

- Micrometeoroid velocity for all particles was 20 km/sec.
- Micrometeoroid particle density was 2.0 gm/cm^3.

The environment models used were those employed by NASA to determine the Space Station micrometeoroid and space debris environments. These are described in NASA document SSP 30425 Revision A and NASA TM 100 471.

Four STV tank material layups were evaluated. The first was simply a single aluminum layer, 0.040 in thick. The second was the same 0.040 in thick aluminum layer, surrounded by 0.375 in of spray-on-foam-insulation (SOFI), in turn surrounded by 0.100 in of multi-layer insulation (MLI). The third layup was the 0.040 in thick aluminum layer surrounded by 0.475 in of kevlar. The final layup considered was the same 0.040 in aluminum layer surrounded by a 0.010 in aluminum layer, stood off by 1 in.

The penetration equation for single layer metal targets, developed by NASA during the Apollo program, was used for the first configuration. Quick, easy-to-use penetration analysis techniques for multiple layers of different materials do not exist. So to assess the second configuration, the MLI layer was assumed to act as an optimum bumper with a 0.375 in spacing between it and the 0.040 in aluminum rear sheet. The rear sheet design equation for optimum bumpers, also developed by NASA, could then be used to determine the minimum particle size to penetrate the bumper-rear sheet configurations. This approach does not account for any bumper material properties and therefore assumes that MLI is as effective a bumper as any other material. This should be reasonable, considering the very high impact velocities. This approach also ignores any benefit from the SOFI layer, and should result in a conservative assessment. Explicit analysis of the third layup was also untenable, so the same assumption, that the kevlar acts as an optimum bumper, was made. Because bumper material properties were not considered in this approach, configurations 2 and 3 were predicted to have the same penetration protection effectiveness because they have the same effective spacing. The fourth and final configuration also used the bumper-rear sheet approach with a 1 in spacing between the two aluminum layers.

Both STV design options were evaluated. The relevant difference between the two, for this analysis, is the surface area. The surface area used to determine penetration probability was that of the exposed tank skin only (the tank area beneath intertank structure was not included). A payload was assumed to be atop the STV, thus providing shielding for the top surface. The exposed surface areas used to determine the total number of penetrations of the vehicle were 80.9 sq meters for Option 1 and 414 sq meters for Option 2.

The worst case mission was determined to be Number 3 - Sun-Synchronous Orbit. The mission parameters were 900 km circular orbit, 99° inclination and 12 hour duration. This mission maximizes the space debris flux and despite not having the longest duration will result in the worst case environment.

Table 1 summarizes the results. The results clearly show the positive effect of additional material surrounding the aluminum tank wall. The addition of SOFI and MLI or kevlar significantly increased the probability of no penetration over the single aluminum layer, and the aluminum bumper provided the best protection of the four configurations analyzed. The difference in results between the second and third configurations and the fourth is strictly due to the greater spacing assumed for configuration 4. The analytical approach did not consider bumper material properties; it only assumed that they would be equally effective at vaporizing the projectile. Comparing the results for Option 1 and Option 2 illustrates how exposed area impacts the results. The larger target is much more susceptible to penetration.

Table 1 Summary of Analysis Results

	Minimum Particle Diameter to Penetrate (cm)		Probability of No Penetrations	
Configuration Analyzed	Space Debris	Meteoroids	Option 1	Option 2
1. 0.040 in Al	0.0293	0.0217	0.924	0.668
2. 0.040 in Al 0.375 in SOFI 0.100 in MLI	0.0531	0.0472	0.986	0.928
3. 0.040 in Al 0.375 in kevlar	0.0531	0.0472	0.986	0.928
4. 0.040 in Al 1.0 in Space 0.010 in Al	0.103	0.914	0.997	0.987

Table 2 provides some material properties and the weight impact of the additional materials surrounding the exposed tank area for the four material layups analyzed. These properties are provided for use in a system level evaluation.

For more information regarding this subject, please contact Rick Hjelm (x1-9131) or Lyle Bareiss (x1-9108)

Table 2 Material Properties and Weights

Configuration Analyzed	Density (gm/cm^3)	Total Mass to Cover Exposed Area (kg)	
		Option 1	Option 2
1. 0.040 in Al	2.7	--	--
2. 0.040 in Al	2.7	--	--
0.375 in SOFI	0.035	27	140
0.100 in MLI	0.045	9.2	47
3. 0.040 in Al	2.7	--	--
0.375 in kevlar	0.9	880	4500
4. 0.040 in Al	2.7	--	--
1.0 in Space	--	--	--
0.010 in Al	2.7	55	280

TD16-011
Upper Stage Technical Requirements Document
Source Analysis for Paragraph:
1.8b Environments - Launch Vehicle Acceleration

Requirements Analysis Task Plan

Requirement: 1.8b The upper stage must be designed to withstand the launch system acceleration of 4-6 G's (TBR)

Responsible Individual/Supporting Individual(s): Bob Spencer

Summary of Approach:

- Obtain results from Req. 1.14d to determine launch environments
- Identify pertinent acceleration loads for upper stages
- Categorize payload groupings for upper stage analysis
- Separate out the primary structural elements of the upper stage configuration (s) for analysis
- Analyze configurations against the acceleration loads for the given payloads
- Summarize sensitivity data

Task Description:

Analyze the effects of specific payload mass's at varying Acceleration levels on the dry mass of the upper stage.

Interfaces: N/A

Inputs

- Inputs from Req. 1.14d
- Launch Environments
- Mission Profiles & Timelines
- Candidate payload mass's
- Identify primary structure elements of upper stage

Outputs

- Graphical summary of acceleration effects on upper stage dry mass
- Data base of launch environments

Schedule

	March	April	May	June
Program Milestones	Δ Start TD 8; Δ Prelim Concept Description 17	Δ Internal Progress Review 12	Δ End TD 31	
Task Milestones				

The 4–6 G's. ~~[illegible]~~ is the figure used for compatibility with the Titan IV Launch vehicle, taken as the most stringent case

TD16-012
Upper Stage Technical Requirements Document
Source Analysis for Paragraph:
1.8c Environments - Maximum Allowed Acceleration

Launch Vehicle accelerations were determined to be the highest acceleration loads which would be seen by the upper stage. Transportation & handling loads should be far lower.

The 4-6 g figure does not take into account vibration or pyro shock transient loads, which can by rule of thumb be assumed to double the axial g. max. loads, to 8-12 g.

Interoffice Memo

MARTIN MARIETTA

Refer To: DYN-92-102

Date: 15 JULY ~~2 June~~ 1992

To: Bob Spencer

cc:

From: Tim Gasparrini

Subject: Preliminary STV Loads Based On Saturn V Data

Preliminary STV Avionics module loads have been derived based on Saturn V test flight data for AS-501 and AS-502 (Apollo 5 and 6). The S-IVB measured acceleration and acoustic data from Chapters 9 and 16 of MPR-SAT-FE-68-3 (Saturn V Launch Vehicle Flight Evaluation Report-AS-502 Apollo 6 Mission) was used a the data input to the loads derivation. Design Load Factors for the avionics module structure as well as flight level random vibration environments for the avionics boxes were derived. This memo presents these loads and summarizes any assumptions made.

Design Load Factors

Measured acceleration data for the S-IVB was used to compute the design load factors. The accelerometer locations shown in Table 1 were used as a database for the acceleration data from which the load factors were computed. These locations were selected as being representative of the STV avionics platform and are shown in Figure 9-27 of Appendix A. The maximum envelope of the measured peak acceleration was used for the design load factors. In the document, peak measured accelerations are 1.4 times the Grms accelerations.

Table 1 - Design Load Factors For STV Avionics Module.

Location	Thrust(g) - GRMS	Radial(g) - GRMS
Sequencer Panel	5.9	7.8
Switch Selector Panel	4.8	6.6
APS Aft Attach	2.8	5.7
Thrust Structure	1.8	3.0
Engine Gimbal	4.5	6.8
Field Splice I	6.0	8.9
Field Splice II	4.4	9.1
Maximum (GRMS)	6.0	9.1
Design Load Factors	8.4	12.7

Avionics Random Vibration Environment

The acceleration envelope time histories used above represent the vibration as a fun ction of time from 50 to 3000hz. A review of the data for the aft components indicates that the vibration levels are greatest during liftoff and maximum dynamic pressure (MaxQ). This implies that the acoustically generated vibration overshadows the mechanically transmitted vibration during J-2 engine start. Based on this data, the liftoff and MaxQ acoustic environments were used to derive random vibration environments for the avionics mounted to the avionics platform. Measurements were used from the S-IVB Aft Skirt and the S-II Forward Skirt. The external liftoff measurements from S-IVB and S-II were averaged to form the external liftoff acoustic environment and the external MaxQ measurements from S-IVB and S-II were averaged to form the external MaxQ acoustic environment (Appendix A presents these measurements). The envelope of these environments as a function of frequency was used to form the "STV external acoustic environment". The external environment was reduced by 3 dB to account for transmission losses through the skin. This reduced external environment is defined as the STV internal acoustic environment. These environments are shown in Table 2.

Table 2 - STV Derived Internal And External Acoustic Environments

Center FREQ(HZ)	External SPL(dB)	Internal SPL(dB)
25	133.634	130.6340012
31.5	134.633	131.6329499
40	135.638	132.6375996
50	138.607	135.6066953
63	137.643	134.6432499
80	140.624	137.6242278
100	140.098	137.098075
125	139.124	136.1237046
160	140.182	137.1817419
200	140.627	137.627304
250	140.384	137.3840084
315	139.383	136.3829615
400	138.638	135.6376151
500	136.607	133.6067218
630	137.393	134.3932725
800	137.124	134.1242576
1000	136.598	133.5981049
1250	136.124	133.1237334
1600	134.682	131.6817761
2000	134.127	131.1273405
2500	132.134	129.1340488
OASPL	151.1	148.1

The internal Acoustic environment was used to derive the avionics random vibration environment. The random environment was computed from the structural response of similar hardware by scaling acoustic test results with the ratio of the predicted STV acoustic level to the actual acoustic test level. These scaled responses for a number of acoustic tests were enveloped to define the random vibration environment. Figure 1 presents the scaled structural response database and the STV random vibration envelope. Figure 2 presents the STV envelope as compared to some recent flight program random vibration environments for selected locations where avionics were mounted. As can be seen the STV environment is much more severe.

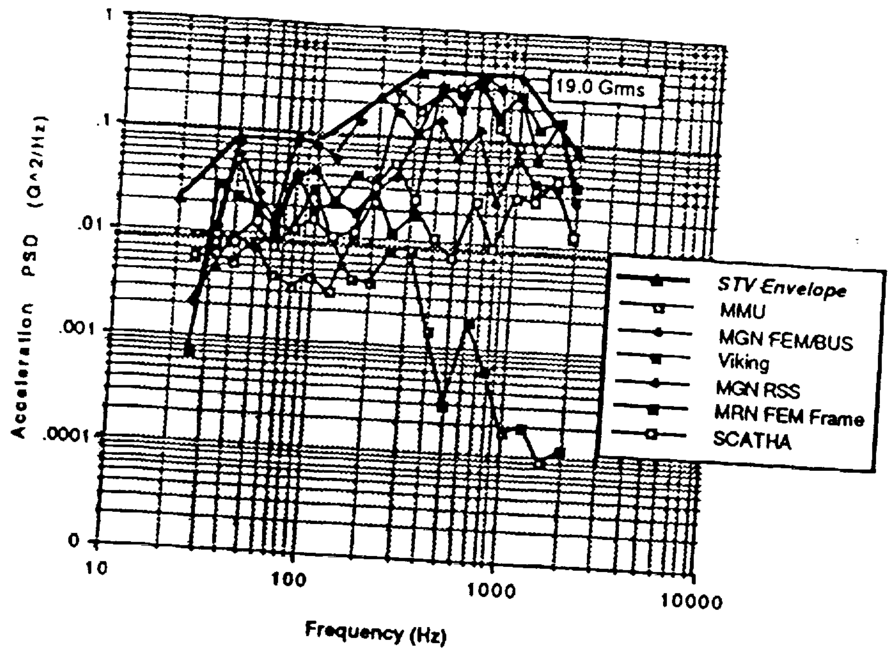


Figure 1 - STV Random Vibration Database.

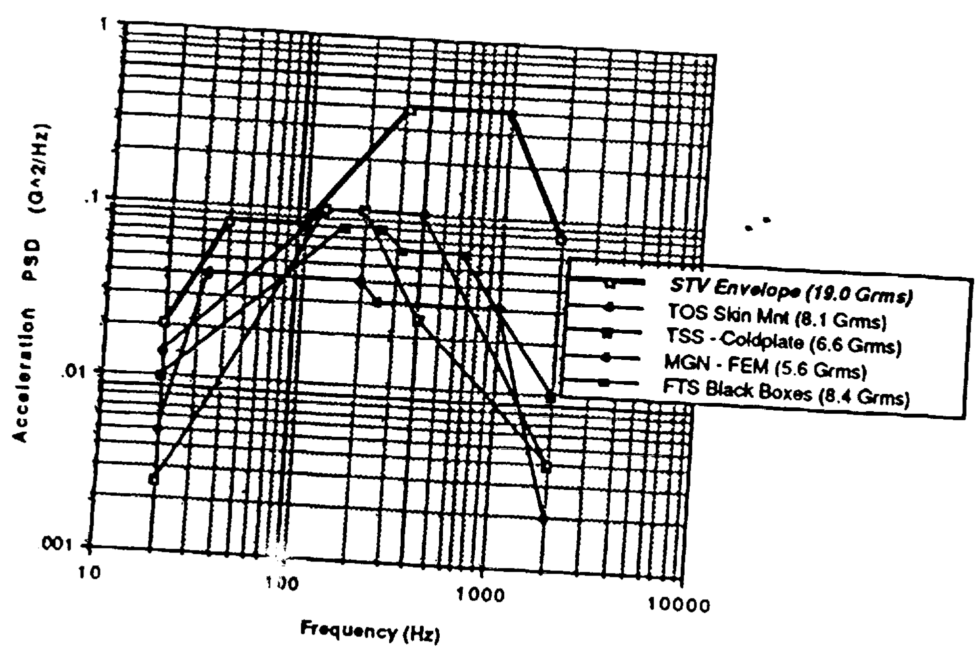


Figure 2. STV Random Vibration Compared To Recent Flight Programs.

11:37 AM 7/15/92

This preliminary avionics module environment data is intended to cover both the J-2 configuration and the SSME configuration. This data will be updated as more Saturn V data or SSME data becomes available. Any questions concerning this information can be directed to Tim Gasparrini at 7-8964.

Tim Gasparrini
EOS Dynamics

11:37 AM 7/15/92

APPENDIX A
SATURN V TEST DATA

9.3.3 S-IVB Stage and Engine Evaluation

Nine vibration measurements were made on the structure, twenty-two at components and six on the engine. Measurement locations are shown in Figure 9-27. The maximum composite (50 to 3000 hertz) vibration levels on the structure, forward components, aft components, and engine are summarized in Figure 9-28 and Table 9-4. For comparison purposes, the vibration levels are shown with measurements taken during AS-501 flight.

9.3.3.1 S-IVB Stage Structure and Components. The maximum vibration levels measured on the S-IVB structure were slightly lower on AS-502 than on AS-501. Forward component maximum vibration levels were greater on AS-502 than measured at similar locations during the AS-501 flight. The maximum vibration levels measured at the aft components were 70 percent of those measured at similar locations during the AS-501 flight.

9.3.3.2 S-IVB Stage J-2 Engine. The maximum vibration levels measured on the engine were almost identical to those measured during the first S-IVB burn of the AS-501 flight.

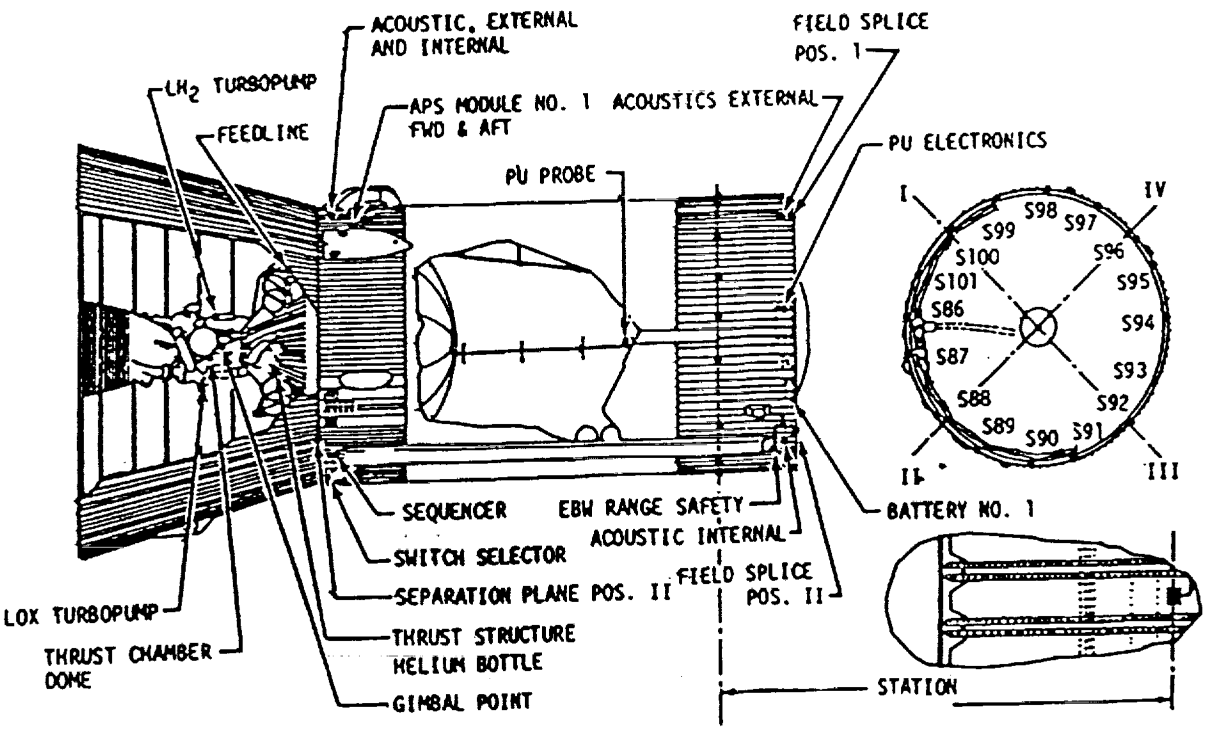


Figure 9-27. S-IVB Acoustics, Vibration and Dynamic Strain Measurements

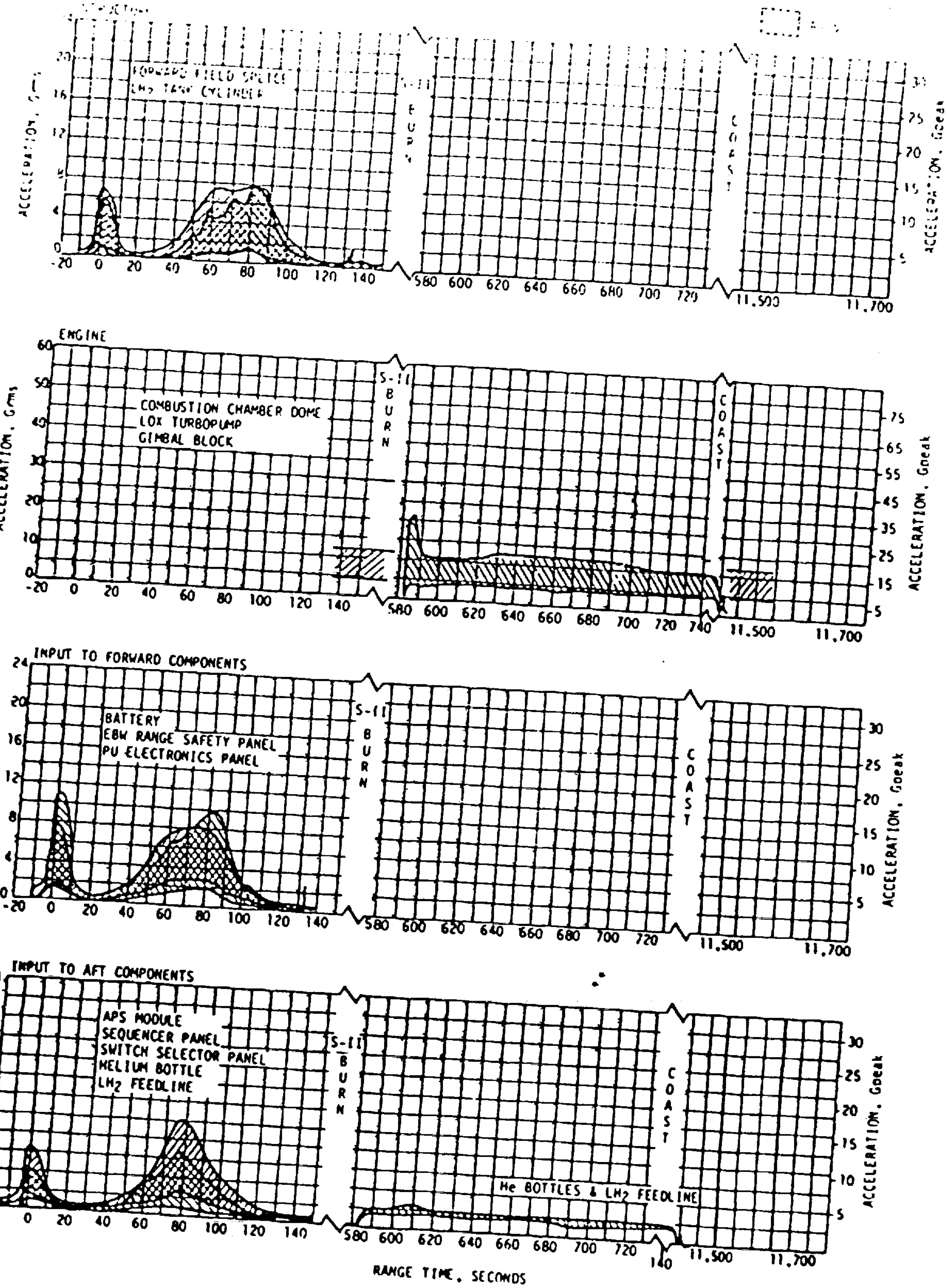


Figure 9-28. S-IVB Stage Vibration Envelopes

Table 9-4. S-IVB Vibration Summary

	AREA MONITORED	MAX LEVEL (Grms)	RANGE TIME (SEC)	REMARKS
Structures	Separation Plane, Pos II - Thrust	0.8	3	The maximum vibration due either to sound pressure at liftoff, turbulence at maximum dynamic pressure, or to J-2 engine operation.
	Field Splice Pos I (1f) - Thrust	0.7	0	
	Field Splice Pos I - Thrust	6.0	89	
	Field Splice Pos I - Pitch	7.6	89	
	Field Splice Pos I - Yaw	4.7	89	
	Field Splice Pos II - Thrust	4.4	80	
	Field Splice Pos II - Pitch	3.0	86	
	Field Splice Pos II - Yaw	8.6	83	
Component (LH_2 Tank)	LH_2 PU Probe Input - Radial	6.8	1	
Engine	Gimbal Point - Thrust	4.5	745	The maximum vibration occurred during J-2 engine start transient.
	Gimbal Point - Pitch	5.3	739	
	Gimbal Point - Yaw	4.2	739	
	Combustion Chamber, Dome - Thrust	9.1	589	
	LOX Turbopump - Lateral	2.1	586	
Component (Fwd Skirt)	PU Electronic Panel Input - Thrust	4.3	78	
	PU Electronic Panel Input - Radial	10.6	1	
	PU Electronic Panel Response - Radial	5.5	1	
	EBW Range Safety Panel Input - Thrust	3.2	80	
	EBW Range Safety Panel Input - Radial	Invalid		
	EBW Range Safety Panel Response - Radial	2.2	2	
	Battery No. 1 Input - Thrust	2.1	78	

Table 9-4. S-IVB Vibration Summary (Continued)

	AREA MONITORED	MAX LEVEL (Grms)	RANGE TIME (SEC)	REMARKS
Component (Fwd Skirt) (Cont)	Battery No. 1 Input - Radial	4.2	72	
	Battery No. 1 Input - Tangential	1.9	1	
Component (Aft Skirt)	Sequencer Panel Input - Thrust	5.9	78	Due to a loose connector.
	Sequencer Panel Input - Radial	7.8	78	
	Sequencer Panel Response - Radial	2.8	78	
	Switch Selector Panel Input - Thrust	4.8	83	
	Switch Selector Panel Input - Radial	6.6	80	
	Switch Selector Panel Response - Radial	3.5	1	
	APS Mod-1 Aft Attach Point Input - Thrust	2.8	80	
	APS Mod-1 Aft Attach Point Input - Radial	4.2	75	
	APS Mod-1 Fwd Attach Point Input - Radial	5.7	75	
Component (Thrust Structure)	He Bottle Input - Thrust	1.8	589	
	He Bottle Input - Pitch	2.2	589	
	He Bottle Input - Yaw	2.1	2	
	LH2 Feedline Input - Thrust	2.2	2	

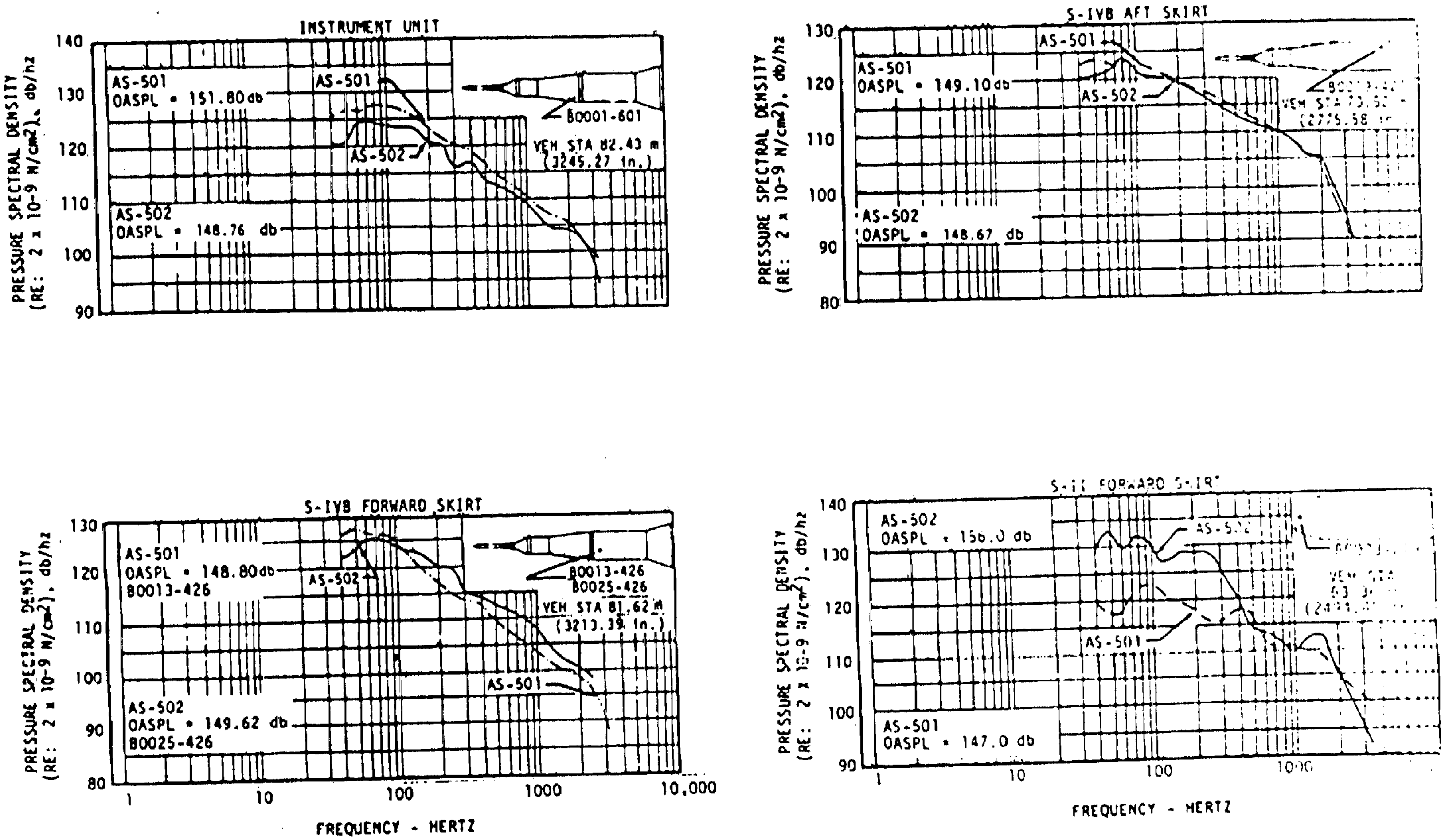


Figure 16-17. Vehicle External Sound Pressure Spectral Densities, Sheet 1 of 2

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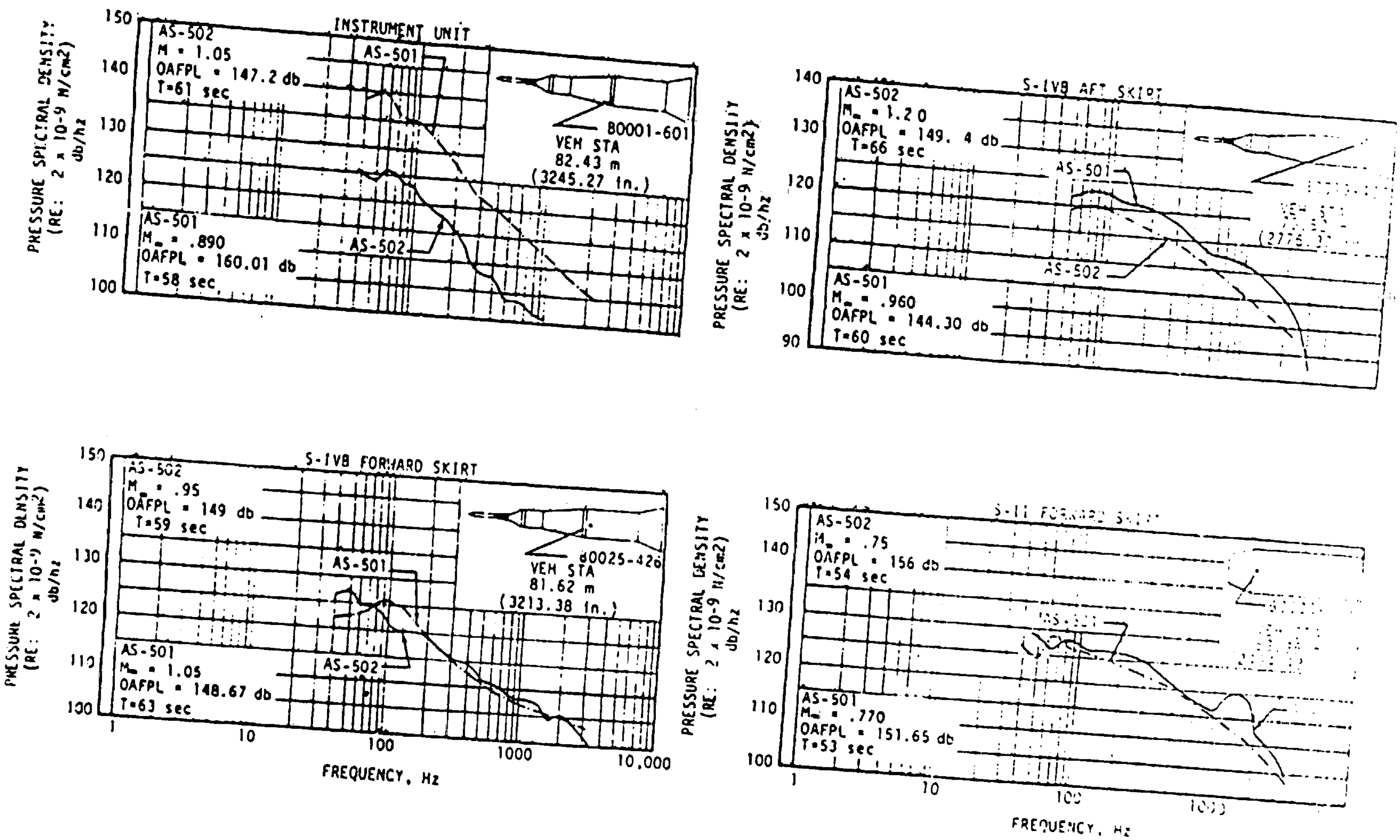

Figure 16-19. Vehicle External Fluctuating Pressure Spectral Densities, Sheet 1 of 2

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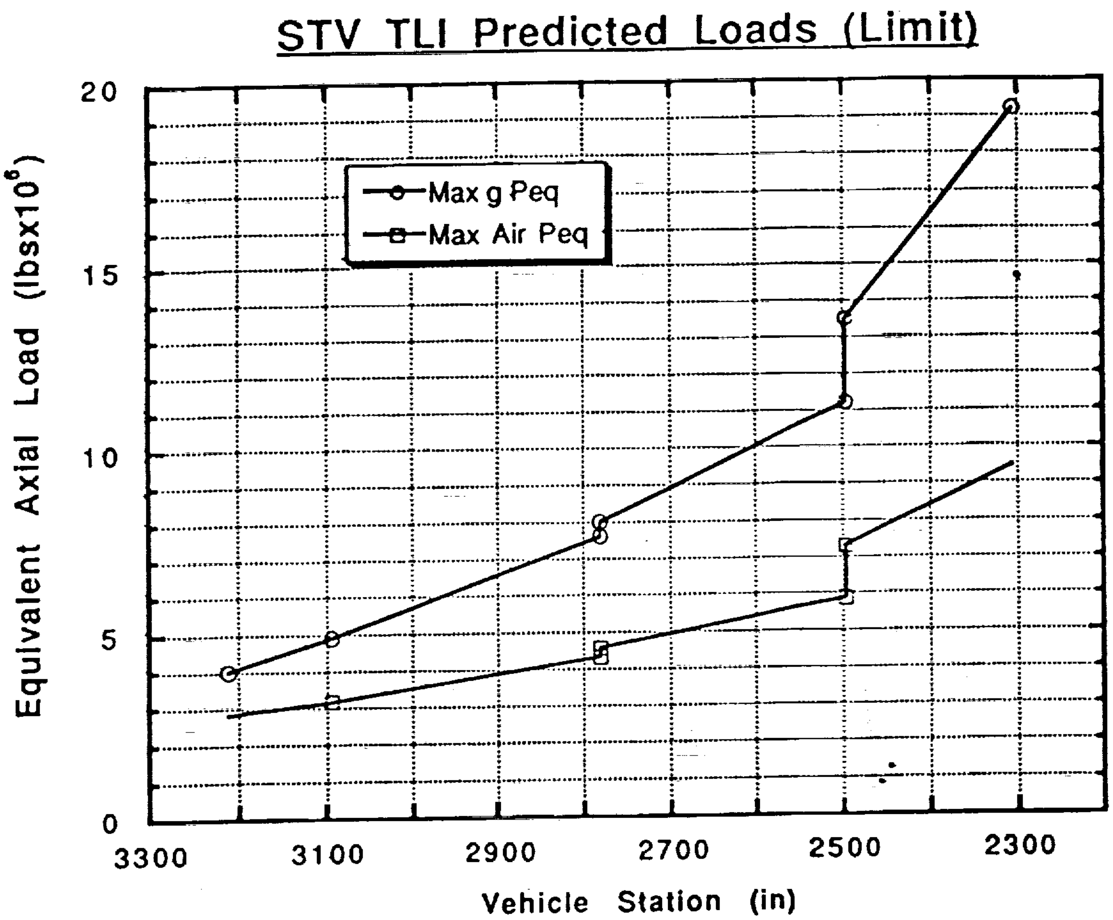
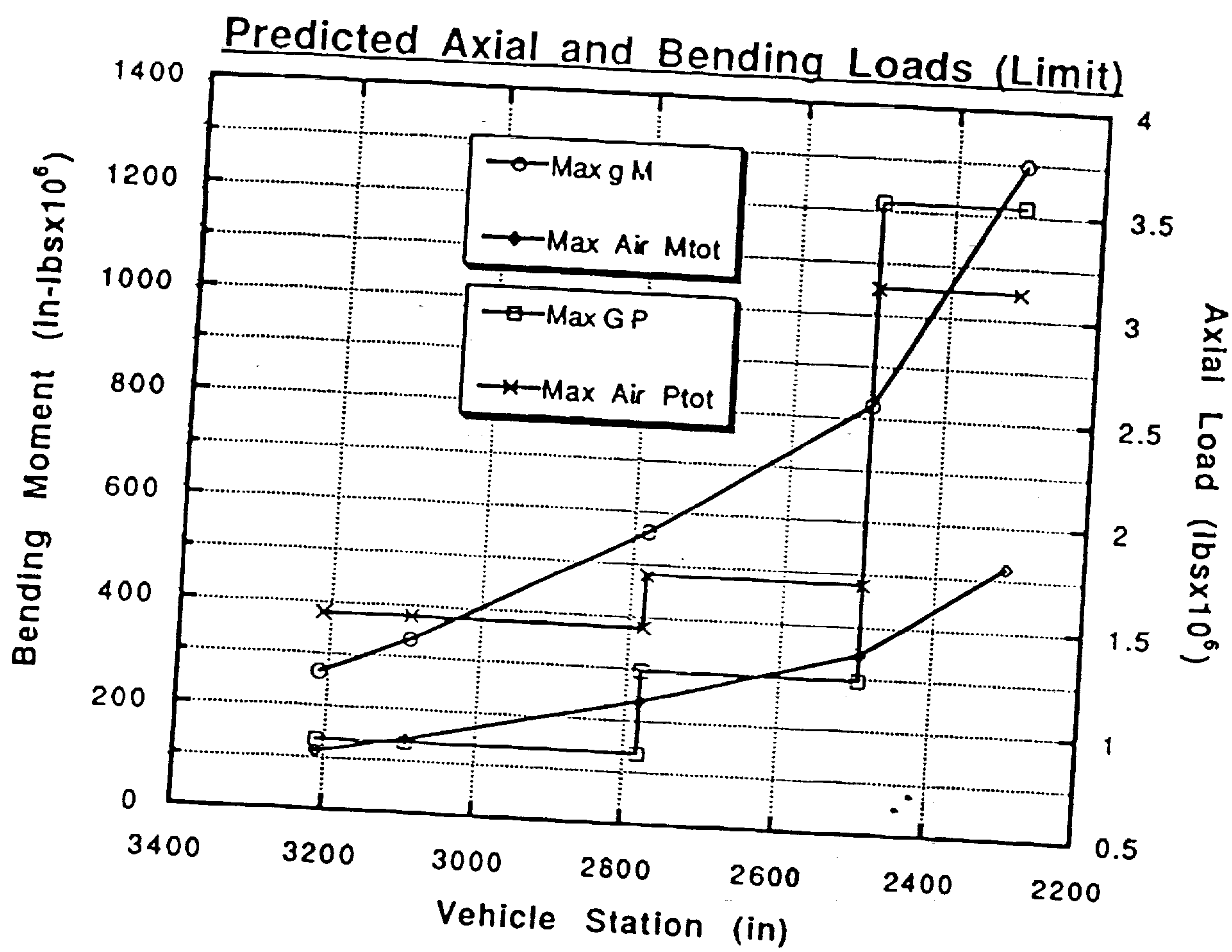


STV TLI Predicted Loads (Limit)
Max g Peq
Max Air Peq
Equivalent Axial Load (lbsx10^6)
20
15
10
5
0
3300
3100
2900
2700
2500
2300
Vehicle Station (in)



Predicted Axial and Bending Loads (Limit)
Max g M
Max Air Mtot
Max G P
Max Air Ptot
Bending Moment (In-lbsx10^6)
Axial Load (lbsx10^6)
Vehicle Station (in)
1400
1200
1000
800
600
400
200
0
4
3.5
3
2.5
2
1.5
1
0.5
3400
3200
3000
2800
2600
2400
2200

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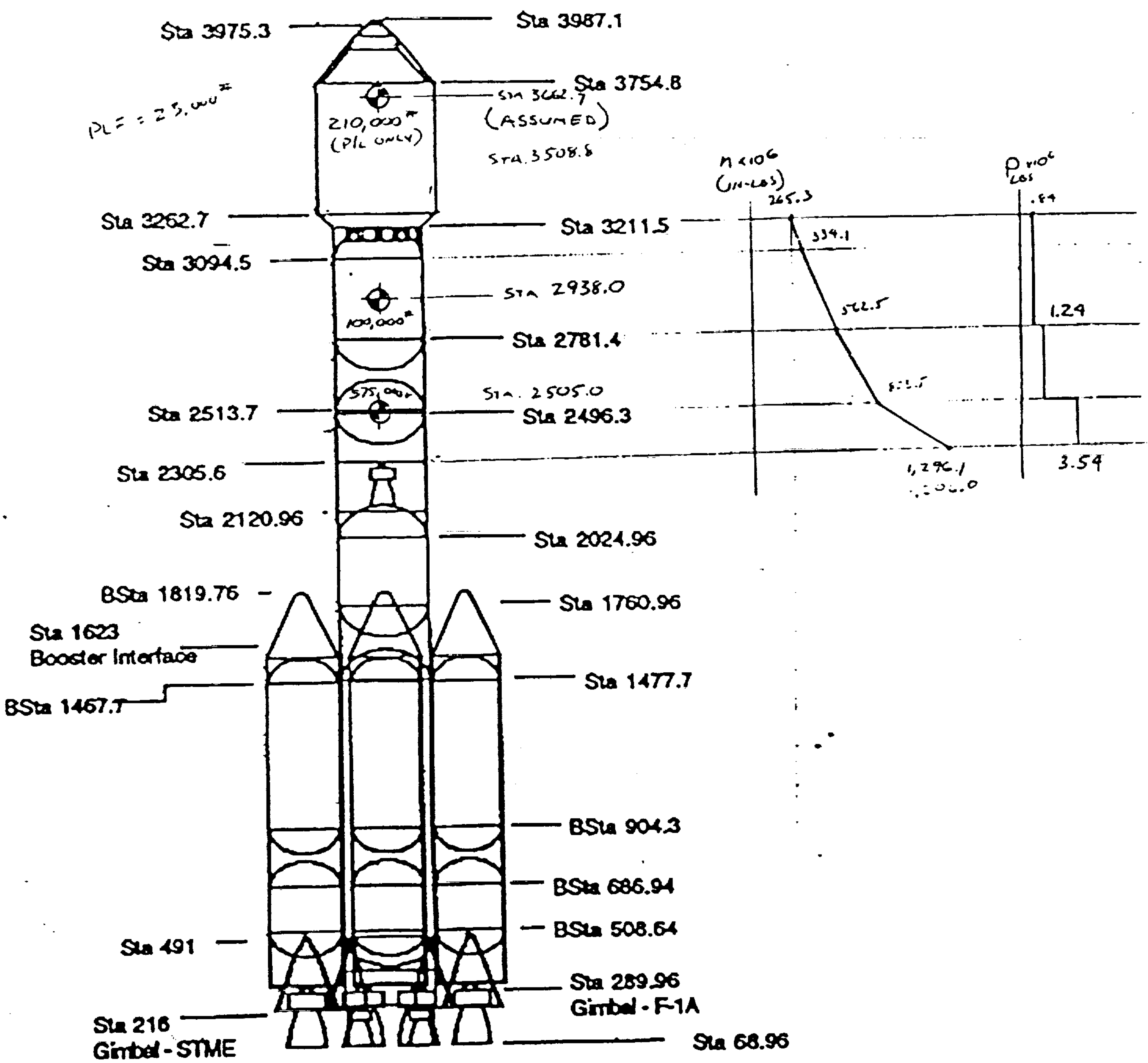
BY LDC DATE 10/5/92 SUBJECT STV TLI LOADS SHEET NO. 1 OF

CHKD BY DATE JOB NO.

MAXIMUM ACCELERATION LOADS; AXIAL 4.0g (LIMIT)
LATERAL 2.8g (LIMIT)

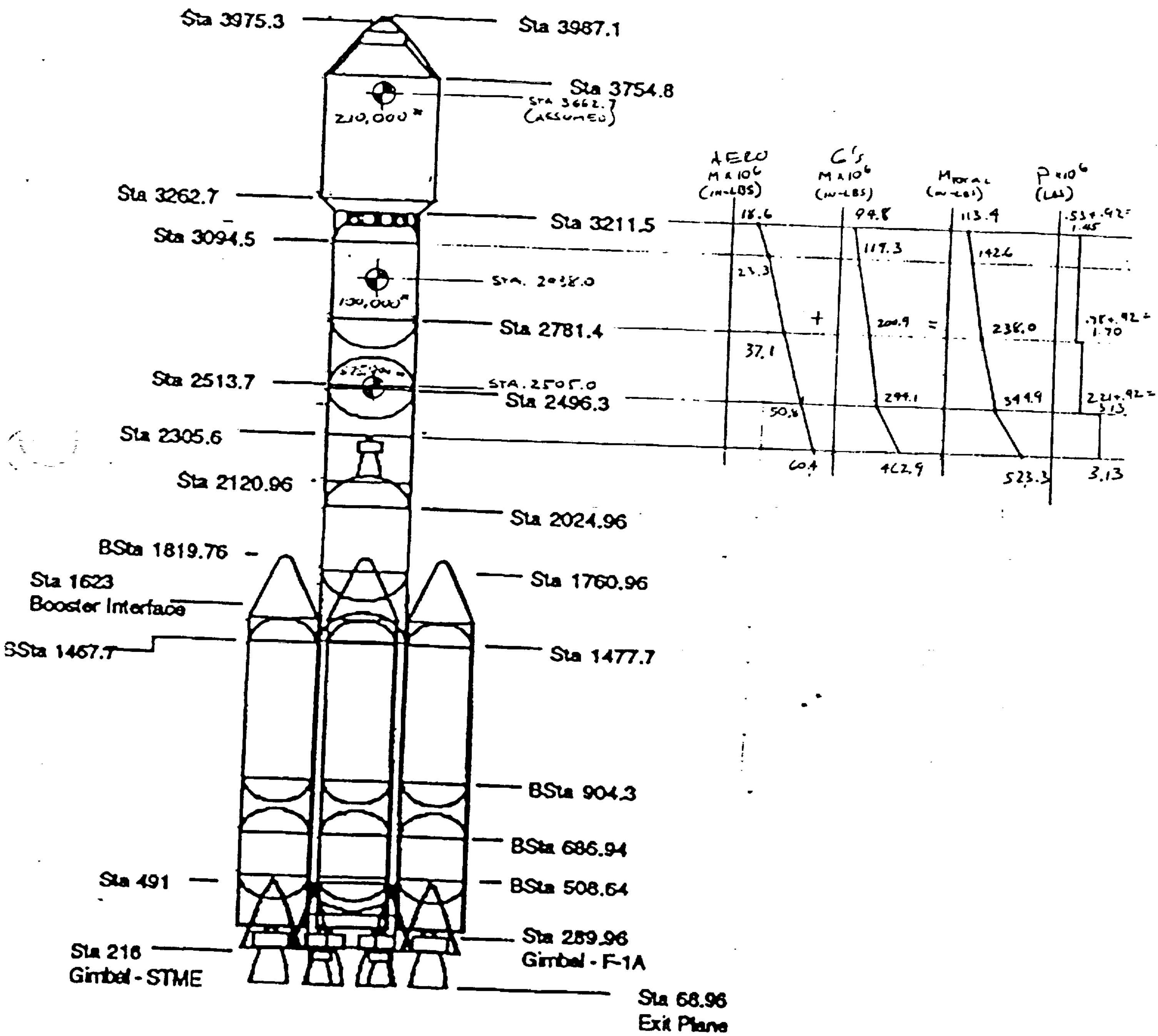


FOR THE LOAD CALCULATION, THE TLI IS ASSUMED TO BE CANTILEVERED FROM STATION 2305.6.

MARTIN MARIETTA

BY J.D.C. DATE 10/5/92 SUBJECT S.T.V. T.L.I. LOADS SHEET NO. 2 OF
KD BY DATE JOB NO.

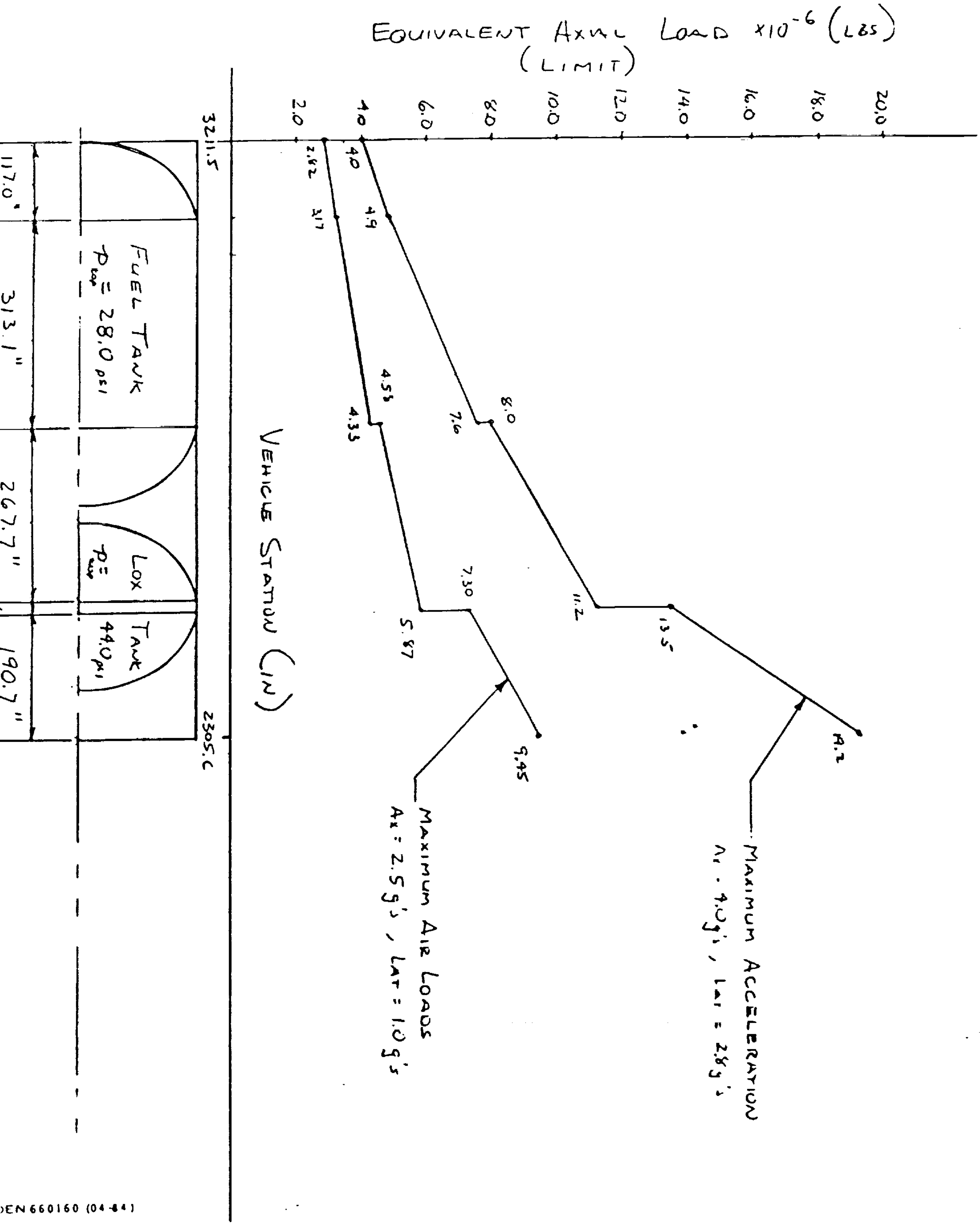
MAXIMUM AIR LOADS: AXIAL 2.5 g (LIMIT)
(REF. CS&C MEMO: 9/21/92) LATERAL 1.0 g (LIMIT)



MARTIN MARIETTA

BY JDC DATE 10/5/92 SUBJECT STV TLI LOADS SHEET NO. 3 OF

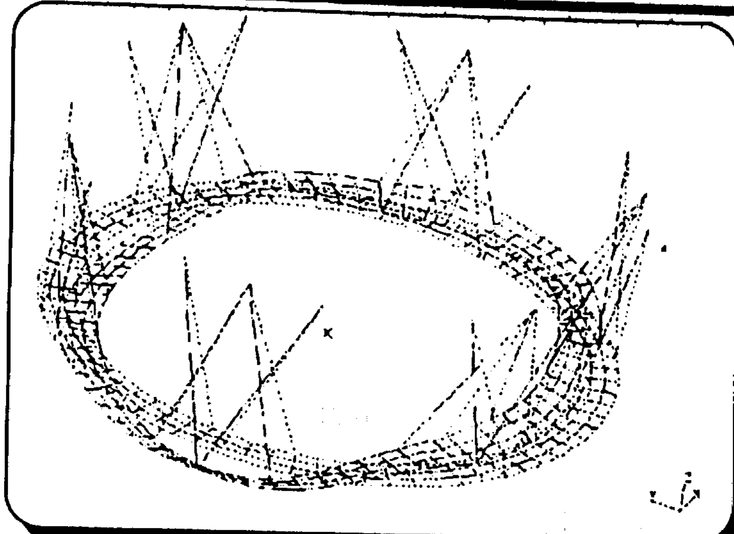
CHKD BY DATE JOB NO.



P/A Module - Avionics Deck Analysis

MSFC

Representative
Deflection Plot
Showing ± .5 cm



Design Load Factors

Location	Thrust(g) GRMS	Radial(g) GRMS
Sequencer Panel	5.9	7.8
Switch Selector Panel	4.8	6.6
APS Aft Attach	2.8	57
Thrust Structure	1.8	3.0
Engine B=Gimbal	4.5	6.8
Field Splice I	6.0	8.9
Field Splice II	4.4	9.1
Maximum (GRMS)	6.0	9.1
Design Load Factors	8.4	12.7

Derived Internal and External Acoustic Envr.

Center Freq. (HZ)	External SPL(dB)	Internal SPL(dB)
25	133.634	130.634
Thru	-	-
2500	132.134	129.134
Total*	151.1	148.1

* 3 dB Delta Due to Transmission Losses Through Skin

compare with SSME

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RS920821-01A

TLI Stage Avionics Placement Issues

MSFC

USRS Aft Placement Summary:

- Multiple Configurations For Forward Mounted Avionics Based on Different Diameter Payload Adaptors. Single Configuration For Aft Placement
- Parallel Processing of Subsystem Apart from Primary Structure

Saturn V Based Avionics Deck Analysis:

- Acoustic OSPL Delta Between Forward and Aft is ± 2dB. Total is ~ 150 dB
- Random Vibration at Both Locations Differs by ~ ± 10 - 15 %
- Design Load Factors Will Remain Unchanged

C-9

TD16-013
Upper Stage Technical Requirements Document
Source Analysis for Paragraph:
1.8d Environmental Impact - Facilities Development

Environmental impact is an area of increasing concern with respect to any new development, especially those of technical or industrial character.

The reasonable expectation is that any new facilities associated with processing, production, or launch of a new upper stage will have to comply with environmental impact requirements, and that costs can be reduced by taking a pro-active approach

TD16-014
Upper Stage Technical Requirements Document
Source Analysis for Paragraph:
1.8e Environmental Impact - Hazardous Effects

Increasing environmental awareness argues for attention to the details concerning hazardous or toxic emissions. Addressing such concerns in the planning stage can reduce costs incurred for toxic byproduct disposal from manufacturing, lower transportation costs & expand options.

TD16-015
Upper Stage Technical Requirements Document
Source Analysis for Paragraph:
1.9a Safety

Based on conversations with Martin, MSFC, and KSC Safety Community representatives:

1) Operations on the ETR facility will be governed by provisions of ESMCR 127-1
2) Operations on the WTR facility will be governed by provisions of WSMCR 127-1
3) Operations on the KSC facility will generally comply with the provisions of KSC 1098.

Any flight hardware associated with the STS will comply with NSTS 1700.7B, and any ground support equipment (GSE) which interfaces with such hardware will comply with KHB 1700.7B (joint air force document is 45SPWHBS10
Implementation procedures will be taken from NSTS 13830 Rev. B.
Specific subsystem/element procedures will be followed as applicable

TD16-016
Upper Stage Technical Requirements Document
Source Analysis for Paragraph:
1.10a Disposal

The current concept is to use Broad Ocean Area disposal for LEO missions, Disposal Orbits for GEO missions, and Deep Space disposal for any interplanetary missions. The impacts of the disposal requirement were looked at in terms of propellant costs for 3 cases; use of the RCS, propulsive venting, and use of an additional main propulsion burn.

RCS: Isp of 220s, stage mass fraction 0.85

For a 100 m/s BOA disposal maneuver, additional RCS propellant required would represent a 0.7% delta to the stage liftoff mass, exclusive of tankage & plumbing. For a 30 m/s Disposal Orbit, or deep space disposal this would come down to 0.2%

Propulsive vent: using an Isp of 170s for hydrogen, other assumptions similar to RCS case: 0.85% delta to liftoff mass for 100 m/s, 0.26% for 30 m/s.

Additional main propulsion burn: Isp ≈ 440s, same assumptions as above;

0.35% delta liftoff mass for 100 m/s disposal burn,
0.1% delta L/O mass for 30 m/s burn

With the mission-operational complexity of additional propellant settling & main engine ignition and ops.

TD16-017
Upper Stage Technical Requirements Document
Source Analysis for Paragraph:
1.11a Piloted Flights

Behind this requirement is the desire to address potential manned exploration missions, such as a Lunar outpost or Mars mission. The current perception on human-rating of systems is that the requirement are nebulously defined, and attempting to meet all proposed requirements would be economically if not technically impossible.

Because of this, and the philosophy behind the New Upper Stage that it is ment to be a low cost, ~~[illegible]~~ multi-purpose mission element, the proposed approach is similar to the one taken with the SIVB stage on the Apollo program. The New Upper Stage will depend on assured crew safety, crew escape and safe haven systems provided by other mission elements. In support of piloted flights, the New Upper Stage should be free from potential catastrophic failure modes, and provide Caution & Warning status/condition information to the crewed mission element (and to mission control for piloted flights).

TD16-018
Upper Stage Technical Requirements Document
Source Analysis for Paragraph:
1.12a Guidance, Navigation and Control - Accuracies

3σ GN&C accuracies for the New Upper Stage

Mission:	R_p (nm)	R_a (nm)	inc. (deg.)
Geo Equitorial	115	115	0.2
GEO Low Inclination	115	115	0.2
GEO High Inclination	100	100	0.15
12-hr. Eccentric	10	100	0.1
LEO (all)	5	5	0.1

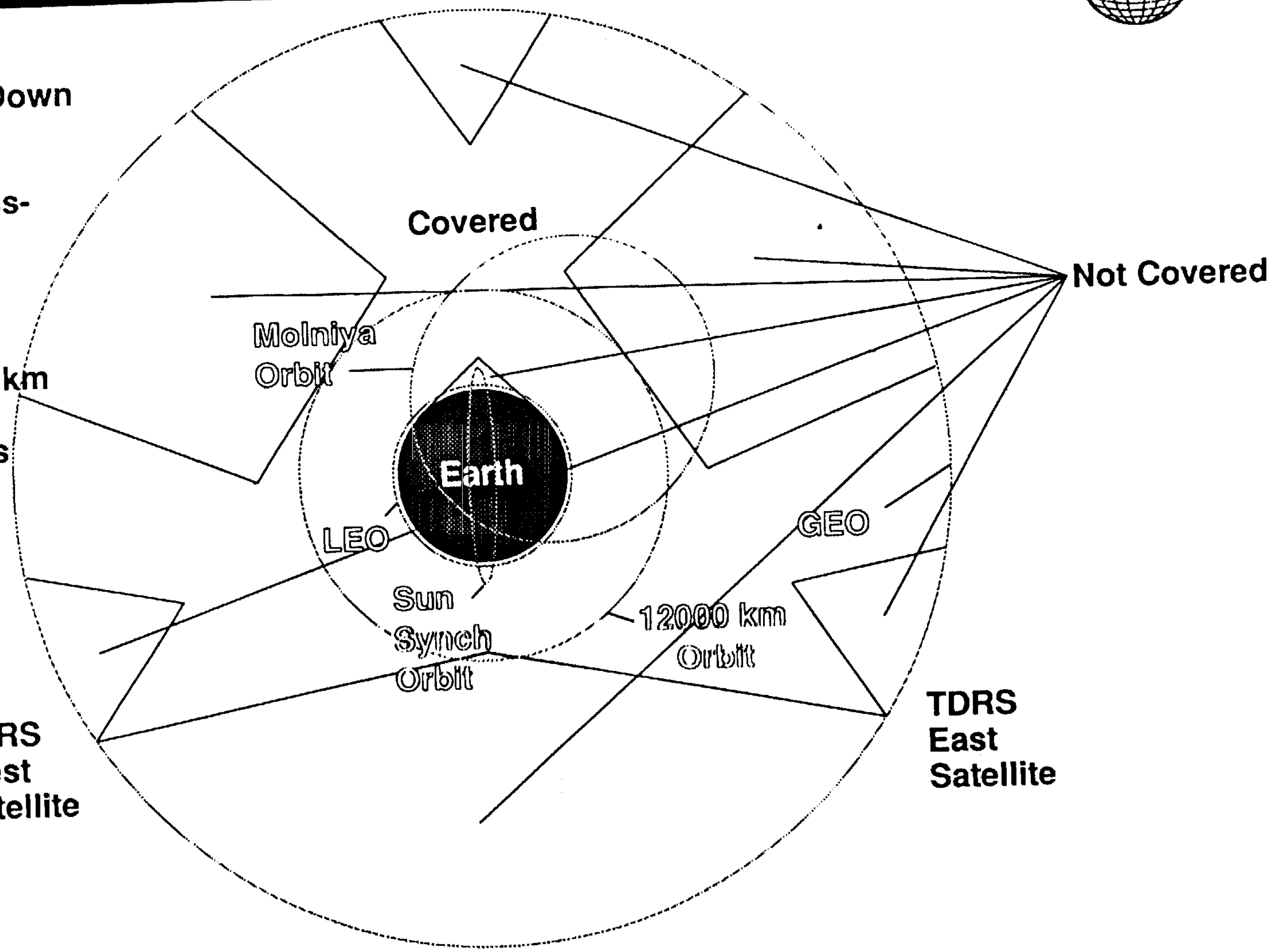
Studies were made of USRS requirements (AF customer), Centaur capabilities & Titan III capabilities. Following this, analysis was done to indicate the performance requirements which these figures would dictate for Inertial Navigation hardware. Results of this analysis indicated that medium accuracy accelerometers (50 μg bias) and gyroscopes (0.2 deg./hr.) in combination with GPS could achieve the above accuracies using very economical hardware.

TD16-019
Upper Stage Technical Requirements Document
Source Analysis for Paragraph:
1.13a Communication

TDRS System Coverage

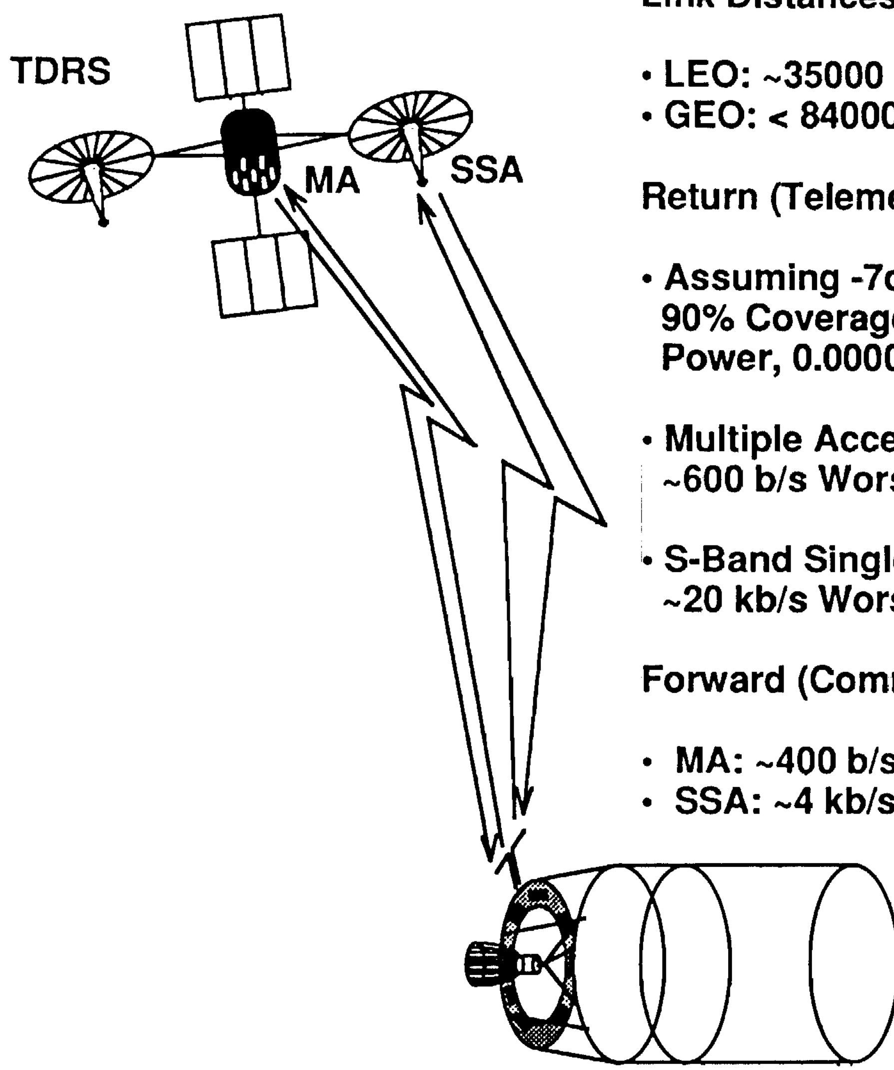
MSFC

- View Looking Down on North Pole
- Equitorial Cross-Section
- Coverage Is Complete between 1200 km and 12000 km Altitude Orbits



TDRS System Link Characteristics

MSFC



Link Distances:

- LEO: ~35000 km to 48000 km
- GEO: < 84000 km

Return (Telemetry) Link Rates:

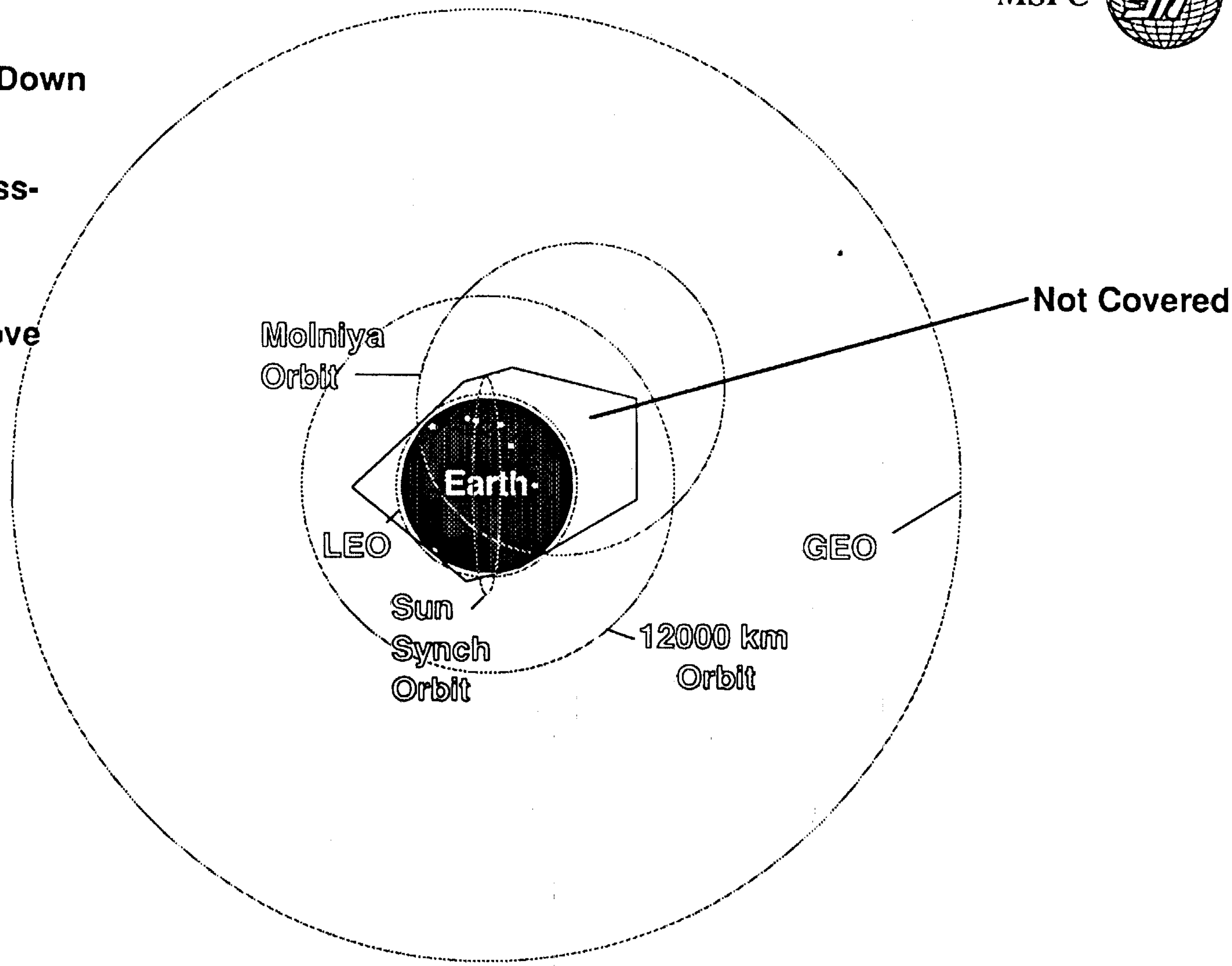
- Assuming -7dBi U/S Antenna Gain (Worst Case 90% Coverage Condition), 20 Watts Transmitter Power, 0.00001 Bit Error Rate, S-Band Operation
- Multiple Access (MA) Link: ~1300 b/s LEO, ~600 b/s Worst Case in GEO (if in Coverage Area)
- S-Band Single Access (SSA) Link: ~40 kb/s LEO, ~20 kb/s Worst Case GEO (if in Coverage Area)

Forward (Command) Link Rates:

- MA: ~400 b/s LEO, ~200 b/s Worst Case GEO
- SSA: ~4 kb/s LEO, ~2 kb/s Worst Case GEO

SGLS System Coverage

- View Looking Down on North Pole
- Equitorial Cross-Section
- Coverage Is Complete Above 12000 km



Communications:

RTS/SGLS characteristics -

2mb/s. data rate max

LEO - 95% antenna coverage → -7dBi w/c
(with polarity & diversity combining)

2 watts transmit power provides 1mb/s data down link w/ 10^{-5} BER
→ large LOS zones, short (~8 min.) link periods

GEO - 90% antenna coverage → -7dBi w/c

20 watts (xmitter amp max power) provides 100kb/s down link w/ 10^{-5} BER
→ virtually no LOS zones

→ front end cost of SGLS flight hardware is approx. ½ of TDRSS equivalent systems - \$2m vs. 4m for redundant U/S comm system

TD16-020
Upper Stage Technical Requirements Document
Source Analysis for Paragraph:
1.14a Operability - Ground Operational Process

Candidate operational processes considered were UES or Universal Environmental Shelter, ITL or Integrate-Test-Launch, and a hybrid of these two previously mentioned approaches which partially integrated the upper stage, payload & fairing, allowing a reduced capability UES structure to be used. The ITL approach was selected (also referred to as "Integrate-Encapsulate-Launch") based on its response time (ref. "payload substitution" rqt.) and parallel processing capabilities, which could support higher launch rates than were perceived as possible with the UES or partial UES approach. Also, given the hazardous operations nature of much on-pad activity, processing costs could be reduced using the ITL approach as opposed to UES, by requiring an absolute minimum of on-pad activity.

TD16-021
Upper Stage Technical Requirements Document
Source Analysis for Paragraph:
1.14b Operability - Standard Payload Interfaces

Standard interfaces, not just for payloads but for launch systems (to upper stage) and within the upper stage, are central to the cost reduction approach for the next generation upper stage. Integration costs represent as much as 25% of total launch costs, and interface control is a sizeable fraction of the integration cost. Standardization allows for reduction of interface costs to a minimum, while increasing reliability through continuous improvement across the life of the system.

Standard interfaces to the payload shall cover both mechanical & electrical I/F. Payload services shall be minimized to reduce I/F complexity. Payload I/F shall support portability of upper stage functions to the payload, and shall be transparent to the operation of the system in terms of those functions.

TD16-022
Upper Stage Technical Requirements Document
Source Analysis for Paragraph:
1.14c Operability - Payload Substitution

The P/L substitution requirement is drawn from past experience with the military launch services market. Flexibility is desired to support rapid changes in P/L priority.

Taken from:

AF SPACECOM SORD, para. 4.1.1.1.c.2

TD16-023
Upper Stage Technical Requirements Document
Source Analysis for Paragraph:
1.14d Operability - Launch System Compatibility

Access To Space Arch. Vehicle Options

MSFC

Launch Vehicles Description	Option #1				Option #2				Option #3			
	LEO lbs	P/L Dia.	P/L Lng.	G's A/L	LEO lbs	P/L Dia.	P/L Lng.	G's A/L	LEO lbs	P/L Dia.	P/L Lng.	G's A/L
STS	50k	15	60	3.2/2.5	50k	15	60	3.2/2.5	50k	15	60	3.2/2.5
STS Upgrades	?	15	60	?	?	15	60	?	-	-	-	-
ELV's												
Delta 7920	11k	9.1	12	6/2	11k	9.1	12	6/2	11k	9.1	12	6/2
Atlas IIAS	18.5k	12	13.7	6/2	18.5k	12	13.7	6/2	18.5k	12	13.7	6/2
Titan IV	40k	15	66	6.5/1.5	40k	15	66	6.5/1.5	40k	15	66	6.5/1.5
ELV's Upgrades												
Titan IV/SRMU	48k	15	66	?	48k	15	66	?	-	-	-	-
Spacelifter												
20K	20k	?	?	?	20k	?	?	?	-	-	-	-
50K	50k	?	?	?	50k	?	?	?	-	-	-	-
Vehicle/CTV/PLS												
50k	-	-	-	-	50k	?	?	?	-	-	-	-
80k	-	-	-	-	80k	?	?	?	-	-	-	-
SSTO/TSTO	-	-	-	-	-	-	-	-	45k 25k	?	?	?
Exploration Vehicle	-	-	-	-	250k 310k	?	?	?	250k 310k	?	?	?

Which will new U/S be compatible with? see next pg

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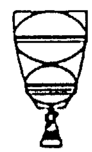
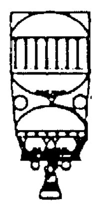
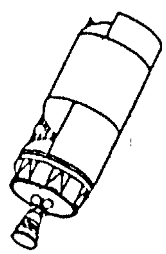
RS930416-05A

Need VG of these for 1.14d

Access To Space Upper Stage Options

MSFC

Upper Stages	Mass Properties	Application Option	Poss. Launch Veh.
 Small US	Dry Mass ? ~~3,681~~ Propellant 11,618 RCS Prop ? ~~136~~ Total Mass 15,435 Contingency 20% Diameter 4.1 m Length 8.0 m	Option #1 Option #2 Option #3	STS, Titan IV, Spacelifter STS,Titan IV, Spacelifter,CTV/PLS STS, Titan IV, SSTO
 Medium US	Dry Mass 3,349 Propellant 23,024 RCS Prop 136 Total Mass 26,509 Contingency 20% Diameter 4.3 m Length 10.3 m	Option #1 Option #2 Option #3	Titan IV, Spacelifter Titan IV, Spacelifter,CTV/PLS Titan IV
 Exploration US	Dry Mass 26,600 Propellant 306,331 RCS Prop 1,257 Total Mass 334,188 Contingency 20% Diameter 8.4 m Length 24.7 m	Option #1 Option #2 Option #3	N/A Exploration Vehicle Exploration Vehicle

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Preliminary Data Still In Work

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RS930428-07

TD16-024
Upper Stage Technical Requirements Document
Source Analysis for Paragraph:
1.15a Maintainability - Detection / Isolation of Failures

Cost of Built-In-Test (BIT) Coverage Trade

- Trade Study Was Performed to Determine the Life Cycle Costs (LCC) of Various Degrees of Bit Coverage

- Three Areas of LCC Were Analyzed
  - Design And Development
  - Production
  - Operations Support

- A Cost Model Was Derived from MIL-STD-1591a to Provide a 'Cost of BIT' Comparison Versus Percent Bit Coverage

- A First Analysis Consisted of a Single Representative ~~A~~US LRU Containing Approximately 20 Circuit Cards. Design Hours, Test Costs, and Operations Data Were Derived from Martin Marietta Experience and Industry Data

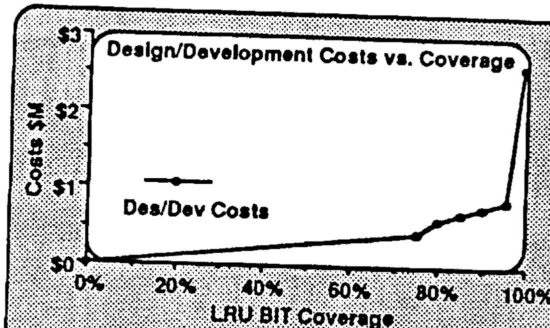
- A Second Analysis Was Conducted Representing a Full ~~AUS~~ VS Avionics Suite

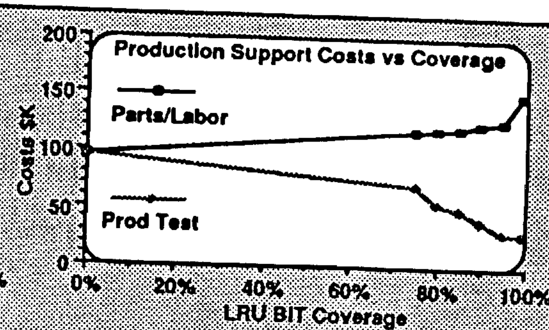
1.15a

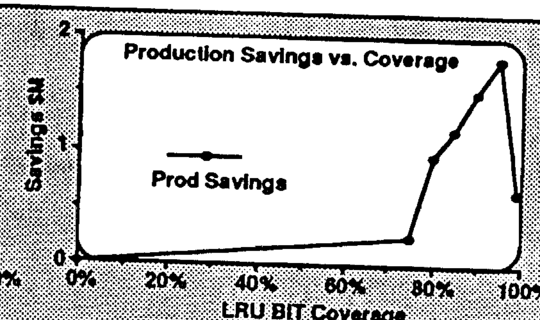
US - Built-In-Test Requirements Analysis

MSFC

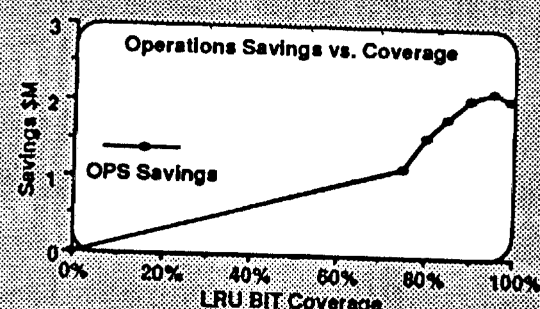
Life Cycle Costs vs. Percent of Fault Coverage for Typical Line Replaceable Unit Design

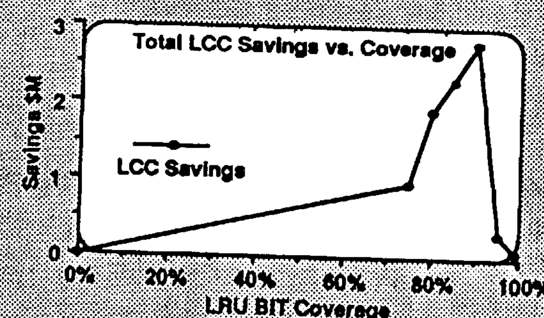






Optimum Range of BIT Coverage Versus Life Cycle Cost is 90 to 95%





LCC Improvement of Approximately $3M Possible for Typical Electronic LRU

Net LCC Improvement for Avionics System Consisting of Multiple Electronic LRUs is Approximately $20-40 Million

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TD16-025
Upper Stage Technical Requirements Document
Source Analysis for Paragraph:
1.15b Maintainability - Routine Maintenance

Maintenance performed on the pad represents a costly option, Largely based on the hazardous nature of such activity. Faster Launch turn-around & reduced costs and hazards can be achieved by design of the system with the understandin from the outset that once on-pad status has been achieved, no <u>routine</u> maintenance activity should be required. This takes into account only expected on-pad pre-Launch times of (TBD), after which, maintenance may be required. Also allowed with demonstrated operational benefits.

~~[illegible]~~ drawn from:

AF SPACECOM SORD para. 4.1.2.A

TD16-026
Upper Stage Technical Requirements Document
Source Analysis for Paragraph:
1.15c Maintainability - LRU Failure Handling

This requirement governs handling of line replaceable units determined to be non-conforming during normal ~~[illegible]~~ processing. This replacement philosophy allows the limitation of required spares, and reduces GSE and general processing facility capability requirements

drawn from:

AFSPACECOM SORD 4.1.2.A

TD16-027
Upper Stage Technical Requirements Document
Source Analysis for Paragraph:
1.15d Maintainability - Paperless Work Environment

This requirement is drawn from:
AFSPACECOM SORD 4.1.2A

TD16-028
Upper Stage Technical Requirements Document
Source Analysis for Paragraph:
1.16a Transportation - Federal, State and Local Req'ts

Make it good ... 1.1lea ... Need conclusions

Upper Stage Transportation Reqts

MSFC

Air Transportation

- Maximum Diameter Allowed is:
  C-130: 8 ft
  C-141: 9 ft
  C- 17: 13.5 ft
  C-5: 13.5 Ft
  C-5 SCM: 14.5 ft
  Super Guppy: 20 ft

- Road Transport is still an issue since shipping to airfield is required

- Air Transport can cost up to $1.5 M per flight (Super Guppy)

Rail Transportation

- Maximum Diameter is 14 ft unless special routing can be supplied (avoid signal crossings, transfer tracks, tunnels, etc.)

- Maximum Weight is 250 Klbs per axle set

- Rail Transportation approximately $200K for dedicated service

Upper Stage Transportation Reqts (Cont'd)

Road Transportation

- Maximum Diameter is 20 ft without special DOT permits and state by state approved routing
- Maximum Weight is 25 Klbs per axle set
- Road Transportation cost is a function of weight delivered and distance travelled

Barge Transportation

- Maximum Diameter is XX ft
- Maximum Weight is XX Klbs
- Barge Transportation requires access to waterway from manufacturing site
- Barge Transportation is the least costly option for long distance shipping

1.15b MAINTAINABILITY

- THOUGHT ABOUT THIS A LITTLE

1.16a TRANSPORTATION

- BARGING IS CHEAPEST, BUT SLOWEST

1.20a PROXIMITY OPS

- THOUGHT ABOUT THIS A LITTLE

1.21a COMMONALITY

- COMMON COMPONENTS/SUBSYSTEMS BETWEEN LARGE AND SMALL US
- DEPENDENT ON MISSION MODEL MIX

TD16-029
Upper Stage Technical Requirements Document
Source Analysis for Paragraph:
1.16b Transportation - Delivery from Manufacturer

1.16a

Upper Stage Transportation Reqts

MSFC

Air Transportation

- Maximum Diameter Allowed is:
  C-130: 8 ft
  C-141: 9 ft
  C- 17: 13.5 ft
  C-5: 13.5 Ft
  C-5 SCM: 14.5 ft
  Super Guppy: 20 ft
- Road Transport is still an issue since shipping to airfield is required
- Air Transport can cost up to $1.5 M per flight (Super Guppy)

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Upper Stage Transportation Reqts (Cont'd)

MSFC

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- THOUGHT ABOUT THIS A LITTLE

1.21a COMMONALITY

- COMMON COMPONENTS/SUBSYSTEMS BETWEEN LARGE AND SMALL US
- DEPENDENT ON MISSION MODEL MIX

TD16-030
Upper Stage Technical Requirements Document
Source Analysis for Paragraph:
1.17a Security

Security requirement derives from DoD / classified component of upper stage / Launch services market which it is desired to address.

The level of security supported by all routine operations and facilities is typically determined by the highest requirements of any given user; certain activities may be neglected in missions with lower requirements, but care must be taken to avoid compromising facilities & service shared with higher classification missions.

TD16-031
Upper Stage Technical Requirements Document
Source Analysis for Paragraph:
1.18a Availability - System Life Cycle

1. 18a

US - Availability Requirements Analysis

MSFC

Availability is the Probability that a System is Operating Satisfactorily at any Point in Time when used under Stated Conditions

There are Three Types of Availability:

- Inherent
- Achieved
- Operational

For the Technical Requirements Document, Operational Availability is used since its measure includes both Inherent and Achieved Availability as well as the Additional Parameters of Logistics and Administrative Downtime

MTTR and MDT data where taken from the Advanced Upper Stages Technology Study

118a

US - Availability Requirements Analysis

MSFC

Same options used in reliability analysis.

Configuration	On-Pad Availability MDT = 56 Hours	On-Pad Availability MDT = 32 Hours
Baseline	.9244	.9590
Option 1	.9269	.9615
Option 2	.9195	.9550
Option 3	.9156	.9517
Option 4	.8905	.9343
Option 5	.8585	.9139

- Operational Availability (Ao) = MTMBA/(MTBMA + MDT)

  Where: MTMBA = Mean-Time-Between-Maintenance-Actions
  MDT = Mean-Down-Time

And: MTBMA = 6.67 (MTBF) .7 On-Pad Environment
MDT Centaur = 56 Hours
MDT Advanced US = 32 Hours

Requirement is 0.90 Probability for System Availability

Sensitivity Analysis Shows MDT has Greatest Influence on System Availability

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TD16-032
Upper Stage Technical Requirements Document
Source Analysis for Paragraph:
1.18b Availability - Stand Down Duration, Probability

The allowed system stand-down duration and probability levels were taken from:

AFSPACECOM SORD 4.1.1.4A

Compliance determination program should be established to track Availability risks via analytical model.

TD16-033
Upper Stage Technical Requirements Document
Source Analysis for Paragraph:
1.19a Dependability - Definition

The definition of the term "Dependability" was drawn from:

AFSPACECOM SORD para. 4.1.1.4.1a

TD16-034
Upper Stage Technical Requirements Document
Source Analysis for Paragraph:
1.19b Dependability - Factors in Calculation

The external factors to be included in the calculation of dependability were taken from:

AFSPACECOM SORD para. 4.1.1.4.1A

TD16-035
Upper Stage Technical Requirements Document
Source Analysis for Paragraph:
1.19c Dependability - Required Rate

The required rate of dependability was drawn from:
AFSPACECOM SORD para. 4.1.1.4.1.A

TD16-036
Upper Stage Technical Requirements Document
Source Analysis for Paragraph:
1.20a Proximity Operations

The inclusion of the prox ops requirement was motivated by the desire to address missions which would approach the space station Freedom. Full support of proximity operations was seen as requiring stable 3dof translation capability, in addition to higher than planned levels of redundancy in all systems associated with guidance, navigation and control or propulsion.

Because the impact of supporting even a scaleable architecture to approach full prox-ops capability, the decision was to scar the upper stage to "support" prox ops by providing the necessary capability to act as a stable, passive (& cooperative) target for a more specialized prox ops vehicle.

Currently, the intention is to provide RGPS relative position determination and assured main propulsion disable as the prox ops support capabilities, ~~[illegible]~~ and to planning ~~allowing~~ the inclusion of docking structure on the upper stage payload (as opposed to the upper stage)

TD16-037
Upper Stage Technical Requirements Document
Source Analysis for Paragraph:
1.21a Commonality

Requirements Analysis Task Plan

Requirement: 1.21a Commonality among hardware/software and operations must be emphasized in the event that a family of concepts is needed to fulfill the mission requirements.

Responsible Individual/Supporting Individual(s): Bob Spencer / Rob Mason / Jim Cathcart

Summary of Approach:

- Generate a list of common elements for the upper stage concept
- Derive a credible set of evaluation criteria for assessing the impacts of multiple vehicle fits of upper stage common elements.
- Perform evaluation of this data to the upper stage architectures identified for this task.
- Summarize results in a matrix that will identify the cost savings/penalties for each element and it vehicle quantity match.

Task Description:

Identify areas with high feasibility for HW / SW , op's commonality and assess the benefits of multiple implementation across family of upper stages

Interfaces: N/A

Inputs

- Top level configuration definition of an upper stage
- Well defined set of evaluation criteria to be used on a range of common configurations
- Results of Req. Analysis 1.20a, 1.14a.

Outputs

- Data base of common elements
- Matrix of cost savings/penalties for different number of vehicle matches

Schedule

	March	April	May	June
Program Milestones	Δ Start TD 8; Δ Prelim Concept Description 17	Δ Internal Progress Review 9	Δ End TD 31	
Task Milestones				

1.15b MAINTAINABILITY

- THOUGHT ABOUT THIS A LITTLE

1.16a TRANSPORTATION

- BARGING IS CHEAPEST, BUT SLOWEST

1.20a PROXIMITY OPS

- THOUGHT ABOUT THIS A LITTLE

1.21a COMMONALITY

- COMMON COMPONENTS/SUBSYSTEMS BETWEEN LARGE AND SMALL US
- DEPENDENT ON MISSION MODEL MIX

TD-09 Upper Stage Common Elements

FUNCTIOI CODE	DESCRIPTION Access to Space Architectures	Access to Space Arch. Option Upper Stages US #1	US #2	US #3	US #4	US #5	Moment Arm (in)	Moment (lb-in)
01	UPPER STAGE							
1	**Tank Structure**							
	01 FWD Dome LOX							
	02 Kick Ring							
	03 Cylindrical Section							
	04 Kick Ring							
	05 AFT Dome LOX							
	06 Debris Sheilding LOX							
	07 Slosh Baffle LOX							
	08 Vortex Baffle LOX							
	09 FWD Dome LH2							
	10 Kick Ring							
	11 Cylindrical Section LH2							
	12 Kick Ring							
	13 AFT Dome LH2							
	14 Debris Sheilding LH2							
	15 Slosh Baffle LH2							
	16 Vortex Baffle LH2							
2	**Additional Structure**							
	01 FWD Interface Ring							
	02 Intertank							
	03 Avionics Mounting Structure							
	04 Engine Thrust Structure							
3	**Thermal Management**							
	01 LOX MLI (0.1 in)							
	02 LOX SOFI (0.375 in)							
	03 LH2 MLI (0.1 in)							
	04 LH2 SOFI (0.375 in)							
	05 Avionics Blankets							
4	**Engines**							
	01 SSME (1x)							
	02 Actuator System							
	03 Lines/Valves/Fittings							
	04 Instrumentation							

1.21a (cont)

TD-09 Upper Stage Common Elements

FUNCTIOI CODE	DESCRIPTION Access to Space Architectures	Access to Space Arch. Option Upper Stages US #1	US #2	US #3	US #4	US #5	Moment Arm (in)	Moment (lb-in)
5	GHe System							
01	Helium Tank (6x@4500 psi-24"OD)							
02	Helium							
03	Lines/Instumentation							
6	Reaction Control System							
01	Thrusters (12 @ 4.1lbf Thrust ca.)							
02	Hydrazine Tank (4x@450 psi-36"OD)							
03	Hardware/Lines/Instrumentatoin							
04	GHe Pressurant							
Total Avionics System								
7	Guidance Navigation/Control							
01	INU							
02	GPS Receivers							
03	GPS Antenna							
04	RF Switch							
05	Optical Horizon Sensors							
06	Optical Sun Sensors							
07	TVC Control Unit							
08	RCS Control Unit							
8	Mission Management							
01	Mission Manager							
02	Data Aquisition							
9	Communication (Grnd/SSF)							
01	Antennas							
02	Diplexer							
03	RF Combiner							
04	RF Transfer Switch							
05	Transponder							
06	Transmit Amplifier							
07	Comminications System I/F							
10	Power System							
01	Laser Ordnance Firing Unit (Interface)							
02	Power System I/F							
03	Protection / Switching							

1.21a (cont)

TD-09 Upper Stage Common Elements

FUNCTIOI CODE	DESCRIPTION Access to Space Architectures	US #1	US #2	US #3	US #4	US #5	Moment Arm (in)	Moment (lb-in)
04	Batteries (LiSOCL2) 20hrs							
05	Heater Batteries							
05	Core Umbilical							
06	Cabling							
11	**Range Safety Subsystem**							
01	Lighting Stubs							
02	UHF Antennas							
03	Hybrid Coupler							
04	Directional Coupler							
05	Receiver/Decoders							
06	C- Band Transponder							
07	RF Ciculator							
08	C-Band Antennas							
09	RS Batteries (Silver Zinc)							
10	RS Power Distribution							
11	Pyro Charge							
12	Laser Ordnance Firing Unit (Sys/Des)							
12	**Growth Contingency (20%)**							
13	**Propellant (1:6.0)**							
01	LH2							
02	LOX							
03	Unusable (%)							
14	**RCS Propellant**							
01	Usable							
02	Unusable (5%)							
	Upper Stage Dry							
	Upper Stage & Propellant							
02	**Upper Stage-Booster Adaptor**							
01	Structure							
02	Contingency (20%)							
	Total Adaptor							
	Stage Effective Mass Frac.							

TD16-038
Upper Stage Technical Requirements Document
Source Analysis for Paragraph:
1.22a Technology - Criteria for Pursuit

Prioritized Upper Stage Technologies

Enhancing Technologies Prioritized by Functional Area

Due to the Large Number of Technologies, each Function Subdivided into Two or Three Prioritized Groups with Technologies Listed in Group 1 Having the Highest Priority.

Technologies are Prioritized within each Group allowing Focus on First One or Two Technologies in Group 1 of each Functional Area

Technologies Prioritized Based Upon the Following Evaluation Criteria:

Cost
Schedule
Risk
Readiness Level
Safety/Reliability
Commonality to ELVs

Inclusion OK. per Jim McKinney's 7.27.93 pem

~~comments~~ ~~to~~

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Prioritized Upper Stage Avionics Technologies

Upper Stage Avionics Enhancing Technologies Prioritized Based Upon Cost, Schedule, Readiness Level, Risk, Commonality to ELVs, Safety/ Reliability, and Commercial Applications

Group 1

1. IVHM (Sensors, S/W, etc.)
2. Laser Initiated Pyrotechnics
3. Fault Tolerant Avionics
4. Fiber Optics Data Bus
5. GPS Assisted GN&C

Group 2

1. Solid State IMUs (IFOG)
2. Electromechanical Actuators (valves/TVC)
3. Standard Interfaces (power/data)
4. Software Development/Management Tools

Group 3

1. High Density Power (long life batteries)
2. Pentad/Hexad Technology
3. Common Processor Set
4. Open Avionics Architecture

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Prioritized Upper Stage Advanced Operations

Upper Stage Operations Enhancing Technologies Prioritized Based Upon Cost, Schedule, Readiness Level, Risk, Commonality to ELVs, Safety/Reliability, and Commercial Applications

Launch Operations

- Group 1
  - 1 Auto Detection of Anomalies
  - 2 Real Time System Status
- Group 2
  - 1 Computerized Data Acquisition
- Group 3
  - 1 On-Line Historical Database

Ground Operations

- Group 1
  - 1 Automated Checkout & Test
  - 2 Laser Initiated Ordnance
- Group 2
  - 1 Automated Propellant Loading
  - 2 Automated Leak Detection
- Group 3
  - 1 Robotic Inspections

Mission Operations

- Group 1
  - 1 Real Time Mission Status Display
- Group 2
  - 1 Automated Advisory Tools
  - 2 Advanced Software Systems
- Group 3
  - 1 Vehicle/Fault Simulations

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Prioritized Upper Stage Propulsion Technologies

Advanced Propulsion Enhancing Technologies Prioritized Based Upon Cost, Schedule, Readiness Level, Risk, Commonality to ELVs, Safety/Reliability, and Commercial Applications

Group 1

1. Improved EMA Valves
2. Failure ID Algorithms
3. Standard Acces Interfaces
4. EMA/Electro-HydrostaticTVC
5. Advanced Tank Material

Group 2

1. Advanced Thermal Insulation
2. Integrated Modular Engine
3. Advanced Pressurization
4. 35-70Klb Thrust Cryo Engine
5. Enhanced Throttling Range
6. SSME On-Orbit Start Capability

Group 3

1. Long Term Cryo Storage
2. GH2/GO2 RCS
3. Advanced Fluid Transfer & Instrumentation
4. Plume Spectroscopy
5. Holographic Infared Leak Detection

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Prioritized Structures & Materials Technologies

Structures & Materials Enhancing Technologies Prioritized Based Upon Cost, Schedule, Readiness Level, Risk, Commonality to ELVs, Safety/Reliability, and Commercial Applications

Group 1
- 1 Advanced Composites
- 2 Smart Sensors

Group 2
- 1 Ceramics
- 2 Composite Isogrids

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Conclusions

Health Management Technologies Provide the Most Promising Enhancements Across all Functional Areas

Cost and Schedule are Improved
Risk is Reduced
Many Off-the-Shelf Technologies are Available Today
Safety and Reliability Are Improved
Many VHM Technologies are Applicable to Both Upper Stages and ELVs

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TD16-039
Upper Stage Technical Requirements Document
Source Analysis for Paragraph:
1.22b Technology - Compatibility with Requirements

New technology insertions in the upper stage system shall be planned to comply with all existing system requirements as applicable

Technical Directive 17

Spacecraft Technology Center Transfer

- **Software Packages were Delivered to MSFC**
  - System Engineering Data Base (SEDB) Management System
    - Oracle for Sun SPARC capable of supporting TBD users
    - An option to upgrade Oracle for an additional TBD users
      - RDD100/SD - one (1) copy (Sun IPX workstation)
      - RDD100/RE - one (1) copy (Sun IPX workstation)
      - RDD 100/DVF - one(1) copy (Sun IPX workstation)
        - 4th Dimension for the MacIntosh - TBD copies

- **Training and Installation were Completed at MSFC**

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16. Abstract

Our final report for Phase I addressed the future space transportation needs and requirements based on the current assets, at the time, and their evolution through technology/advanced development using a path and schedule that supported the world leadership role of the United States in a responsible and realistic financial forecast. Always, and foremost, the recommendations placed high values on the safety and success of missions both manned and unmanned through a total quality management philosophy at Martin Marietta.

The second phase of the STV contract involved the use of Technical Directives (TD) to provide short-term support for specialized tasks as required by the COTR. Three of these tasks were performed in parallel with Phase I. These tasks were the Liquid Acquisition Experiment (LACE), Liquid Reorientation Experiment (LIRE), and Expert System for Design, Operation, and Technology Studies (ESDOTS). The results of these TDs were reported in conjunction with the Phase I Final Report

Cost analysis of existing launch systems has demonstrated a need for a new upper stage that will increase America's competitiveness in the global launch services market. To provide a growth path to future exploration class STV's, we must develop near-term low-cost upper stages featuring modularity, portability, scalability, and evolvability.

These recommendations define a program that: (1) leverages ongoing activities to establish a new development environment, (2) develop technologies that benefit the entire life cycle of a system, and (3) result in a scalable hardware platform that provides a growth path to future upper stages.

17. Key Words (Suggested by Author(s))
Space Transfer Vehicle, Advanced Upper Stage, Vehicle Health Management, First Lunar Outpost, Integrated Main Engine, Lunar Transportation System, Propulsion Avionics Module, Avionics Testbed

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